



**SPECIAL REPORT RDMR-AD-16-01**

# **BACKGROUND INFORMATION AND USER'S GUIDE (BIUG) FOR HANDLING QUALITIES REQUIREMENTS FOR MILITARY ROTORCRAFT**

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## 1. SCOPE AND COMPLIANCE

### a. Statement of Requirements

#### 1.1 SCOPE

This specification contains the requirements for the flying and ground handling qualities of rotorcraft.

#### 1.2 APPLICATION

The requirements of this specification are intended to assure that no limitations on flight safety or on the capability to perform intended missions will result from deficiencies in flying qualities. Flying qualities for the rotorcraft shall be in accordance with the provisions of this specification unless specific deviations are authorized by the procuring activity. Additional or alternate special requirements may be specified by the procuring activity. For example, if the form of a requirement should not fit a particular vehicle configuration or control mechanization, the procuring activity may, at its discretion, agree to a modified requirement that will maintain an equivalent degree of acceptability.

Violation of any one requirement is expected to degrade handling qualities. Violation of several individual requirements (e.g., to Level 2) could have a synergistic effect so that the overall handling qualities degrade to Level 3, or worse. Depending on the requirement(s) violated, the contractor will define more extensive and comprehensive flight tests to demonstrate that the overall handling qualities are satisfactory. In any event, final acceptance will be subject to Government flight testing.

### b. Discussion

Figure 1 presents a schematic of the specification structure representing its intended method of use. It is the user's responsibility to define the operational missions and environment. Based on these requirements, the designer develops flight envelopes and determines the required Response-Types. These Response-Types are based on important mission-related factors such as the required degree of divided attention away from actual aircraft control, and the visual environment including the effects of displays and vision-aids, and the aggressiveness of the Mission-Task-Elements. Response-Types and quantitative dynamic response characteristics are defined for hover and low speed (<45 knots) and for forward flight (>45 knots).



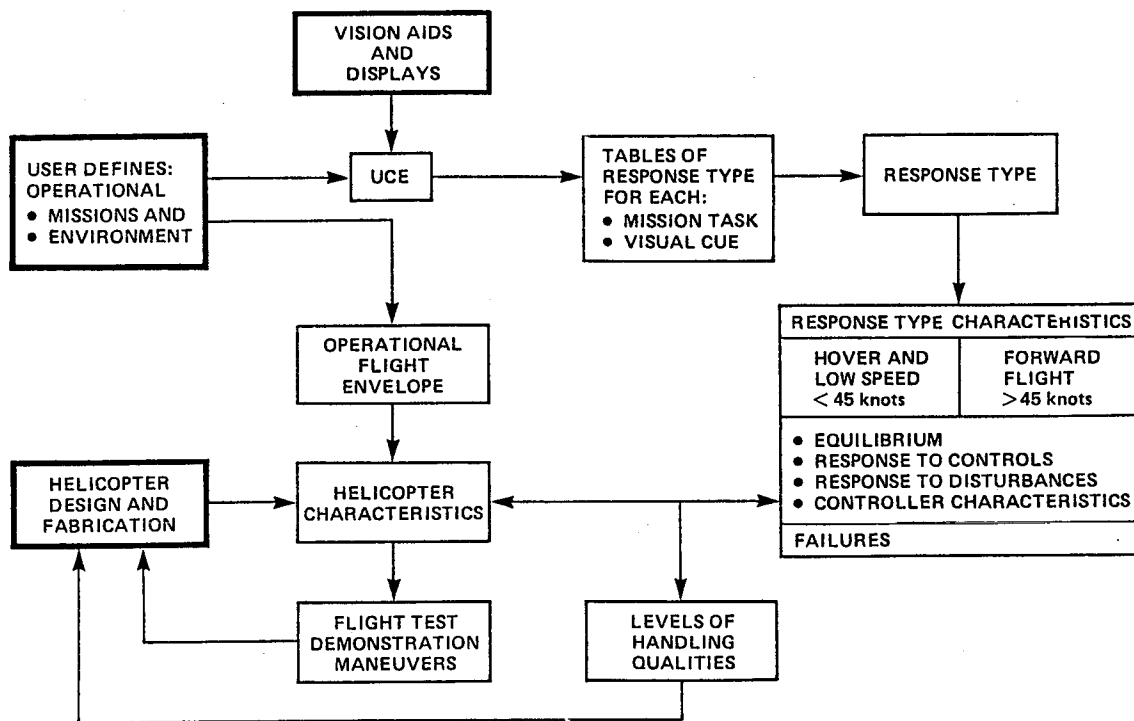


Figure 1(1.1). Schematic Diagram for Specification

Most failures are covered by relating the allowable probability of occurrence to the extent of handling quality degradation involved, but the effects of certain specified failures have to be considered regardless of their probability of occurrence.

The comparison of the rotorcraft characteristics with the requirements provides an analytical assessment of the Level of handling qualities. A comprehensive assessment of any aircraft must include flight evaluations which involve the performance of mission related tasks. To address this, a selection of flight test demonstration maneuvers are included in the specification. These must be accomplished within a specified level of performance and workload (Cooper-Harper Handling Quality Ratings or HQR) as a final demonstration of compliance.

This specification has been developed to include the entire spectrum of operational requirements for the modern helicopter, from the most complex attack missions to be conducted at night and in poor weather, to utility and transport functions which may be restricted to conditions of good visibility or simple IMC only. The price for this generality is that the specification is somewhat more complex than its predecessor, MIL-H-8501A. However, for the user interested in requirements for the simplest missions, many of the more involved requirements can be ignored, notably those related to the usable cue environment (UCE).

Nearly all modern helicopters will have some type of stability augmentation, and many will be controlled entirely by computers through full authority actuators ("fly-by-wire," or "fly-by-light"). To account for this, it has been necessary to abandon many time honored criteria which no longer apply. The replacement criteria are generally a more general version of the old parameters, and represent a more first-principles measure of the response, e.g., bandwidth and phase delay instead of pitch and roll damping. In fact, the old pitch and roll damping parameters are not even valid for unaugmented helicopters with fast responding rotor systems that are strongly affected by inherent time delay.

An important clause in the "application" requirement is: "Additional or alternate special requirements may be specified by the procuring activity. For example, if the form of a requirement should not fit a particular vehicle configuration or control mechanization, the procuring activity may, at its discretion, agree to a modified requirement that will maintain an equivalent degree of acceptability." The purpose of this statement is to emphasize the fact that the writers of the specification do not want to inhibit innovation. It is impossible to cover all of the possibilities for vehicle and/or control configuration; indeed, there is insufficient data to cover all the known possibilities. Some requirements, such as the control force gradient limits, control sensitivity limits, and power-operated control requirements, are written for conventional centerstick and rudder pedal controls. They would clearly be inappropriate for a sidestick controller. In situations such as these, the onus will be on the contractor to demonstrate the acceptability of the system to the procuring agency via flight tests as well as supporting analyses and simulation.

The specification does not directly address the effects of multiple degradations. Certain characteristics which might lead to Level 2 flying qualities in more than one axis of control would be cause for concern that the combined effect could result in a Level 3 aircraft. For example, the following empirically-based formula indicates that the combined effect of degraded handling qualities in more than one axis is a function of the product of the handling qualities ratings (HQRs) in each axis.

$$R_m = 10 + \frac{1}{(-8.3)^{(m-1)}} \prod_{i=1}^m (R_i - 10)$$

Where  $R_m$  = the combined axis pilot rating (HQR)

$R_i$  = the pilot rating in each axis

This product rule was developed in Reference 167, and has been further validated by recent fixed- and moving-base simulations (Reference 197). While the data base is not adequate to incorporate this empirical relationship into the specification, it is given here as a rule-of-thumb to indicate when more extensive flight testing might be required. The second paragraph of 1.2 is intended to draw attention to this possibility.

a. Statement of Requirement

1.3 COMPLIANCE

Compliance with the requirements will be demonstrated using analysis, simulation, and flight test at appropriate milestones during the rotorcraft design and development. In absence of other guidance from the procuring activity, the following will be observed:

- Prior to critical design review - analytical checks shall be computed using available math models (Section 3).
- Prior to first flight - analytical checks shall be accomplished using full nonlinear math models including the feel system and SCAS elements (Section 3), and pilot assessments using flight simulators (Section 4).
- After first flight - flight test verification shall be accomplished for all maneuvers except those deemed too hazardous or impractical. These exceptions will be demonstrated on a flight simulator which has been shown to be valid by comparison with flight test at adjacent conditions (Sections 3 and 4).

b. Discussion

It is desirable to check for specification compliance at several stages during the development of a new aircraft. An assessment of the predicted flying qualities with increasingly complex representations leading up to the critical design review will allow time to take any required corrective action well before hardware is produced. Experience has shown that it is extremely expensive to make changes to a flying prototype, hence the importance of uncovering problems early in the program. It is not expected that a piloted simulation will be available prior to critical design review, and therefore, it will not be possible to demonstrate compliance with the Section 4 maneuvers.

The development of full-authority fly-by-wire flight control systems has become state-of-the-art for fixed-wing aircraft (F-16, A-320, etc.), and is emerging as a mature technology for rotorcraft. Experience has shown that a piloted simulation is essential for safety before first flight with these type of aircraft. Given that such a simulation exists, it would be a natural tool to demonstrate compliance with item 2, i.e., "analytical checks using full nonlinear math models including feel system and SCAS elements." These simulations usually include the flight control system hardware, which would naturally be preferred over math models of the FCS computers, feel system, actuators, etc. It is intended that the simulation be used to show compliance with the quantitative requirements of Section 3, and in most cases would be

run off line (i.e., without a pilot), with additional checks involving "pilot assessments using flight simulators."

For less sophisticated rotorcraft, a simulation such as that described above may not exist. In such cases, it is acceptable to use a simulation which employs math modeling of all control system hardware.

A third and final demonstration of compliance is required using data from the actual aircraft. This will be the most comprehensive assessment of the aircraft flying qualities. It is recognized that demonstration of specification compliance can, in some cases, be hazardous, e.g., testing certain failure cases. Such cases may be checked using a "validated" simulator. Validation is to be achieved at flight conditions which are as close as deemed safe to the case in question by the flight testing activity.

Guidance for compliance with each of the requirements in Section 3 and Section 4 is provided under the heading "Demonstration of Compliance" for each specification paragraph starting with 3.2.

## 2. DEFINITIONS

### a. Statement of Requirements

2.1 Mission-Task-Element (MTE). An element of a mission that can be treated as a handling qualities task. For the purpose of this specification, all proposed missions will be subdivided into Mission-Task-Elements.

2.2 Response-Type. A characterization of the rotorcraft response to a control input in terms of well recognized stability augmentation systems (i.e., Rate, Rate Command/Attitude Hold, etc.). However, it is not necessary to use a stability augmentation system to achieve the specified characteristics.

2.3 Near-Earth Operations. Operations sufficiently close to the ground or fixed objects on the ground, or near water and in the vicinity of ships, oil derricks, etc., that flying is primarily accomplished with reference to outside objects.

2.4 IMC Operations. Operation of the rotorcraft solely with reference to the flight instruments. Occurs when the rotorcraft is clear of all obstacles.

2.5 Extent of Divided Attention Operation. Some requirements are based on the extent to which the pilot must attend to tasks other than flying the rotorcraft.

2.5.1 Fully Attended Operation. The pilot flying the rotorcraft can devote full attention to attitude and flight path control. Requirements for divided attention are minimal.

2.5.2 Divided Attention Operation. The pilot flying the rotorcraft is required to perform non-control-related sidetasks for a moderate period of time.

2.6 Speed Ranges. In the following definitions, ground speed is intended to be the speed with respect to a hover reference which may itself be moving, such as for shipboard operations.

2.6.1 Hover. Hovering flight is defined as all operations occurring at ground speeds less than 15 knots (7.7 m/s).

2.6.2 Low Speed. Low-speed flight is defined as all operations occurring at ground speeds between 15 and 45 knots (7.7 and 23 m/s).

2.6.3 Forward Flight. Forward Flight is defined as all operations with a ground speed greater than 45 knots (23 m/s).

2.7 Step Input. For the purpose of this specification, a step input is defined as a rapid change in the controller force or position from one constant value to another. The input should be made as rapidly as possible without exciting undesirable structural or rotor modes, or approaching any aircraft safety limits. This differs from the classical definition, where the change occurs in zero time.

b. Discussion of 2.6. Speed Ranges

The speed ranges in the specification are divided into Hover, Low Speed, and Forward Flight. These distinctions were made on the basis that the handling qualities tasks (Mission-Task-Elements) are fundamentally different in each of the noted speed ranges. As a practical matter, Hover and Low Speed is treated as a single speed range for most of the specification. The "low speed" category has been retained to account for a few special cases. For example, heading hold is required for Hover, but turn coordination is required for Low Speed (see Paragraph 3.2.10). The term groundspeed is used for simplicity, but is meant to include flight with reference to moving objects, such as ships.

Groundspeed (as opposed to airspeed) was selected as the defining parameter, since it relates directly to most helicopter piloting tasks, and consequently, the control strategy employed by the pilot. The selection of 45 kts groundspeed as the division between Hover/Low Speed and Forward Flight was made on the basis that the pilot's control strategy changes from distinctly helicopter to fixed-wing at approximately 45 kts.

Because of winds, the regions of Hover/Low Speed and Forward Flight will overlap. For example, if the required operational environment includes winds of up to 45 kts, accomplishing Hover/Low Speed Mission- Task-Elements at a groundspeed of 45 kts could result in airspeeds ranging from zero to 90 kts, depending on the direction of flight. Therefore the Hover/Low Speed requirements (Paragraph 3.3) would have to be met at airspeeds up to 90 kts. Likewise, if there is a requirement to accomplish Forward Flight Mission-Task-Elements (Table 2(3.2)) at a groundspeed of 45 kts, in a 45 kt tailwind, the Forward Flight requirements of Paragraph 3.4 must be met at zero airspeed. While this may seem unrealistic, the criteria are based on pilot centered requirements, and not on vehicle aerodynamic considerations. In that light it can be seen that, for this contrived example, the required handling qualities must be tailored to the forward flight tasks, at airspeeds as low as zero.

In summary, the definitions of speed ranges are based on ground-speed, because most helicopter missions involve flight with respect to the ground. An important part of the definition of the operational mission environments, therefore, involves defining the headwind and tailwind components for each Mission-Task-Element selected for Hover/Low Speed, and for Forward Flight (see Tables 1(3.2) and 2(3.2), respectively). The critical flight conditions for specification compliance will often involve limiting combinations of task (groundspeed) and environment (wind), such as noted in the above example.

a. Statement of Requirement

2.8 Levels of Flying Qualities. The acceptability of flying qualities of rotorcraft is quantified herein in terms of "Levels" that are defined for each specific mission task in Figure 1(2.8). Where possible, the requirements of Section 3 are stated in terms of three limiting values of one or more flying qualities parameters. Each value, or combination of values, represents a minimum condition necessary to meet one of the three "Levels" of acceptability. In some cases sufficient simulation or flight test data do not exist to allow the specification of numerical values of a flying quality parameter. In such cases it is not possible to define explicitly the lower boundary of each "Level." These cases are handled by stating the required "Level" of flying qualities for specified piloting tasks, and require compliance by demonstration in flight or via piloted simulation.

b. Discussion

The use of flying qualities Levels has been adapted from existing fixed-wing specifications, i.e., MIL-F-8785C (Reference 66) and MIL-F-83300 (Reference 29). In practice, the Levels serve to reduce the classification of flying qualities from ten categories to three, by grouping the Cooper-Harper handling qualities ratings (HQRs) as shown in Figure 1(2.8).<sup>\*</sup> The value of each handling qualities parameter corresponding to a Level boundary or limit is obtained by fairing lines of constant HQR through experimental data, typically obtained from ground-based or in-flight piloted simulations, but sometimes from operational aircraft. All of the criteria in Section 3 require Level 1 (averaged HQR between 1 and 3.5) for tasks which are normally expected to be required to accomplish the mission for which the rotorcraft has been designed. The specification allows for a degradation in handling qualities to Level 2 or 3 in the event of failures, depending on the calculated probability that the degraded Level will be encountered (see Paragraph 3.1.7). Levels 2 and 3 are also allowed when the rotorcraft is operated beyond the flight envelopes defined by the required operational missions (see Paragraphs 2.9.1 and 2.9.2).

The connection between the flying qualities Levels and the operational missions depends on an accurate definition of the elements of the

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<sup>\*</sup>The use of the standard Cooper-Harper pilot rating scale to define flying qualities levels does not imply that pilot ratings will always have to be obtained in order to comply with the specification; however, it is recommended that pilot ratings be obtained for documentation in demonstrating compliance. The primary use of the scale is to correlate pilot ratings from flying qualities experiments with parameters used in the specification. In a few cases, quantitative parameters may not be available, in which case compliance will have to be accomplished via demonstration using the Figure 1(2.8) scale.

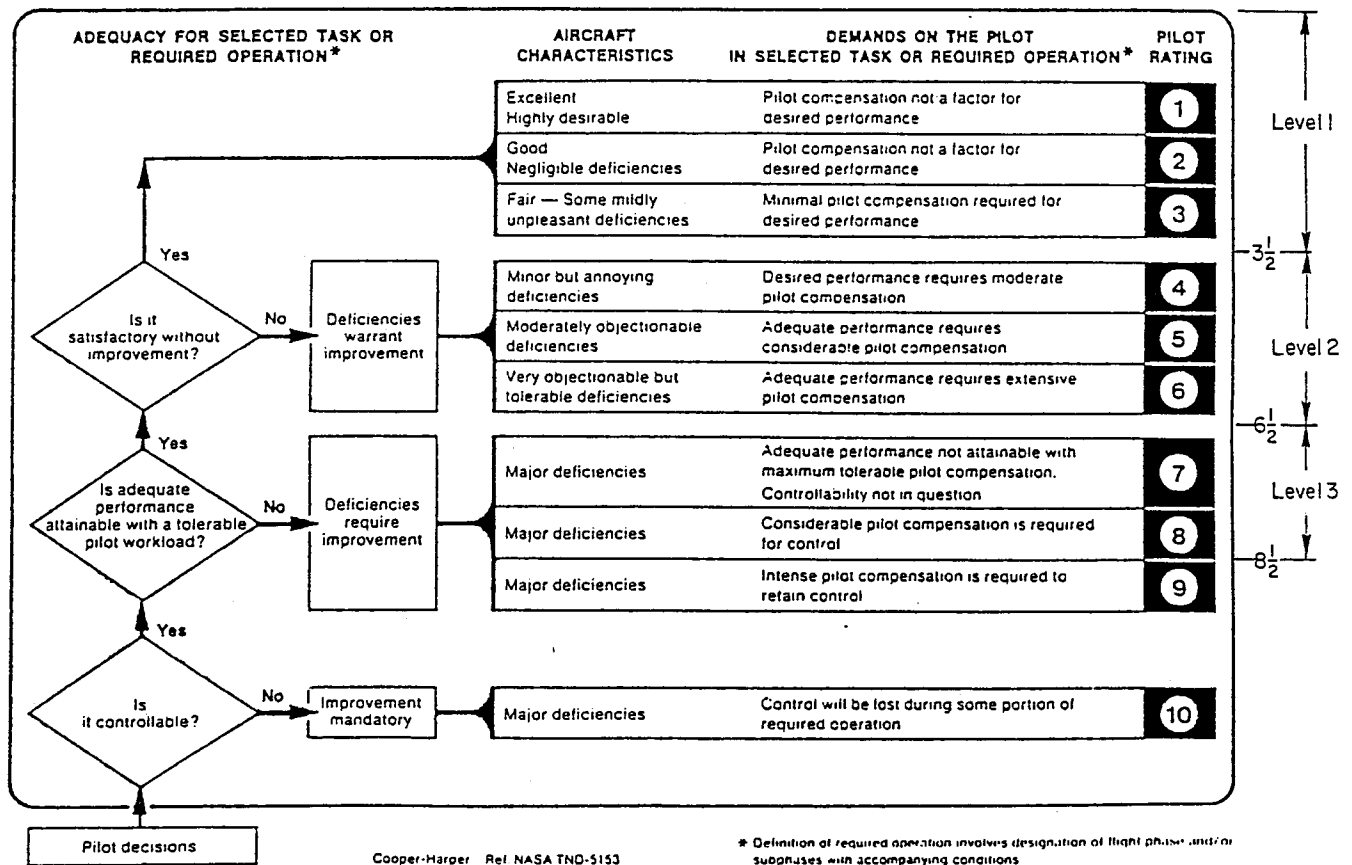


Figure 1(2.8). Definition of Flying Qualities Levels

missions in the context of handling qualities tasks.\* These are formally defined as "Mission-Task-Elements" (MTEs) in this specification. Thus, a "Level of flying qualities" refers specifically to a range of HQRs for one or more specified MTEs.

It is physically insightful to interpret the Levels in terms of the ability to accomplish the required missions. The following interpretations are slightly revised versions of the Level definitions used in the fixed-wing specifications (References 29 and 66).\*\*

\*The terms "flying qualities" and "handling qualities" are used interchangeably in this specification background document.

\*\*This is in contrast to the References 29 and 66 fixed-wing specifications which define the Levels directly in terms of descriptive phrases similar to those shown here (which are unrelated to the Figure 1(2.8) scale). The assumption that the semantics of these mission oriented descriptors are equivalent to the ranges of HQRs in Figure 1(2.8) seems reasonable for physical insight, but questionable when employed as a fundamental component of the specification.



- Level 1 -- Satisfactory without improvement [see Figure 1(2.8)], and therefore handling qualities are adequate for the MTE.
- Level 2 -- Deficiencies warrant improvement, but do not require improvement [see Figure 1(2.8)]. Interpretation is that handling qualities are adequate to complete the MTE, but with some increase in workload or reduced task performance, or both.
- Level 3 -- Deficiencies require improvement [see Figure 1(2.8)] and control can be retained with considerable compensation. Interpreted to imply that the MTE cannot be completed.

Strictly speaking, "proper" use of the Figure 1(2.8) scale dictates that only integer ratings be assigned by the evaluation pilot. However, rating scale research reported in Reference 56 shows that the scale is linear for ratings up to about 7. Hence, it is common practice for pilots to assign half ratings within a given Level; half ratings are never assigned between the Levels, i.e., 3.5 and 6.5 are not allowed. The experimental data upon which the criterion boundaries are based typically involve several pilot subjects. A Level 1 or 2 boundary is defined when the pilot subjects are roughly evenly split, i.e., approximately half rate on one side, and half on the other. Such a situation is manifested as an average rating of 3.5 or 6.5 to define the limits. A boundary may also be defined by fairing lines of constant pilot rating (3.5 and 6.5) through the data, i.e., halfway between ratings of 3 and 4, and 6 and 7.

The Level 3 boundary has historically been set by HQR=9.5. However, in this specification the Level 3 boundary is set at 8.5. There are two reasons for this:

- If the boundary is based on the fact that the pilots are evenly divided between an HQR of 9 and 10, loss of control is expected for half the pilots. The purpose of Level 3 is to guarantee that control will not be lost.
- An HQR of 9 indicates that intense pilot compensation is required to retain control. Such a descriptor is not felt to be an adequate guarantee of a controllable rotorcraft.

a. Statement of Requirement

2.9 FLIGHT ENVELOPES

2.9.1 Operational Flight Envelopes (OFE). The Operational Flight Envelopes define the boundaries within which the rotorcraft must be capable of operating in order to accomplish the operational missions of 3.1.1. These envelopes shall be defined in terms of combinations of airspeed, altitude, load factor, rate-of-climb, side velocity, and any other parameters specified by the procuring activity, as necessary to accomplish the operational missions. Any warnings or indications of limiting or dangerous flight conditions, required by Paragraph 3.1.8, shall occur outside the OFEs.

b. Discussion

The Operational Flight Envelopes are intended to define regions bounded by maximum and minimum values of rotorcraft performance parameters necessary to accomplish the missions for which the rotorcraft is being designed. It follows that the user would usually be expected to take the primary responsibility for defining the Operational Flight Envelopes. If the user does not explicitly define the OFEs, it will be up to the manufacturers to specify these envelopes. In any case, they may be subject to negotiation between the contractor and user as the design develops.

While some specific parameters are called out as obvious candidates for defining the Operational Flight Envelopes, considerable flexibility is allowed by adding "any other parameters" that define limits. This is done in recognition of the fact that certain missions may introduce other parameters such as slope landing gradient, etc.

The Operational and Service Flight Envelopes may need to consider both steady state and transient values. An example of Operational Flight Envelope parameters defined in terms of both steady and transient load factor is given in Figure 1, taken from unpublished and unofficial recommendations based on air-combat maneuvering tests.

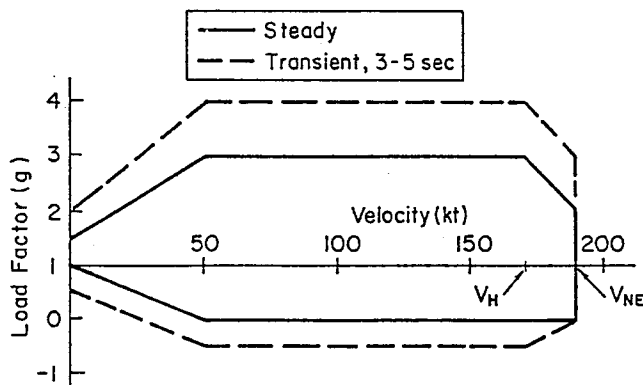


Figure 1(2.9.1). Example of Operational Flight Envelope Defined in Terms of Transient and Steady Load Factor, and Airspeed

a. Statement of Requirement

2.9.2 Service Flight Envelopes (SFE). The Service Flight Envelopes are derived from aircraft limits as distinguished from mission requirements. Based on a list of Normal States (obtained from those tabulated in Paragraph 3.1.6.1), found to be most critical in terms of rotorcraft limits, the contractor shall establish SFEs. These envelopes shall be expressed in terms of the parameters used to define the OFEs, plus any additional parameters deemed necessary to define the appropriate limits. The inner boundaries of the SFEs are defined as coincident with the outer boundaries of the OFEs. The outer boundaries of the SFEs are defined by one or more of the following: uncommanded aircraft motions, or structural, engine/power-train, or rotor system limits. The magnitude of the differences between the inner and outer boundaries of the SFEs shall be based on the guarantee of adequate margins between operations to complete the specified missions, and rotorcraft limits as required by Paragraph 3.1.8.

b. Discussion

The intent of the Service Flight Envelope (SFE) is to insure that reasonable flying qualities which insure at least "adequate performance" (i.e., Level 2) will exist between the limiting flight conditions required to accomplish the operational missions (i.e., the Operational Flight Envelopes or OFEs), and aircraft limits. If unsatisfactory flying qualities (i.e., Level 3) are encountered in the SFEs, the resulting lack of precision of control could cause the pilot to inadvertently exceed the limit. For example, assume that an OFE boundary is set at 0.5g, and that at 0g excessive rotor flapping is known to cause the rotor to contact aircraft structure (mast bumping). Assuming a margin of 0.5g is adequate to satisfy the requirements for "warning and indication of rotorcraft limits," Paragraph 3.1.8.1, the rotorcraft will pass the specification. The SFE would exist between 0.5g and 0.0g where an aircraft limit is encountered (structural failure). If worse than Level 2 flying qualities were to exist in the SFE, and a pilot were to maneuver beyond the OFE, the precision of control could be inadequate to insure avoidance of the 0g limit. With the present requirement, the pilot is assured that at least Level 2 flying qualities will be available to control the aircraft in such a way as to avoid the zero g limit condition (i.e., adequate performance is assured). Another example of the SFE would be the airspeed range between retreating blade stall and the never-exceed speed.

Most rotorcraft limits are well understood by the helicopter manufacturers. It is not the intention of this requirement to make "busy work" by requiring the tabulation of helicopter limits for all the normal states defined in Paragraph 3.1.6.1. Only those states known to be critical must be considered, for example, aft regardless c.g., certain loadings, maximum gross weight, etc. In some cases, it will not be possible to reach rotorcraft limits because of human factors limitations; for example, pilot disorientation in rolls accomplished at elevated load factor.

The rotorcraft limits can take on distinctly different forms. For example, they can consist of an uncommanded roll such as with retreating blade stall, or structural failure such as with mast bumping. It is clearly more important that the pilot be able to avoid the latter type of limit which is catastrophic, as opposed to the former which is easily recoverable if recognized. Catastrophic limits tend to have more margin, and hence a larger SFE. Testing to the limit of such an SFE is clearly not advisable. However, it should be demonstrated, in flight, that small excursions into the SFE do not result in a rapid degradation in flying qualities. Larger excursions into such an SFE should be accomplished using an appropriate math model to insure compliance with the quantitative requirements for Level 2. Limits defined by engine life should receive similar treatment, since it would be unacceptable to risk the destruction of an engine or transmission to demonstrate compliance with the flying qualities specification.

An exceedance of an outer limit of an SFE defined by uncommanded aircraft motions tends to be recoverable, e.g., retreating blade stall, settling with power, etc. However, an interpretation of the recovery techniques in terms of conventional handling qualities metrics (Cooper-Harper HQRs) tends to be ineffective, primarily because the pilot actions are open-loop, and they either work or they don't. Therefore, it is simply required that such an exceedance be recoverable back to the SFE without undue pilot skill (see Paragraph 3.1.6.3). Considerable discussion has occurred as to the need for a flight envelope to define such a recoverable region. It has been concluded that the definition of such regions is not practical, and would only add needless complexity to the specification.

### 3. REQUIREMENTS

#### a. Statement of Requirement

##### 3.1 GENERAL

3.1.1 Operational Missions. The procuring activity will define the operational missions and will specify the Mission-Task-Elements to be considered by the contractor in designing the rotorcraft to meet the flying qualities requirements of this specification. These Mission-Task-Elements will include the entire spectrum of intended operational usage and will in most cases be selected from those listed in Table 1(3.2) or 2(3.2). The procuring activity will specify any Degraded Visual Environment (DVE), the atmospheric disturbances, weather, performance criterion, and degree of divided attention operation (see Paragraph 2.5) to be considered.

#### b. Discussion

This specification is based on the premise that the procuring activity will define the operational missions in detail, and will specify the Mission-Task-Elements to be considered. In the event that the procuring activity does not specify the Mission-Task-Elements, it will be the contractor's responsibility to analyze the user-defined missions, and to select the MTEs required to accomplish those missions. The procuring activity and contractor must reach a mutual agreement on the final list of MTEs as these will play a significant role in defining the handling qualities of the rotorcraft. For example, if all the MTEs in Table 1(3.2) are specified, the rotorcraft will be required to have multiple augmentation modes. However, if a less sophisticated rotorcraft is required, a careful selection of MTEs could result in a single-mode Rate Response-Type which may be satisfied without any augmentation system.

The procuring activity will specify the Degraded Visual Environment (DVE) in which each of the MTEs must be performed. In the case of combat helicopters, this will probably include operations at night and in bad weather in the NOE environment. This implies that all of the pilot's visual cues must be derived from displays and vision aids. Factors which affect the performance of such vision aids are therefore an important component of the definition of the DVE. For example, the following definitions of DVE have been suggested for the LHX procurement.

- For Night Vision Goggles, the DVE is defined as a moonless, overcast night, with a defined ceiling.
- For Forward Looking Infrared, the DVE is defined as a night in rain or fog, following a day with not more than two hours of sunshine.

Night is defined as two hours after sunset to two hours before sunrise. The locale for testing is defined as a flat, open area such as a grass field or airport, clear of artificial lighting.

It may not be practical to require the same level of aggressiveness for day VMC as for NOE operations at night and in poor weather, and hence, the definitions of some MTEs may be modified for degraded usable cue environment. For example, the maneuvers in Section 4.4 and 4.5 are considerably less aggressive than those of 4.1 and 4.2. The maximum recommended forward speed with two generations of night vision goggles in the NOE environment is plotted in Figure 1 as a function of available illumination (taken from pilot's training manual). If such forward speeds were specified in the Request for Proposals (RFP), it would have an impact on the way the maneuvering envelopes in Paragraph 3.2.4 would be constructed. If insufficient knowledge and/or data is available at the time of the RFP (to generate plots such as shown in Figure 1), the user will have to negotiate with the manufacturer during the development program as to the capability of available displays versus the mission requirements. The size of the envelopes (defined in Paragraph 3.2.4), and the UCE metric (from Paragraph 3.2.2.1), are intended to provide a basis for quantifying the effectiveness of the displays.

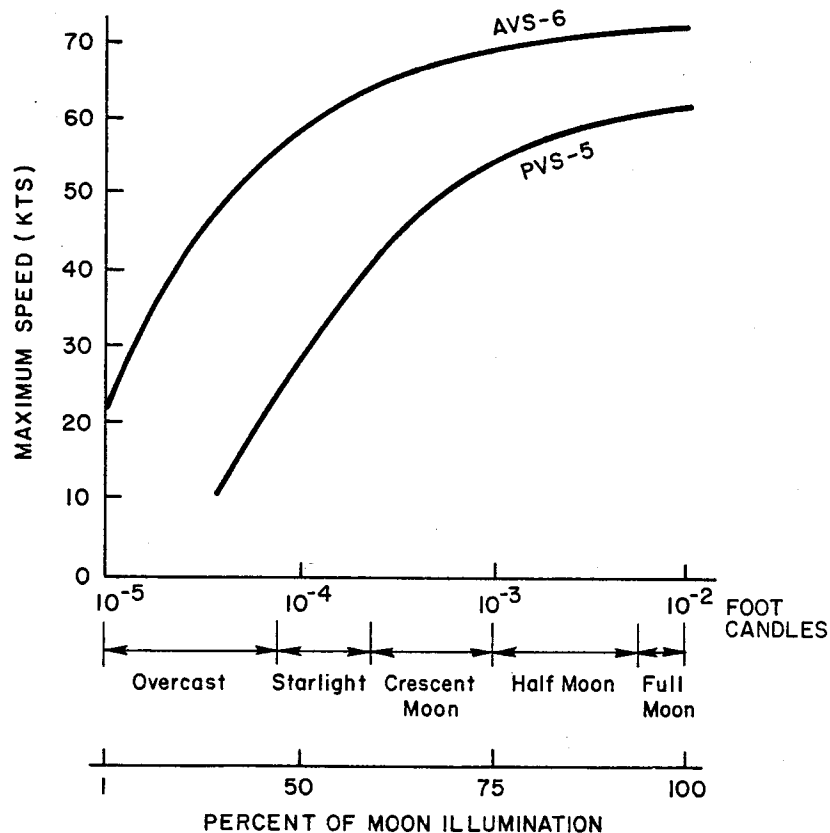


Figure 1(3.1.1). Example of Performance Limits When Using a Pilot Vision Aid (Night Vision Goggles) -- Data Taken From Army Training Manual FM1-204

a. Statement of Requirement

3.1.1.1 Multi-Crew Rotorcraft. Unless otherwise stated, all requirements shall apply for the primary pilot station. The procuring activity will define the Mission-Task-Elements, Degraded Visual Environment, degree of divided attention, and Level of Flying Qualities that are applicable to any other pilot stations.

b. Discussion

This paragraph has been included to account for the possibility that the flying qualities may be different at secondary pilot stations, due to cockpit visibility, displays, etc.

a. Statement of Requirements

3.1.2 Loadings. The envelope of center-of-gravity and weight for each Mission-Task-Element shall be specified by the contractor. In addition, the contractor shall specify the maximum center-of-gravity excursion attainable through failure in systems or components for each Mission-Task-Element.

3.1.3 Moments and Products of Inertia. The contractor shall define the moments and products of inertia of the rotorcraft associated with all loadings of Paragraph 3.1.2. The requirements of this specification shall apply for all moments and products of inertia so defined.

3.1.4 External Stores. The requirements of this specification shall apply for all combinations of external stores and slung loads required by the operational missions. The effects of external stores on the weight, moments of inertia, center-of-gravity position, and aerodynamic characteristics of the rotorcraft shall be considered for each Mission-Task-Element. When the stores contain expendable loads, the requirements of this specification apply throughout the range of store loadings.

b. Discussion

The loading of a rotorcraft is determined by what is in (internal loading), and attached to (external loading), the rotorcraft. The parameters that define different characteristics of the loading are weight, center-of-gravity position, and moments and products of inertia. External stores affect all these parameters and also affect aerodynamic coefficients.

Only permissible center-of-gravity positions need be considered for Rotorcraft Normal States. But fuel sequencing and transfer failures or malperformance that cause the center of gravity to fall outside the established limits are to be considered as Rotorcraft Failure States. The worst possible cases that are not approved Special Failure States (3.1.7.5) must be examined.

Since the requirements apply over the full range of service loadings, effects of fuel slosh and shifting should be taken into account in design. Balance, controllability, and airframe and structural dynamic characteristics may be affected. Factors to consider are fuel sequencing, in-flight refueling if applicable, fuel shifting during acceleration or deceleration, and all arrangements of variable, disposable and removable items required for each operational mission.

The procuring activity may elect to specify a growth margin in c.g. travel to allow for uncertainties in weight distribution, stability level and other design factors, and for possible future variations in operational loading and use.



In determining the range of store loadings to be specified in the contract, the procuring activity should consider such factors as store mixes, possible points of attachment, and asymmetries -- initial, after each pass, and the result of failure to release. The contractor may find it necessary to propose limitations on store loading to avoid excessive design penalties.

The designer should attempt to assure that there are no restrictions on store loading, within the range of design stores. However, it is recognized that occasionally this goal will be impractical, since it may be impossible to avoid exceeding aircraft limits without incurring excessive design penalties. Then, insofar as considerations such as standardized stores permit, it should be made physically impossible to violate necessary store loading restrictions. If this too should not be practical, the contractor should submit both an analysis of the effects on flying qualities of violating the restrictions and an estimate of the likelihood that the restrictions will be exceeded.

a. Statement of Requirement

3.1.5 Configurations. A configuration is defined by the positions and adjustments of the various selectors and controls available to the crew, except for longitudinal, lateral, directional, vertical, power, and trim controls. The selected configurations to be examined must consist of those required for performance of the operational missions of Paragraph 3.1.1. Additional configurations to be investigated may be defined by the procuring activity. Control positions which activate stability augmentation necessary to meet the requirements of this specification are considered to be always ON unless otherwise specified.

b. Discussion

The settings of such controls as speed brakes and landing gear are related uniquely to each aircraft design. The specification requires that the configurations to be examined shall be those required for performance and mission accomplishment. For example, the mast angle of a tilt-rotor design would represent a configuration variable. The position of roll, pitch, yaw and trim controls, and the thrust magnitude control, are not included in the definition of configuration since the positions of these controls are usually either specified in the individual requirements or determined by the specified flight conditions.

a. Statement of Requirements

3.1.6 Normal States. The Rotorcraft Normal States (no component or system failures) are defined by the selected configurations together with the functional status of each of the rotorcraft components or systems, thrust magnitude, weight, moments and products of inertia, center-of-gravity position, and external store complement. The trim setting and the positions of the pitch, roll, and yaw controls are not included in the definition of rotorcraft State since they are often specified in the requirements. The position of the thrust magnitude control shall not be considered an element of the rotorcraft State when the thrust magnitude is specified in a requirement.

3.1.6.1 Tabulation of Normal States. The contractor shall define those Normal States which represent the characteristics of the rotorcraft throughout the OFEs and SFEs. Certain items -- such as configurations, weight, moments of inertia, center-of-gravity position, rotor-tilt angle, or power setting -- may vary continuously over a range of values during a Mission-Task-Element. The contractor shall replace this continuous variation by a limited number of combinations of the parameters in question. These will be treated as specific States and will include the most critical values and the extremes encountered during the required Mission-Task-Element in question.

3.1.6.2 Allowable Levels for Normal States. The allowable Levels in the Operational and Service Flight Envelopes are specified in Table 1(3.1).

TABLE 1(3.1). ALLOWABLE LEVELS FOR ROTORCRAFT NORMAL STATES

	OPERATIONAL FLIGHT ENVELOPE	SERVICE FLIGHT ENVELOPE
Minimum Handling Qualities	Level 1	Level 2

3.1.6.3 Flight Beyond the Service Flight Envelopes. Flight beyond the Service Flight Envelope that does not involve structural failure, or unrecoverable loss of rotor RPM, shall be recoverable to the SFE without undue pilot skill. If such an excursion involves an engine failure, the requirements of 3.7.2 or 3.7.3 apply.

b. Discussion

The OFEs and SFEs are defined in Section 2.9. A required Level of flying qualities in each of these envelopes is formally introduced herein. The rationale for requiring Level 2 in the SFE is given in the discussion of Paragraph 2.9.2.

a. Statement of Requirement

3.1.7 Rotorcraft Failures. When one or more Rotorcraft Failure States exist, a degradation in rotorcraft handling qualities is permitted. The required tabulation and definition of failure states as well as the allowable handling qualities degradations are specified in this paragraph.

b. Discussion

The allowable handling qualities following a failure are specified as follows:

- Tabulate all possible failures (3.1.7.1).
- Determine whether failures will be evaluated based on probability calculations (3.1.7.2).
- Perform required calculations (3.1.7.3, 3.1.7.4).
- Identify special failure states (3.1.7.5).
- Calculate the effect of transients following each type of failure (3.1.7.6).

Each of these items is discussed in the noted paragraphs.

a. Statement of Requirement

3.1.7.1 Tabulation of Failure States. The contractor shall tabulate all Rotorcraft Failure States, which consist of Rotorcraft Normal States modified by one or more malfunctions in rotorcraft components or systems which affect rotorcraft response or UCE (Paragraph 3.2.2.1). Each mode of failure shall be considered.

b. Discussion

There is more to determining Failure States than just considering each component failure in turn. Two other types of effects must be considered. First, failure of one component in a certain mode may itself induce other failures in the system, so failure propagation must be investigated. Second, one event may cause loss of more than one part of the system. Events of "unlikely" origin from recent flight experience (not limited to rotary wing) are listed as illustrations:

- Failure of one bracket that held lines from both hydraulic systems led to loss of integrity of both systems.
- An extinguishable fire that burned through lines from all hydraulic systems, that were routed through the same compartment.
- Spilled coffee on the pilot's console that shorted out all electrical systems; lightning strikes might do this, too.
- A loose nut (too thick a washer was used, so the self-locking threads were not engaged) which shorted all three stability augmentation channels of a triply redundant system.
- Undetected impurities in a batch of potting compound used in packaging stability augmentation system components; all affected channels shorted out at the high temperatures of supersonic flight, after passing ground check-out.
- Complicated ground checkout equipment and lengthy procedures that were impractical to use very frequently on the flight line, resulting in long flight times between flight control system electronics checks.

The insidious nature of possible troubles emphasizes the need for caution in design application.

In discussing redundant systems, it is axiomatic that the whole system must be redundant. A recent design used multiple- redundant SAS which required environmental control for the electronic components; the environmental control system was not redundant. Thus the complex

multiple-redundant SAS could have been put out of action by any failure of the air conditioning equipment.

The interdependency of the flight control system and pilot vision aids and displays (Paragraph 3.2.2) must be considered in this requirement. The loss of a vision aid or display in NOE flight in poor weather and/or at night will have a significant negative impact on the flying qualities. Therefore, it is important that vision aid and displays which are necessary to obtain the UCE values used in Table 1(3.2) must be included in the tabulation of rotorcraft failure states.

a. Statement of Requirement

3.1.7.2 Methods of Compliance. Two methods are presented in order to provide the procuring agency with an option of using probability calculations or considering Specific Failures. The first option involves the following procedures:

- 1) Determine the degree of handling qualities degradation associated with the transient for each Rotorcraft Failure State (3.1.7.6).
- 2) Determine the degree of handling qualities degradation associated with the subsequent steady Rotorcraft Failure State.
- 3) Calculate the probability of encountering each identified Rotorcraft Failure State per flight hour.
- 4) Compute the total probability of encountering Levels 2 and 3 flying qualities in the Operational and Service Flight Envelopes. This total will be the sum of the rate of each failure only if the failures are statistically independent.

The second option assumes that certain failures or combinations of failures will occur regardless of their probability of failure. Steps 3 and 4 of the first option are therefore not performed. The contractor and procuring agency shall mutually agree on which failures in Paragraph 3.1.7.1 shall be treated as "specific failures."

b. Discussion

Two options are included in the specification to allow the procuring agency to quantitatively stipulate the allowable degradation in handling qualities due to failures. The first option has been derived from the fixed wing specifications (MIL-F-8785C and MIL-F-83300) and involves calculating the probabilities of encountering a degraded handling qualities Level. Option 2 assumes that a specific failure will occur with a probability of 1, and assigns a required handling qualities Level to that failure. With the exception of certain Specific Failures designated in Paragraph 3.7, all failures will be treated via Option 1 unless Option 2 is explicitly called out by the procuring activity.

A four step analysis procedure is defined by this requirement, where only Steps 1 and 2 apply for Option 2 (Specific Failures). Step 1 involves calculating the degree of handling qualities degradation due to the transient from the unfailed to the failed state. The methodology for making these calculations is given in Paragraph 3.1.7.6. Step 2 requires that the handling qualities Level be obtained for the rotorcraft in the presence of the failure. In most cases this will involve calculations (or tests) to determine the values of the criterion parameters (i.e., bandwidth, maximum angular rates, etc.) in the failed

state, and comparison of these parameters with the boundaries and limits in Sections 3.3 and 3.4. Failures designated as Specific Failures are compared with the stated design goals. The rest of the failures are subject to a calculation of their probability of occurrence as required by Step 3. Finally, assuming that the failures are statistically independent, the total probabilities of encountering Levels 2 and 3 are calculated by summing the rate of each failure in Step 4.



a. Statement of Requirement

3.1.7.3 Option 1 -- Allowable Levels Based on Probability.  
A degradation in flying qualities, due to the Rotorcraft Failure States (3.1.7.1), is permitted only if the probability of encountering the degraded Level is sufficiently small. These probabilities shall be less than the values shown in Table 2(3.1).

TABLE 2(3.1). LEVELS FOR ROTORCRAFT FAILURE STATES

PROBABILITY OF ENCOUNTERING	WITHIN OPERATIONAL FLIGHT ENVELOPE	WITHIN SERVICE FLIGHT ENVELOPE
Level 2 after failure	$< 2.5 \times 10^{-3}$ per flight hr	
Level 3 after failure	$< 2.5 \times 10^{-5}$ per flight hr	$< 2.5 \times 10^{-3}$ per flight hr

b. Discussion

The similar MIL-F-8785C (Reference 66) requirement specified failure probabilities as a function of number of flights, rather than flight hours. A typical flight time of four hours was used for the 8785C numbers. The Table 2(3.1) recommended probabilities are now a function of flight hour, simply dividing the 8785C numbers by 4. This assures that the requirements are constant with operational mission time, where in the past the requirements were easier to meet for aircraft with very short operational flight times and harder to meet for aircraft with very long flights.

Limited degradation of flying qualities (e.g., Level 1 to Level 2) is acceptable if the combined probability of such degradation is small. If the probability is high, then no degradation beyond the Level required for Normal States is acceptable after the failure occurs. Another way of stating this is that in the Operational Envelope the probability of encountering Level 2 any time at all on a given four-hour flight should not, for example, according to the Table 2(3.1) requirements, exceed  $10^{-2}$ , and the probability of encountering Level 3 on any portion of the flight should not exceed  $10^{-4}$ . Somewhat reduced requirements should also be imposed for flight within the Service Flight Envelope for both Normal and Failure States.

This requirement provides a sound analytical method for accounting for the effects of failures. It serves to force a detailed failure mode and effect analysis from the flying qualities standpoint. Such an analysis is vital as both system complexity and the number of design options increase.

The values of Table 2(3.1) are reasonable, based on experience with past fixed-wing aircraft.\* To illustrate this, the following table presents actual control system failure information for several piloted aircraft:\*\*

<u>SYSTEM</u>	<u>MEAN TIME BETWEEN MALFUNCTIONS (MTBM)</u>
F-101B	86 hours
F-104	300 hours
F-105D (Flight Control plus Electronics)	14 hours
E-1B	185 hours
B-58	20 hours

Unfortunately the flying qualities effects of the reported failures are not given along with the above data. Reference 82 indicates, however, that the mean time between "critical" failures is about five times the MTBM. If "critical" failures are ones that degrade one or more flying qualities to Level 2, then for a typical average flight time of four hours:

$P(\text{Level } 2) = \text{Probability of encountering Level 2 flying qualities during a single flight}$

$$= 1 - e^{-4/[5(\text{MTBM})]}$$

$$\approx \frac{4}{5(\text{MTBM})}$$

This yields:	<u>System</u>	<u>P(Level 2) per 4 hour flight</u>
	F-101B	0.0093
	F-104	0.0027
	F-105D	0.057
	E-1B	0.0043
	B-58	0.040

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\*Only fixed-wing data are available. However, these failure rates should apply to rotorcraft as well.

\*\*This discussion is taken from Reference 82.

These data indicate that all systems, with the exceptions of the F-105D (where electronic components represented in the data might not degrade flying qualities upon failure) and the B-58, meet the requirement for  $P(\text{Level } 2) < 10^{-2}$  (or one out of a hundred flights). Numbers of roughly the same magnitude have been used for both American and Anglo-French supersonic transport design (see Reference 82).

A more comprehensive analysis was conducted on the F-4 by the Air Force Flight Dynamics Laboratory. The level of degradation used in this report was based on whether or not the failure resulted in an abort. Failures without abort were considered degraded to Level 2, and those which caused an abort were considered degraded to Level 3. The results showed that the F-4 handling qualities, in an average 2.57 hour flight, will be degraded to Level 2 on an average of 0.0167 per flight hour, and to Level 3 a maximum of 0.00082 per flight hour. This does not meet the flying qualities requirements for fixed or rotary wing aircraft.

A similar comparison can be made between accident loss rates and the requirement for  $P(\text{Level } 3) < 2.5 \times 10^{-5}/\text{flight hour}$ . It should be emphasized that Level 3 as defined in Paragraph 2.8 and in the requirements represents a safe aircraft to fly. However, due to a lack of knowledge in some instances, especially when many flying qualities are degraded at once, the Level 3 boundaries are, while not necessary totally safe, at least "safety related." Reference 82 indicates the following aircraft accident loss rates during 1967. Also shown is the probability of aircraft loss, per 4-hour flight, for an assumed exponential loss distribution.

<u>AIRCRAFT</u>	<u>1967 LOSS RATE (LOSSES/100,000 HR)</u>	<u>PROBABILITY OF LOSS DURING 4-HOUR FLIGHT</u>
F-101	15	$6 \times 10^{-4}$
F-104	23	$9.2 \times 10^{-4}$
F-105	17	$6.8 \times 10^{-4}$
F-106	10	$4 \times 10^{-4}$
F-4	14.1	$5.64 \times 10^{-4}$
F-102	9	$3.6 \times 10^{-4}$
F-100	<u>10</u>	<u><math>4 \times 10^{-4}</math></u>
	Avg. 14	Avg. $5.6 \times 10^{-4}$

If Level 3 represented a safety problem, which it conservatively does not, then the allowable  $10^{-4}$  probability of encounter per 4-hour flight would account for about 1/4 to 1/9 of the total probability of aircraft loss. That is, flying-qualities-oriented losses would represent about 1/4 to 1/9 of all losses. Other losses could be due to engine failures, etc. Based on experience, therefore, the Table 2(3.1) value is reasonable.

The probability calculations required by this paragraph are designed to expose potential problem areas that may not be obvious from a less rigorous failure analysis. It is obvious from the above discussion that the numerical limits in Table 2(3.1) are not well supported by data, and are more a measure of what is considered to be "reasonable". If after an exhaustive analysis, the contractor finds that it is not possible to meet the specified limits, it is expected that more relaxed values may be negotiated with the procuring activity. However, it will be up to the contractor to provide detailed supporting data which leads to a probability value greater than that specified in Table 2(3.1). This data should take the form of arguments which justify a higher probability based on acceptable safety and mission effectiveness, and/or which show why such probabilities cannot be achieved without incurring excessive cost or weight penalties.

a. Statement of Requirement

3.1.7.4 Option 2 -- Allowable Levels for Specific Failures.

The requirements on the effects of Specific Failures shall be met on the basis that the failure has actually occurred. The allowable Level of flying qualities for each Specific Failure shall be specified by the procuring activity. Alternatively, the procuring activity may specify specific piloting tasks and associated performance requirements in the failed state. As a minimum, the failures in Paragraph 3.7 shall be treated as Specific Failures.

b. Discussion

This approach assumes that a given component, or series of components, will fail. Furthermore, it is assumed the failures will occur in the most critical flight condition; for example, a roll SAS failure during an aggressive high speed NOE slalom. Based on the comments made by users of MIL-F-8785B (see Reference 1), this approach is a reflection of the way things are often being done with fixed-wing aircraft.

The selection of failures to be considered is based on preliminary estimates of handling quality degradations. For example, the loss of one to three channels of a quad-redundant SCAS may have no effect. Conversely, the failure of a single-channel limited authority damper would warrant a complete analysis and/or simulation to determine the resulting degradation in flying qualities.

Because the selection of failure modes is highly dependent on the details of the design, close coordination between the contractor and the procuring activity will be required when identifying failure modes to be analyzed. Indeed, this is currently standard practice.

In most cases, demonstration of compliance will consist of showing that the flying qualities parameters in question fall within the prescribed boundaries. Finally, the combined effects of failures and turbulence should be investigated utilizing a piloted simulation.

a. Statement of Requirement

3.1.7.5 Rotorcraft Special Failure States. Certain components, systems, or combinations thereof may have extremely remote probability of failure during a given flight. These failure probabilities may, in turn, be very difficult to predict with any degree of accuracy. Special Failure States of this type need not be considered in complying with the requirements of Section 3 if justification for considering the Failure States as Special is submitted by the contractor and approved by the procuring activity.

b. Discussion

Regardless of the degree of redundancy, there remains a finite probability that all redundant paths will fail. A point of diminishing returns will be reached, beyond which the gains of additional channels are not worth the associated penalties.

Several categories of Special Failures States can be distinguished. Certain items might be approved more or less categorically:

- Control-stick fracture.
- Basic airframe, main rotor, or control-surface structural failure.
- Dual mechanical failures in general.

In most cases, a considerable amount of engineering judgment will influence the procuring activity's decision to allow or disallow a proposed Rotorcraft Special Failure State. Probabilities that are extremely remote are exceptionally difficult to predict accurately. Judgments will weigh consequences against feasibility of improvement or alternatives, and against projected ability to keep high standards throughout design, qualification, production, use, and maintenance.

Note that the approval of Rotorcraft Special Failure States is at the discretion of the procuring activity. In conjunction with certain requirements that must be met regardless of component or equipment status, granting or refusing approval can be used as desired to require a level of stability for the basic airframe, to rule out fly-by-wire control systems, to demand consideration of vulnerability, or even to rule out a type of configuration. For example, a propeller pitch control failure on a dual rotor helicopter which uses differential thrust from fore and aft or left and right hand side rotors to provide pitch or roll control in hover, will result in loss of control; clearly no requirements can then be met, and the configuration is excluded, unless the pitch control failure is allowed as a special failure. The procuring activity should state the considerations to be imposed, as completely as possible at the outset; but it is evident that many decisions must be made subjectively and many will be influenced by the specific design.

a. Statement of Requirement

3.1.7.6 Transients Following Failures. The transient following a failure or combination of flight control system failures shall be recoverable to a safe steady flight condition without exceptional piloting skill. The contractor shall conduct tests to define the transients for comparison with the values in Table 3(3.1), and the results of these tests shall be made available to the procuring activity. For rotorcraft without failure warning and cueing devices, the perturbations encountered will not exceed the limits of Table 3(3.1).

TABLE 3(3.1). TRANSIENTS DUE TO FAILURES

LEVEL	FLIGHT CONDITION		
	HOVER AND LOW SPEED	FORWARD FLIGHT	
		NEAR-EARTH	UP-AND-AWAY
1	3 deg roll, pitch, yaw 0.05g $n_x$ $n_y$ $n_z$ No recovery action for 3.0 sec	Both Hover & Low Speed & Forward Flight Up & Away Reqs Apply	Stay within the OFE. No recovery action for 10 sec
2	10 deg attitude change or 0.2g acceleration. No recovery action for 3.0 sec	Both Hover & Low Speed & Forward Flight Up & Away Reqs Apply	Stay within OFE. No recovery action for 5.0 sec
3	24 deg attitude change or 0.4g acceleration. No recovery action for 3.0 sec	Both Hover & Low Speed & Forward Flight Up & Away Reqs Apply	Stay within OFE. No recovery action for 3.0 sec

b. Rationale for Requirement

The treatment of failures must include the transient of disturbance and recovery, plus the subsequent steady state. The steady state handling qualities are covered by applying the appropriate handling qualities requirements through the mechanism of Levels. This paragraph is to address the transient.

The intent is to cover the following three concerns:

1. Loss of control.
2. Exceeding structural limits.
3. Collision with some nearby object.

The envelope definitions address concerns 1 and 2 by requiring that failure recovery shall not exceed the limits of the Service Flight Envelope. Item 3 is of concern when close to the ground or other obstacles, or when performing formation flight, etc., and limitations need to be placed on the translational excursions which result from failures, specifically a perturbation from hover or from a desired flight path.

Table 1 shows the excursions used in previous specifications (UTTAS and AAH PIDS, References 130 and 131). Presumably these were reasonable transients to recover from and to design for; however, they do not address the above concerns explicitly. For example, transient g levels would be quite severe for a 1.5g limit load factor helicopter but overly restrictive for a 3.5g helicopter. Similarly, if the helicopter were hovering close to an obstacle, the excursions induced by 0.5g translational acceleration could be excessive.

TABLE 1(3.1.7.6). MAXIMUM ALLOWABLE ROTORCRAFT RESPONSE FOLLOWING A FAILURE IN TRIMMED LEVEL FLIGHT (FROM UTTAS AND AAH PIDS)

	MAXIMUM ALLOWABLE RESPONSE WITH THE CONTROLS FREE FOR 3 SECONDS FOLLOWING SINGLE FAILURES IN TRIMMED LEVEL FLIGHT
Pitch	10 deg/sec
Longitudinal	0.5g longitudinal acceleration
Vertical	0.5g normal acceleration
Roll	10 deg/sec
Yaw	10 deg/sec; if below 40 knots the 10 deg/sec can be increased to 20 deg/sec
Lateral	0.5g lateral acceleration

Modern helicopters are quite likely to have multi-redundant, fly-by-wire, multi-mode stability and control augmentation systems, and consequently, quite different failure effects in different flight conditions. Hence a more comprehensive requirement is required. Table 3(3.1) was developed to satisfy this need.

Hover and Low Speed ( $\leq 45$  kt). It is assumed that most hover and low speed flight will be performed "near earth." In such situations it would be desirable to specify a box size which should not be penetrated within a certain time. Unfortunately, this is only meaningful in hover; at any forward speed, limits have to be placed on deviations from the desired flight path. The specification, therefore, places limits on translational accelerations and corresponding attitude changes. The Level 1 values are barely perceptible. This should be achievable by a



fail operate or fail passive system. Typical transient time histories for Level 2 and Level 3 are shown in Figure 1. Assuming that the failure causes an attitude rate, the Level 2 limits of 10 deg attitude and 0.2g acceleration at 3 seconds result in a transient displacement of 8.4 feet, a velocity of 8.4 feet per second, and an angular rate of 3.4 deg/sec. The Level 3 limits of 24 deg and 0.4g result in 20 feet, 20 ft/sec and 8 deg/sec. The Level 3 attitude rate is somewhat lower than the Table 1 value (8 deg/sec versus 10 deg/sec) but it is considered that the displacement and velocity are more critical and the chosen displacements are reasonable maximums.

Forward Flight (> 45 kt). Mission-Task-Elements flown at greater than 45 knots could be "up and away" or "near earth."

For "up and away" there should be no concern about collision with objects except, perhaps, close formation flight. The critical limits could be controllability or structural, and hence criterion parameters could be combinations of speed, load factor, angular attitudes and rates, etc., depending on the helicopter's characteristics. It was decided to limit the divergence by specifying a control-free time and demanding recovery without violating the Operational Flight Envelope. The requirements were developed in terms of Operational Flight Envelope boundaries to avoid safety of flight issues that would arise if the requirements were developed in terms of the Service Flight Envelope. The divergence times are 10 seconds for Level 1, a mild divergence, 5.0 seconds for Level 2, more noticeable, and 3.0 seconds for Level 3.

Figure 2 shows typical acceleration, attitude and attitude rate time histories for failures which induced constant pitch rate. Superimposed on these plots are the Table 1  $n_z$  and  $\dot{\theta}$  limits. The Table 1  $\dot{\theta}$  limits are greater than the new requirement allows. However, the Table 1  $n_z$  limit was much more restrictive. In particular, if the divergence was a linear increase of  $n_z$ , the  $n_z = 0.5g$  limit would be more restrictive than the new Level 1. If  $n_z$  had an exponential divergence the Table 1 limit would be worse than Level 1 but better than Level 2. These relative values would of course be different for a helicopter with a lower limit load factor. For example, on a 2g helicopter the Table 1  $n_z$  limit would fail Level 2 but pass Level 3 if the divergence was exponential, and would fail Level 1 but pass Level 2 if the divergence was linear.

For "near earth" operations at greater than 45 knots, concerns about displacement from desired flight path and the possibility of exceeding structural limits or losing control, all apply; so both sets of requirements have been applied.

The criteria in Table 3(3.1) are based on estimates and analysis since data is not available. However, an experiment was conducted as part of the Single Pilot Advanced Cab Engineering Simulations (SPACES II) (Reference 155) as a check on the Table 3(3.1) limits. That experiment is documented in Reference 184 and is summarized below.

The study in Reference 184 investigated the limits prescribed in the specification by creating single-axis hard-over failures which

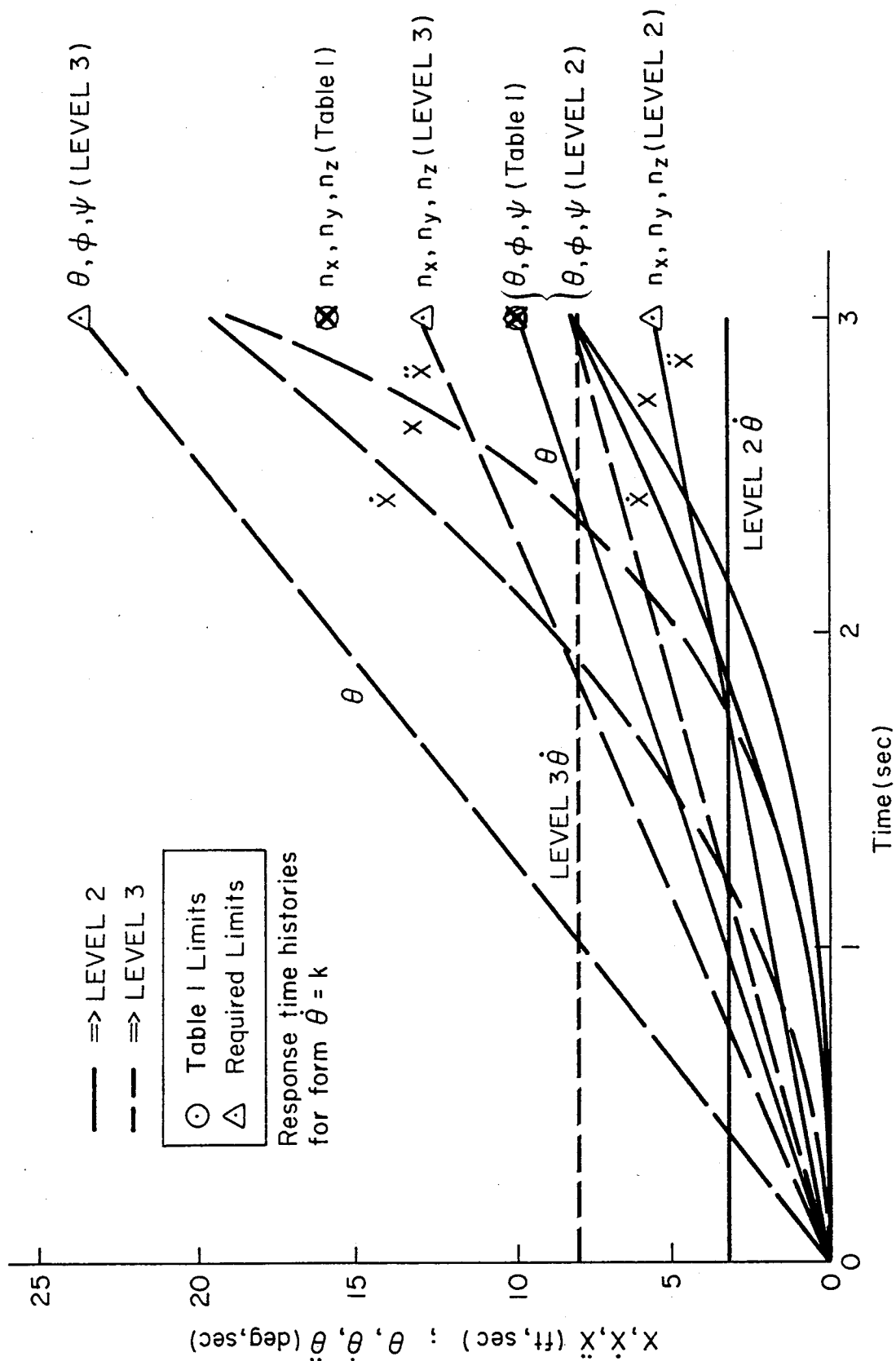


Figure 1(3.1.7.6). Hover and Low Speed Allowable Failure Transients for Levels 2 and 3

Assumptions: Failure causes  $\dot{\theta} = k$

$\Delta n = Z_w V \theta$ ,  $V = 145 \text{ kt}$ ,  $Z_w = -1.0$

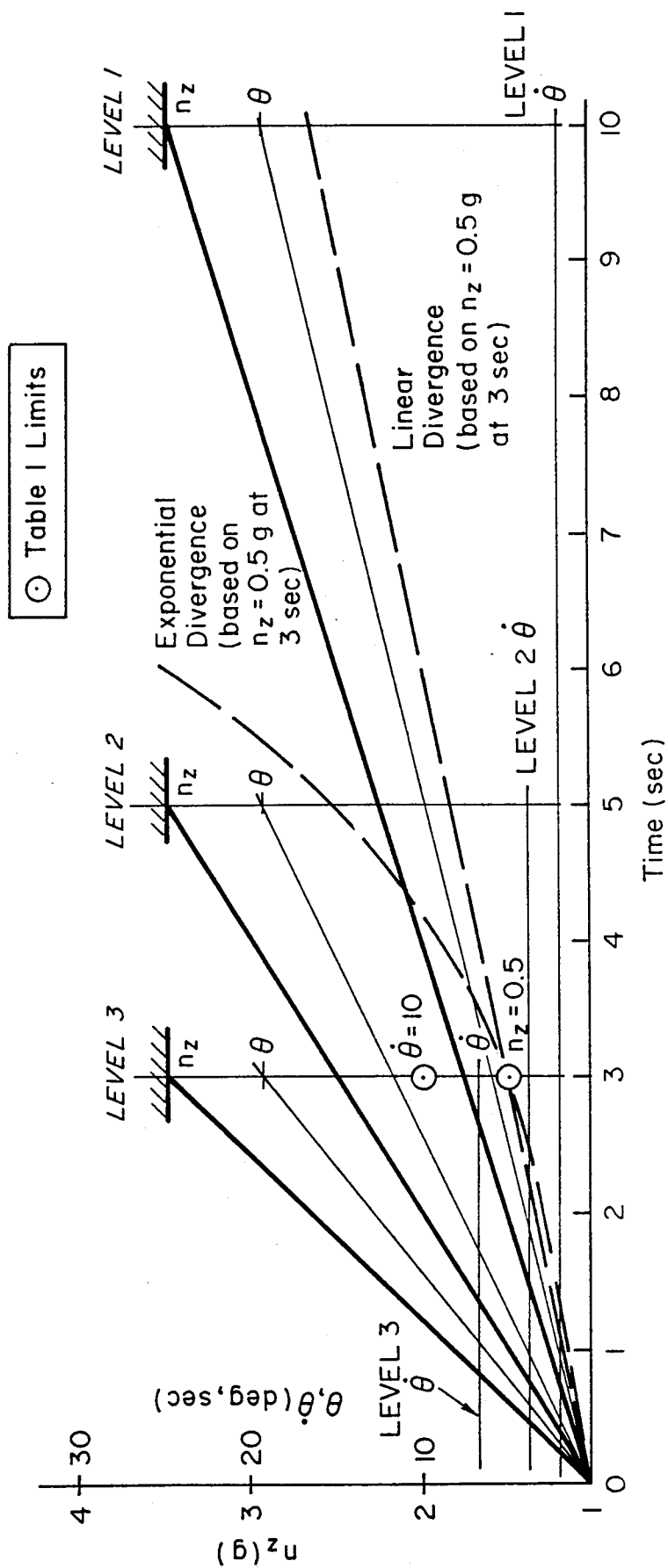


Figure 2(3.1.7.6). Forward Flight Allowable Failure Transients

resulted in attitude and acceleration excursions (controls free) conforming to the Levels 1, 2 and 3 limits in Table 3(3.1). The failures were induced randomly during two relatively low workload segments of the simulated mission and initial conditions such as altitude, airspeed etc., were essentially constant for all pilots at the time of failure. The failures were introduced through the AFCS by disconnecting the augmentation and inserting a step input in the axis of the simulated failure. The magnitude of the step was chosen to create the desired attitude excursion in 3 seconds, assuming no recovery action by the pilot. The post-failure aircraft was unaugmented in the axis of failure. Failures were induced in four axes (pitch, roll, yaw and heave) during hover, 60 kt NOE, and up and away flight at 100 kt forward flight. There was no cockpit indication to warn the pilots of a failure. The pilots were therefore reliant on the visual and motion cues of a transient to warn them of a failure.

The post-failure aircraft was rated mostly Level 3 by the pilots due to two factors; 1) the lack of available control authority in the axis of failure (trimming out the hard-over reduced available control power in that axis), and 2) the degradation in handling qualities due to the sudden lack of augmentation in one axis. However, controllability of the post-failure system was never in question.

The attempt to recover from an attitude excursion with no augmentation in the failed axis caused significant PIO's on several occasions.

Evaluation of the failures in the Reference 184 simulation proved to be somewhat inconclusive. This was due mainly to a wide scatter in the failure rating data. In broad terms, the hover data indicate that the pitch and roll requirements are reasonable, yaw is overly stringent, and heave is lenient. For forward flight, Reference 184 concludes that the Table 3(3.1) may be excessively stringent in all axes. There is insufficient data, however, to warrant the redefinition of the specification limits.

Reference 184 documents that the time taken by the pilot to recognize the failure and take corrective action was a key parameter in determining the success of the recovery effort. This is not surprising since this elapsed time determines the magnitude of the attitude or acceleration excursion at the time recovery action is initiated. Failure warning instrumentation should reduce this elapsed time and improve chances of recovery.

Reference 184 also states that the shape of the transient time response might also be a significant factor in early failure recognition and hence successful recovery. The specification may be expanded to include such a factor when more data becomes available.

a. Statement of Requirement

3.1.7.7 Indication of Failures. Immediate and easily interpreted indications of failures shall be provided, if such failures require a change of strategy or crew action.

b. Discussion

This paragraph requires that the pilot be provided with an immediate and easily interpreted indication of failures. It is expected that the manufacturer will work closely with the user activities to insure that the cockpit displays and/or annunciators meet the intent of this requirement.

a. Statement of Requirements

3.1.8 Rotorcraft Limits. Limiting and potentially dangerous conditions may exist where the rotorcraft should not be flown. When approaching such limits, it shall be possible by clearly discernible means for the pilot to recognize the situation and take preventive action.

3.1.8.1 Warning and Indication of Rotorcraft Limits. Warning and indication of approach to a rotorcraft limit shall be clear and unambiguous so that the pilot can avoid dangerous conditions. In Near-Earth Operations, the pilot shall be able to interpret warnings and avoid limiting and potentially dangerous conditions with eyes out of the cockpit.

3.1.8.2 Devices for Indication, Warning, Prevention, and Recovery. It is intended that limiting and dangerous flight conditions be eliminated and the requirements of this specification be met by appropriate aerodynamic design and mass distribution rather than through incorporation of special devices for indication, warning, prevention, and recovery. Neither normal nor inadvertent operation of such devices shall create a hazard to the rotorcraft or prohibit flight within the Operational Flight Envelope.

b. Discussion

The high level of pilot workload associated with operation in the battlefield environment leaves insufficient time for the pilot to continuously monitor all flight and propulsion system instruments. Experience has shown that many aircraft losses are due to operational problems, as opposed to effective enemy fire. Examples are settling with power when operating as the number two or three helicopter in an assault landing, and attempting to operate outside the performance envelope in high density altitude conditions. This paragraph requires new and innovative warning systems to assist the pilot in extreme workload conditions. Not only must the system warn the pilot of the approaching dangerous flight condition, but also provide advisories for corrective action. Voice annunciation may be necessary for these advisories. The development of these warning/advisory systems should include piloted simulations which include a realistic high workload tactical environment such as that recently completed by the U.S. Army Aeroflightdynamics Directorate at NASA Ames on the VMS Simulator (Reference 155, discussed in supporting data for Paragraph 3.2.2).

The specification calls for "prevention" (Paragraph 3.1.8.2) as well as warnings. Devices for the prevention of dangerous flight conditions are intended to be similar to the envelope limiting systems employed on fixed wing fly-by-wire aircraft such as the F-16. The F-16 can be flown to the limits of its angle-of-attack and normal acceleration envelopes by simply applying full aft control. The pilot knows that he can achieve maximum performance without monitoring his flight instruments, allowing him to allocate more time, mental effort, and psychological stress workload to attaining a tactical advantage over his

opponent. Similar envelope limiting devices are needed for the helicopter required to perform high workload Mission-Task-Elements. For example, the pilot should be able to pull full collective if attacked by another helicopter (tests show height advantage is critical), or should be able to pull maximum load factor without exceeding critical propulsion system limits. In the words of one manufacturer attending the interim progress reviews for this specification, "the system should protect the aircraft from the pilot."

a. Statement of Requirement

3.1.9 Interpretation of Subjective Requirements. In several instances throughout the specification, subjective terms have been employed where insufficient information exists to establish absolute quantitative criteria. Quantitative definition of these requirements shall be agreed upon by the procuring activity and the contractor prior to contract initiation.

b. Discussion

In most cases it has been possible to avoid the use of subjective phrases -- such as objectionable flight characteristics, realistic time delay, normal pilot technique, and excessive loss of altitude or buildup of speed. However, when there is no data whatsoever, and a requirement is clearly necessary, such phrases are unavoidable. They represent a reminder not to forget about a particular area, and rely on the judgment of the manufacturer and user to decide what constitutes an unsatisfactory item.



a. Statement of Requirement

3.1.10 Pilot-Induced Oscillations. There shall be no tendency for pilot-induced oscillations (PIO), that is, sustained or uncontrollable oscillations resulting from the efforts of the pilot to control the rotorcraft.

b. Discussion

This requirement is redundant because there should be no possibility of pilot induced oscillation (PIO) if the requirements of Paragraphs 3.3 and 3.4 are all complied with. It is included in recognition of the fact that it seems possible to "fool" even the most well thought out handling qualities criteria, and any PIO is clearly unacceptable.

a. Statement of Requirement

3.1.11 Residual Oscillations. Any sustained oscillations in any axis in calm air shall not interfere with the pilot's ability to perform the specified Mission-Task-Elements. For Level 1, oscillations in attitude and in acceleration at the pilot's station greater than 0.5 degrees and 0.05 g will be considered excessive for any Response-Type and Mission-Task-Element. These requirements shall apply with the cockpit controls fixed and free. Residual motions which are classified as a vibration shall be excluded from this requirement. The distinction between residual motions and vibration shall be based on mutual agreement between the manufacturer and procuring activity.

b. Discussion

This requirement is essentially the same as 3.2.2.1.3 of MIL-F-8785C and MIL-F-83300. The primary purpose of the requirement is to direct attention at limit cycles in the control system and structural oscillations which might affect pilot performance in the tactical mission, cause pilot discomfort, etc.

### 3.2 REQUIRED RESPONSE-TYPE

#### a. Statement of Requirement

3.2.1 Applicability of Requirements. The required Response-Types are specified as a function of the Mission-Task-Elements, and the Usable Cue Environments resulting from mission visual environments, including available vision aids and displays. These specified Response-Types are intended to be minimums, and an upgrade may be provided if superior or equivalent flying qualities can be demonstrated. If such an upgrade is selected, the detailed requirements in Paragraphs 3.3 and 3.4 pertaining to the upgraded Response-Type apply.

#### b. Discussion

Experience has shown that there are a very large number of response characteristics that can be created using current technology for sensing, actuation, and real-time computation. It is therefore not practical to attempt to define Response-Types and formulate the corresponding dynamic requirements for every conceivable augmented response. The approach taken has been to specify certain fundamental response characteristics in each axis. A "nonspecified Response-Type" category (Paragraph 3.2.11) has been defined to allow Response-Types which do not fit the descriptions of these fundamental response characteristics.

The requirements under Paragraph 3.2 are designed to define the minimum Response-Type that will allow accomplishment of the Mission-Task-Elements with Level 1 flying qualities. However, in some cases, the manufacturer may choose to incorporate increased stabilization to produce superior handling for some MTEs. In such cases, the dynamic response requirements that apply to the Response-Type actually being incorporated into the design shall be used. For example, if the manufacturer decides to upgrade from simple Rate to an Attitude Command/Attitude Hold (ACAH) Response-Type, the rotorcraft must meet the specific dynamic response requirements for ACAH. The rank ordering of Response-Types is given in Paragraph 3.2.2. It is important to note that adding stabilization may make it impossible to meet the requirements for moderate and large amplitude attitude changes in Sections 3.3 and 3.4.

a. Statement of Requirement

3.2.2 Required Response-Types for Specified Mission-Task-Elements And Usable Cue Environments. The Response-Types required for applicable Mission-Task-Elements and corresponding Usable Cue Environments (UCE) are specified for each axis of control for Hover and Low Speed (Table 1(3.2)) and Forward Flight (Table 2(3.2)). Unless Response-Types consistent with UCE=3 are already incorporated, the UCE is to be determined according to Paragraph 3.2.2.1. If the Mission-Task-Elements and UCE that comprise the required operational missions for the rotorcraft dictate more than one Response-Type, the Response-Type transfer requirements of Paragraph 3.8 must be complied with.

Nominally, the performance requirements for each required MTE will be specified by the procuring activity. If such performance requirements are not specified, the limits in Sections 4.1 and 4.2 will apply. Furthermore, it is recognized that maneuvering will, in general, be less aggressive in a Degraded Visual Environment. If relaxed performance standards are not specified by the procuring activity, the criteria in Sections 4.4 and 4.5 will apply.

b.. Rationale for Requirement

The intent of this requirement is to establish a methodology which allows the specification to relate the required handling qualities to the mission requirements, and to the operational visual environments. It implies an intimate knowledge of the mission tasks, the mission visual environments, and vision aids. It is intended that the MTEs be obtained from compliance with Paragraph 3.1.1, and the UCE value(s) from compliance with Paragraph 3.2.2.1. The MTEs in Tables 1(3.2) and 2(3.2) should encompass most missions, although some "common sense" interpretations will invariably be required. For example, a jump takeoff requires essentially the same handling qualities as a normal vertical takeoff and a bob-up.

The Response-Type abbreviations are defined below Table 1(3.2), where the user is referred to the appropriate paragraph. In some cases an Attitude Command or Rate Response-Type is specified in addition to Position Hold which gives the appearance of an over-constrained flight control system (i.e., how is it possible to have Position Hold and Rate or ACAH at the same time?). Experience has shown that it is necessary to retain the integrity of the attitude-to-stick response at all times (see discussion in Paragraph 3.2.11). Therefore, it is reasoned that if the pilot commands an attitude, he no longer wants to hold position, and his command should override the Position Hold.

It is important to take the notes below Table 1(3.2) into account as they are an integral part of the requirement. They include a relaxation in the requirement for RCHH, a blanket requirement for a turn coordination mode in the Low Speed flight regime, and a rank-ordering of Response-Types to define what constitutes an allowable upgrade.

TABLE 1(3.2). REQUIRED RESPONSE-TYPE FOR HOVER AND LOW SPEED - NEAR EARTH

	UCE = 1		UCE = 2		UCE = 3	
	Level 1	Level 2	Level 1	Level 2	Level 1	Level 2
Vertical takeoff and transition to forward flight - clear of earth.	Rate	Rate	Rate	Rate	Rate	Rate
Precision Hover. Slung load pickup and delivery. Slung load carrying. Shipboard landing including RAST recovery. Vertical takeoff and transition to near-earth flight. Hover-taxi/NOE-traveling. Rapid Slalom (note a).			ACAH+RCDH +RCHH ↓	Rate+RCDH ↓	TRC+RCDH +RCHH+PH ↓	ACAH+RCDH +RCHH ↓
Slope landing. Precision vertical landing. Pull-up/Push-over (note a).			ACAH+RCDH ↓			ACAH+RCDH ↓
Rapid Bob-up & Bob-down (note a). Rapid Hovering turn.	↓		ACAH+RCDH +RCHH+PH			Rate+RCDH +RCHH+PH
Tasks involving divided attention operation (see Para. 2.5.2). Sonar dunking (note b). Mine sweeping (note b).	Rate+RCDH +RCHH+PH ↓					
Rapid transition to precision hover (note a). Rapid sidestep (note a). Rapid accel and decel (note a). Target acquisition and tracking (note a and c).	Rate ↓	↓	↓	↓	↓	↓

Notes:

- a. High levels of aggressiveness may not be achievable for UCE = 2 and 3.
- b. These tasks are normally accomplished in an environment where the visual cuing may be consistent with UCE = 2 or 3 even in "day VFR conditions".
- c. Increase in rank to TRC not recommended for pitch pointing tasks.

1. A requirement for RCHH may be deleted if the Vertical Translational Rate Visual Cue Rating is 2 or better, and divided attention operation is not required. If RCHH is not specified, an Altitude-Rate Response-Type is required (see Paragraph 3.2.9).

2. Turn coordination (TC) is always required as an available Response-Type in the Low Speed flight range as defined in Paragraph 2.6.2. However, TC is not required at airspeeds less than 15 knots.

3. For UCE = 1, a specified Response-Type may be replaced with a higher rank of stabilization, providing that the Moderate and Large-Amplitude Attitude Change requirements are satisfied.

4. For UCE = 2 or 3, a specified Response-Type may be replaced with a higher rank of stabilization.

5. The rank-ordering of combinations of Response-Types from least to most stabilization is defined as:

1. Rate
2. ACAH+RCDH
3. ACAH+RCDH+RCHH
4. Rate+RCDH+RCHH+PH
5. ACAH+RCDH+RCHH+PH
6. TRC+RCDH+RCHH+PH

Rate => Rate or Rate Command Attitude Hold (RCAH) Response-Type (Paragraphs 3.2.5 and 3.2.6).

TC => Turn Coordination (Paragraph 3.2.10.1)

ACAH => Attitude Command Attitude Hold Response-Type (Paragraphs 3.2.6 and 3.2.7)

RCHH => Vertical-Rate Command with Altitude (Height) Hold Response-Type (Paragraph 3.2.9.1)

RCDH => Rate-Command with Heading (Direction) Hold Response-Type (Paragraphs 3.2.5 and 3.2.6)

PH => Position Hold Response-Type (Paragraph 3.3.11)

TRC => Translational-Rate-Command Response-Type (Paragraph 3.2.8)

TABLE 2(3.2). REQUIRED RESPONSE-TYPES IN FORWARD FLIGHT

Pitch and Roll Attitude	
RATE	PITCH - RATE OR ATTITUDE, ATTITUDE HOLD REQUIRED (RCAH OR ACAH) ROLL - RATE WITH ATTITUDE HOLD (RCAH).
Ground attack Slalom Pull-up/Push-over Assault landing Sonobuoy deploy VMC cruise/climb/ descent Air combat	IMC cruise/climb/descent IMC departure IMC autorotation IMC approach (constant speed) IMC decelerating approach (3-cue flight director required) Air-to-air refuel Mid-air retrieval Weapons delivery requiring a stable platform
Heading -- All require Turn Coordination (see Paragraph 3.4.6.2)	
Height -- No specific Response-Type (see Paragraph 3.4.3)	

The requirement for turn coordination in Table 1(3.2) is based on the NASA/Army variable stability CH-47 in-flight simulation reported in Reference 185. It was found that the RCDH characteristics were inappropriate for low-speed forward maneuvering flight, since they suppressed any inherent weathercock stability. This was considered the most objectionable flight control system characteristic tested. If TC and RCDH are both required, the switching between these Response-Types can be objectionable if activated based on a reference speed. For example, the Reference 185 tests found such switching objectionable (HQR=4.5) due to lack of pilot awareness of the heading SCAS mode, and switching transients in and out of TC/RCDH. Manual switching via a pedal switch is suggested as a viable alternative. Control mode switching logic is covered only briefly in Paragraph 3.8 due to a lack of data in this area. Therefore, the specification must rely on the Section 4 flight test maneuvers to uncover such deficiencies.

In some cases, a user may find that a certain flight control system mechanization does not fit any of the Response-Types. In such cases, the definitions in Section 3.2 should be checked carefully to insure that the basic intent of a defined Response-Type is not satisfied, even if the mechanization is not consistent with the semantics. For example, an acceleration command, velocity hold flight control system is not a defined Response-Type but, if mechanized properly, can be shown to fall into the ACAH Response-Type category. This is because the short term response will almost certainly satisfy the requirements for Attitude Command ( $\ddot{x} \triangleq g\theta$  until the  $X_u$  term begins to cause a slow bleedoff in attitude). The "Attitude Hold" criterion requires that attitude return to within 10% of peak following a pulse stick input, which it will for

most helicopters (because the trim attitude for the small velocity change induced by a pulse will be small). If the attitude does not return to within 10% of trim, the response starts to look like Rate Command/Attitude Hold, which falls into the Rate Response-Type category. This is as it should be, since such a system would not provide the workload relief of ACAH which always tends to return to the trim attitude.

The UCE concept was introduced to account for the use of vision aids (e.g., night vision goggles, infrared sensing, superimposed symbology, etc.) in the NOE environment. A detailed methodology for determination of the UCE is given in Paragraph 3.2.2.1. In essence, an increase in UCE implies degraded visual cueing, and a need for increased stabilization. Vision aids are designed specifically to allow the pilot to navigate and avoid collisions with objects, and in fact, this is assured by Paragraph 3.2.4. There is, however, a second and less recognized requirement to provide cues for stabilization. The loss of microtexture, common with most vision aids, seriously degrades the ability of the pilot to stabilize the helicopter, a deficiency which is accounted for in this specification by the requirements for increased rotorcraft stability for UCE=2 and 3 in Table 1(3.2).

It is important to make the distinction between the need for increased stabilization, and the need to see to navigate. For example, experience has shown that it is difficult to hover over a spot on the ocean, even though the visibility is unlimited. The lack of fine-grained texture, rotor wash, and a moving surface all combine to remove the translational cues necessary to conduct precision hover with low workload and high pilot confidence. The methodology devised to determine the UCE is specifically oriented towards cues for stabilization as opposed to navigation. For example, the UCE for sonar dunking may be 2 or 3 (i.e., poor), even on a CAVU day due to the lack of translational rate cues. The routine use of doppler radar coupled to the autopilot for such tasks is evidence of this.

The requirement is worded so that it is not necessary to determine the UCE if the Response-Type for UCE=3 is already incorporated in the design. For example, if a manufacturer proposes to include TRC + PH + RCDH + RCHH, operation in UCE=3 should be Level 1, and the only issue is guidance (Paragraph 3.2.4).

An additional element in the definition of MTEs is the procuring activity's prerogative to designate each MTE as a "fully attended" or "divided attention" task. Recent handling qualities experiments (see Reference 155) have indicated that a higher level of Response-Type is necessary if the pilot is required to divide his or her attention from the primary flying task for significant amounts of time. The results of the Reference 155 piloted simulation experiment are discussed in detail under subsection 4 of the supporting data. The person responsible for designating a task as divided or full attention, should be familiar with the results of those tests.

It may be necessary to incorporate more than one Response-Type in the design of a modern helicopter flight control system to achieve Level 1 flying qualities for all the MTEs required to complete the mission. This will usually be manifested by the inability to meet the moderate or large amplitude attitude requirements in Section 3.3, for Response-Types which embody a high level of stabilization. For example, it would be difficult to meet the  $q_{pk}/\Delta\theta$  and  $p_{pk}/\Delta\phi$  requirements of Paragraph 3.3.3 with a TRC or ACAH Response-Type. The specification avoids requiring a specific Response-Type for each MTE (only a minimum is required). It is hoped that the flight control system designer will be encouraged to use clever techniques to avoid loading the pilot with complex switching between modes to achieve the necessary stabilization for some MTE/UCE combinations, and still meet the moderate and large amplitude dynamic requirements. For example, it would be possible to meet all of the moderate and large amplitude requirements with an ACAH or TRC system if mode switching to a Rate Response-Type were included as a function of cockpit control amplitude. Such "clever" mechanizations require careful scrutiny. For example, the transition from ACAH or TRC to a Rate Response-Type at large stick amplitudes could result in excessive nose down pitch attitudes for a rapid acceleration from hover (for example, see Reference 12). It is not possible to account for all such idiosyncracies in a flying qualities requirement, which is the primary motivation for including the Section 4 flight test maneuvers. And, of course, it is the responsibility of the flight control system designer to thoroughly investigate new or novel mechanizations for "flying qualities cliffs," i.e., situations where the flying qualities can degrade suddenly during some kind of maneuver.

#### c. Supporting Data

A considerable amount of supporting data are required to justify the requirements for Response-Types in Tables 1(3.2) and 2(3.2). This information is therefore presented in several sections as summarized below.

1. Near-Earth; UCE=1; Flight Test Data. This section supports the UCE=1 column in Table 1(3.2). The data supports a requirement for a simple Rate Response-Type for most MTEs (when the aircraft control task can be fully attended).
2. Near-Earth; UCE=2 and 3. Both in-flight and ground-based simulation data are available as background for the UCE=2 and 3 columns in Table 1(3.2). These data indicate that, in general, ACAH is required for Level 1 when UCE=2, and TRC+PH when UCE=3. In addition, Heading Hold and Height Hold are required for most tasks when  $UCE \geq 2$ .
  - a. Flight Test Data. This section supports the UCE=2 and 3 columns in Table 1(3.2) based on flight test data taken on the National Research Council (NRC) of Canada



variable stability Bell 205A. Night vision goggles with daylight training filters were used to vary the visual environment. The data applies to the case where the pilot can devote full attention to the aircraft control task.

- b. Ground-Based Simulation Data. This section also supports the UCE=2 and 3 columns in Table 1(3.2) based on recent simulation data (Reference 198) taken from the NASA Ames Vertical Motion Simulator (VMS).
3. Up-and-Away; IMC; Ground-Based Simulation. This section provides supporting data for the tasks to be performed in instrument meteorological conditions (IMC) in Table 2(3.2).
4. Near-Earth; UCE=1; Single Pilot; Ground-Based Simulation. This section provides support for the Table 1(3.2) requirement for additional stabilization in good visual conditions (UCE=1) if significant division of attention from the rotorcraft control task is required to accomplish the missions. The data consists of ratings for a single pilot operating in simulated battlefield conditions, and is the result of two ground-based simulations on the NASA Ames Vertical Motion Simulator (VMS) facility.

Because of the fundamental importance of Tables 1(3.2) and 2(3.2), it was essential that supporting data contain and comply with at least the following:

- Pilot assessments of vehicle dynamics for tasks similar to one or more of the Mission-Task-Elements of Table 1(3.2) or 2(3.2).
- Pilot ratings as a minimum; preferably pilot comments as well.
- Identification of vehicle dynamics to a degree sufficient to ascertain Response-Type and control-response characteristics.

The supporting data include the results of both jet-lift V/STOL and rotary-wing hover and low speed experiments, since little data exist for either class of aircraft alone. Fortunately, the response characteristics of the vehicles tested were quite similar (i.e., RCAF and ACAH are implemented using the same control laws on either class of aircraft), and the differences (cross-axis coupling on helicopters, low heave damping on V/STOLs) were almost always alleviated or removed entirely through augmentation.

References providing valuable background pertinent to this paragraph, from both flight and ground simulation experiments, are summarized in Table 1.

#### 1. Supporting Data; Near-Earth; UCE=1

The supporting data and rationale for the UCE=1 columns of Table 1 are presented in this section.

##### UCE=1; Level 1

A Rate Response-Type is required for all stationary MTEs when UCE=1. This simply means that there is no specific requirement on the shape of the response as long as it meets the dynamic requirements of Paragraph 3.3. The supporting data for this is quite extensive, and is summarized below.

Reference 5 reports the results of an experiment conducted to study simulation fidelity for the NASA Ames Vertical Motion Simulator (VMS). The work also provides valuable flight data for this specification.

An Army UH-60A was flown through a series of tasks at the Army Aviation Engineering Flight Activity (AEFA), Edwards Air Force Base by five Army pilots. The pilots also participated in a simulation of the UH-60A to fly similar tasks. The simulation ratings are discussed separately. Six NOE-related maneuvers (all Mission-Task-Elements in Table 1(3.2)) were performed at Edwards AFB: 90- and 180-deg hover turns, accel/decel, bob-up, sidestep, dolphin (pull-up/push-over), and slalom. Pilot ratings for the six tasks and the five pilots are shown in Figure 1. Level 1 pilot ratings were attained for all the tasks, and as the dashed line in Figure 1 shows, the average ratings were 3-1/2 or better.

Time histories and frequency responses presented in Reference 186 show that the UH-60A, as tested, can be classified as a Rate Response-Type, without Attitude Hold. As an example, Figures 2 and 3 show frequency responses of  $q/\delta_{LONG}$  and  $p/\delta_{LAT}$ , respectively, for the UH-60A at hover. Frequency sweeps from the AEFA flight tests were reduced to frequency-response form by fast-Fourier transform (FFT) techniques as shown in Figures 2 and 3. The bandwidths of the pitch and roll attitude responses are approximately 3 and 6.5 rad/sec, respectively -- Level 1 by the requirements of Paragraph 3.3.2.1.

The pilot ratings in Figure 1 provide direct support for pitch/roll classifications in Table 1(3.2) for UCE=1. Specifically, the Mission-Task-Elements defined as Bob-Up/Down, Lateral Sidestep, Accel/Decel, Slalom, Dolphin, and Hover Turn are all placed under the requirement for Rate Response-Type in pitch and roll. The tasks performed in Reference 5 involve some or all of the elements of these Mission-Task-Elements. The only unresolved issue for these data is the effect of turbulence, because the Reference 5 tests were conducted early in the morning in order to avoid the normally high winds and gusty conditions that develop

TABLE 1 (3.2.2). SUMMARY OF EXPERIMENTS PROVIDING HOVER AND LOW SPEED DATA

REF. NO. (YEAR)	TEST AIRCRAFT OR SIMULATOR	AIRCRAFT SIMULATED	DESCRIPTION OF TASK(S)	TURBULENCE/WINDS	SAS TYPES
5 (1984)	UH-60A and VMS	Articulated Rotor Helicopter	Assessment of simulation fidelity. Low speed tasks: hover turn, dash/quickstop, bob-up, side step, dolphin. 5 pilots.	Simulation: light turbulence. Flight: Ambient conditions, normally calm.	Rate Response-Type in pitch, roll, and yaw.
45 (1984)	XV-15 (Flight Tests)	(XV-15 Tiltrotor)	Hover ICE (2-6 ft AGL) and OCE (100 ft AGL); sideward and fore/aft hover translations (IGE); turns; STOL takeoffs and landings (70 kt).	"Ideal" conditions.	Rate responses in pitch, roll, and yaw. Evaluations made SAS-on, SAS-off, and with single axis off.
13 (1979)	FSAA	Type A V/STOL	Shipboard landings; varied Sea State (SS), wind over deck (WOD), wind angle ( $\psi_{WOD}$ ). 2 pilots. Sophisticated HUD may have had an influence on per- formance and pilot ratings.	Turbulence varied with Sea State: SS-0 to 6, $\sigma_{ug} =$ 1.4 to 4.2 ft/sec. $V_{WOD} = 15$ to 43 kt, $\psi_{WOD} = 0$ and -30 deg.	Translational Rate/Position Hold (VCPH) with heave augmentation; ACAH in pitch/roll (Baseline) with heave; RCAH in pitch/roll/ yaw, no heave augmentation (Backup FCS).
14 (1984)	Boeing Vertol (Phase 1) VMS (Phase 2)	Generic Helicopter (UH-60A)	Day VMC and night IMC (with helmet-mounted display). Bob-up, accel-decel, NOE, slalom, approach to hover, hover. Variations in side stick controller configura- tions and SAS types. 4 Pilots. Phase 1 used terrain board; Phase 2, CGI.	Included gusts and shears in hover and approach to hover tasks; otherwise, none.	Pitch and roll: Acceleration Command/Rate Hold, RCAH, ACAH, Attitude Command/Groundspeed Hold. Heave: RCHH. Yaw: RCDH.
48 (1983)	VMS	V/STOL (AV-8A Model)	Approach to landing, station- keeping, landing on DD963-Class Destroyer. Several HUD and Response-Type variations. 1 pilot.	Sea States (SS) varied from SS-0 to 6; winds/ turbulence varied correspondingly.	AV-8A (rate damping); rate, ACAH, TRC in pitch/roll; with and without vertical velocity damping.

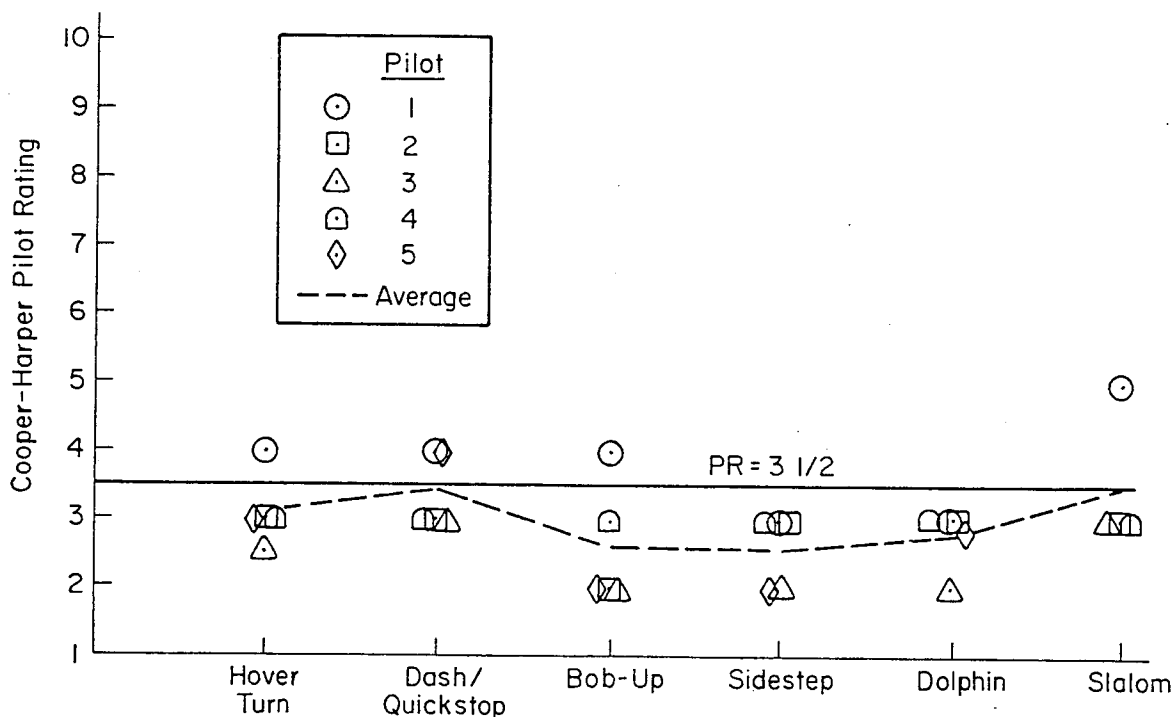
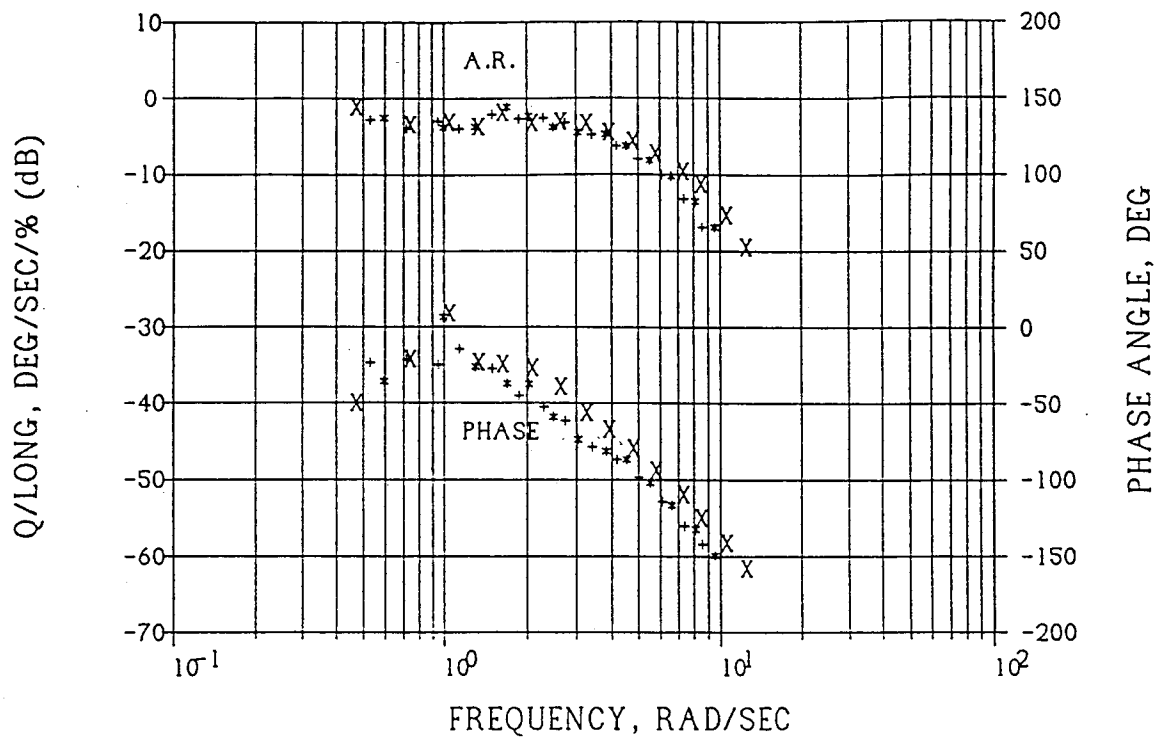


Figure 1(3.2.2). Pilot Rating Results from Flight Test of UH-60A (Reference 5)

at Edwards AFB during the day. No quantitative data is available to resolve this issue at this time; however, for most of the tasks of Figure 1, a pilot rating degradation of one-half to one still would not result in Level 2 average ratings. More recent flight tests conducted at the NRC in Canada (discussed below, and in detail in the "Supporting Data" for 3.3.2.1) strongly suggest that pure Rate Response-Types are satisfactory for Level 1 in higher turbulence levels; i.e., the need for Attitude Hold or an Attitude Response-Type appears to be more a function of visual cues than of atmospheric disturbances.

Flight tests using the NASA Ames Research Center XV-15 Tilt-Rotor Research Aircraft (Reference 45) provide further support for Rate Response-Types. The pitch/roll stability and control augmentation system (SCAS) used rate feedback for disturbance rejection, and forward loop shaping for bandwidth improvement. The frequency responses in pitch and roll, with the SCAS on and off, are shown in Figure 4. As this figure shows, the SCAS-off bandwidth is about 0.4 rad/sec in both pitch and roll and the response characteristics are primarily acceleration. The SCAS improves the response to a Rate Response-Type with a bandwidth of 2.4 rad/sec in roll and 2.9 in pitch.

# UH-60 FREQUENCY RESPONSE, 0 KTS



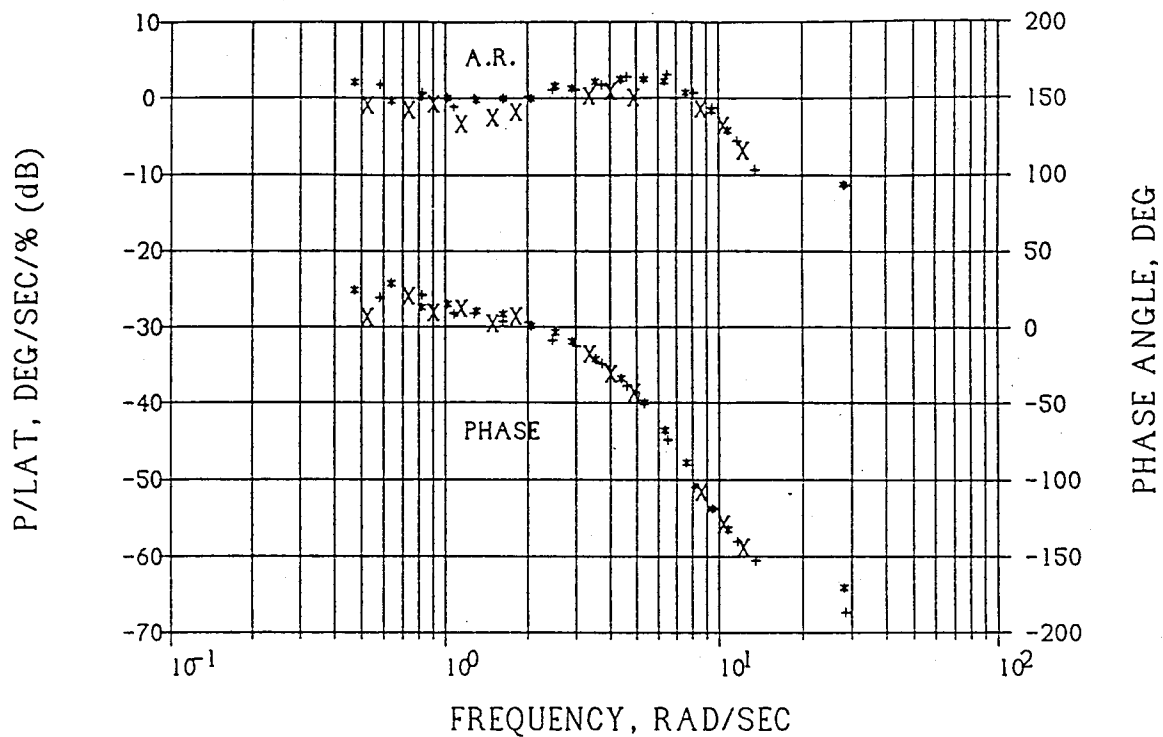
## Average Helicopter/Atmospheric Conditions

Gross Weight	=	16,760.0 lb
Longitudinal C.G. Position	=	354.7 inch
Outside Air Temperature	=	19.8 deg C
Pressure Altitude	=	2,455.0 ft
Density Altitude	=	3,578.0 ft

Symbol	Run
+	748004.09X
*	748004.10X
X	748004.11X

Figure 2(3.2.2). Pitch Rate Response to Longitudinal Cyclic Stick Displacement Describing Function at Hover  
(from Reference 186)

# UH-60 FREQUENCY RESPONSE, 0 KTS



## Average Helicopter/Atmospheric Conditions

Gross Weight	-	16,934.0 lb
Longitudinal C.G. Position	-	354.7 inch
Outside Air Temperature	-	20.3 deg C
Pressure Altitude	-	2,445.0 ft
Density Altitude	-	3,627.0 ft

Symbol	Run
+	748004.04X
*	748004.05X
X	748004.07X

Figure 3(3.2.2). Roll Rate Response to Lateral Cyclic Stick Displacement Describing Function at Hover  
(from Reference 186)

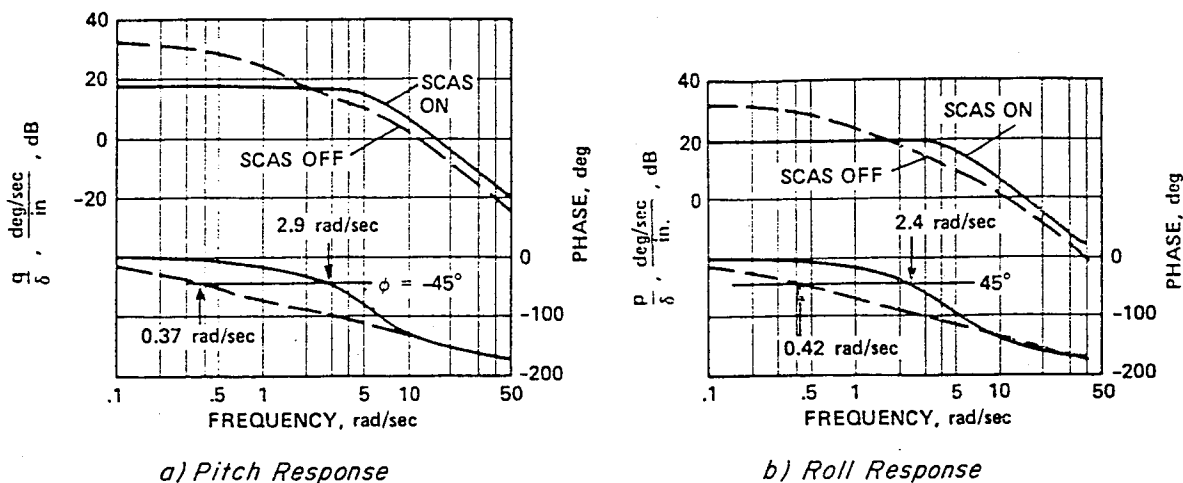


Figure 4(3.2.2). XV-15 Frequency Responses at Hover  
(from Reference 45)

For the Reference 45 investigation, a single pilot flew several low-speed tasks (precision hover in and out of ground effect, and hover translations), turns in tilt-rotor (T/R) and airplane (A/P) modes, and STOL takeoffs and landings at 70 kt. Figure 5 shows the pilot ratings. The XV-15 was also evaluated with various channels of the SCAS turned off, as indicated in Figure 5. Level 1 pilot ratings were obtained for all tasks with SCAS on, which supports the Table 1(3.2) requirement for a Rate Response-Type for the MTEs in Figure 5. Selectively turning off the pitch, roll and yaw SAS channels in Figure 5 generally produced Level 2 ratings, indicating that an acceleration Response-Type\* in one axis is acceptable for Level 2. However, when more than one SAS channel was turned off ("all off" in Figure 5), the ratings degrade to Level 3 for hover translations.

The requirement for a Rate Response-Type for Level 2 in Table 1(3.2) (UCE=1) means that the Level 2 boundaries in Section 3.3 apply. However, this only applies to the response to control inputs. Paragraph 3.2.12 removes the requirement for a Rate Response-Type for disturbance regulation (inputs to control actuator) to allow input shaping (i.e., no feedback, and hence, no sensors) for the backup flight control systems. Hence, there is no requirement on the response to atmospheric disturbances for Levels 2 and 3, UCE=1.

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\*The specification does not define an "acceleration" Response-Type. However, a configuration which does not meet the minimum Bandwidth requirements of Paragraph 3.3.2.1 will exhibit an acceleration response to a step stick input.

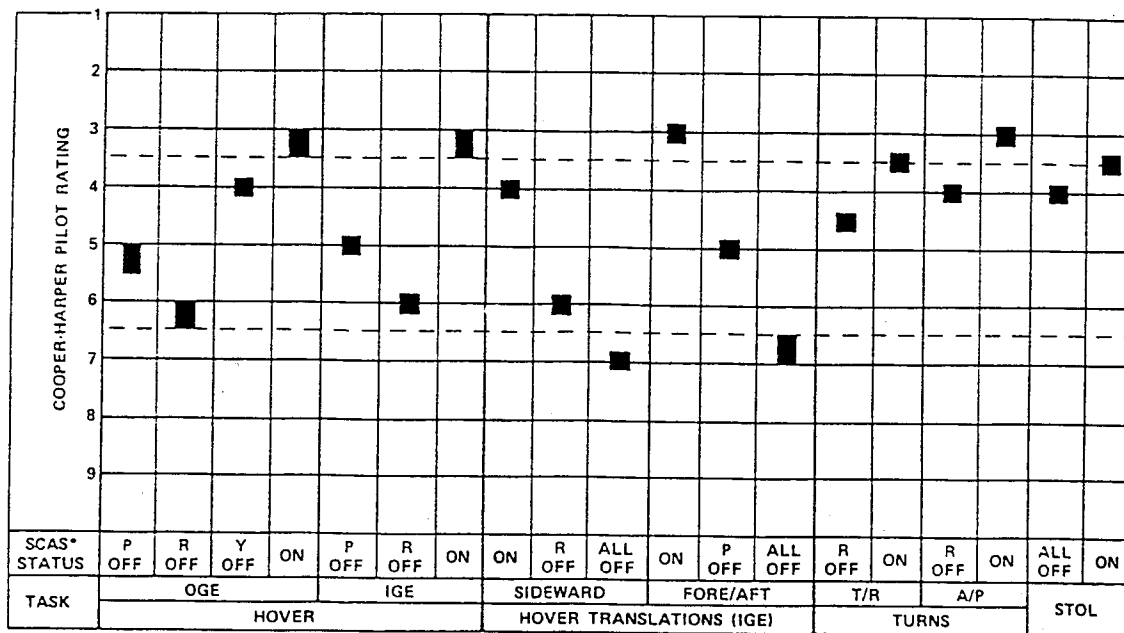


Figure 5(3.2.2). Pilot Rating Summary from XV-15 Flight Tests (from Reference 45)

Some additional supporting data for the Rate Response-Type classifications for UCE=1 in Table 1(3.2) can be obtained from operational trials of the AV-8A Harrier (Reference 46). Reference 47 contains some control system information for the AV-8A, and this has been used to correlate the data from Reference 46. The unaugmented aircraft is unstable, and pitch and roll rate feedbacks are utilized in the SCAS to produce a Rate Response-Type with Level 1 short-term characteristics.

Low-speed tasks covered in Reference 46 include vertical takeoffs, transitions to and from hover, hover, and vertical landings. Most low-speed deficiencies are related to poor directional stability. For daytime operations, the airplane was rated Level 1 for all tasks except the vertical takeoff, where the airplane is unstable due to plume impingement effects. Transitions to and from hover in the AV-8A were rated Level 1 (HQR 3) in daylight. SAS-on hover was rated 2, and SAS-off a 4. These data lend operational support to the Table 1(3.2) requirement for Rate Response-Types for Takeoff and Approach to Hover (including shipboard operation).

It is interesting to note that all AV-8A operations reported in Reference 46 became more difficult at night -- i.e., in conditions of reduced visual cues. Most tasks that were rated 2 or 3 in daytime received ratings of 5-7 at night. This lends support for the need for increased visual cueing or additional stabilization, or both, as outside cues are removed.



A flight program was conducted in direct support of this specification at the Canadian National Research Council (NRC) using the Bell 205A Airborne Simulator (Reference 157). Several Rate, RCAH and ACAH Response-Types were simulated on the NRC helicopter and flown over a closed course as sketched in Figure 6. Five pilots flew the complete program. The pilot ratings for the best Rate Response-Types averaged 2-3 and the pilot comments indicated that simple rate damping was as good as, if not superior to, RCAH or ACAH for hover and low speed Mission-Task-Elements in UCE=1. Some of these results were obtained in moderately gusty air with winds of 15 gusting to 25 knots. These results are discussed in detail in Reference 157.

Several Mission-Task-Elements (MTEs) requiring Rate Response-Types in Table 1(3.2) have not been discussed here. These Mission-Task-Elements have been assumed to be sufficiently similar to others that were covered that they have been categorized as shown without direct supporting data. More directly applicable supporting data is obviously desirable, but, unfortunately, not available.

UCE=1; Level 1; Rate+RCDH+RCHH+PH

The need to add heading hold, height hold, and position hold for conditions of divided pilot attention is based on the data discussed in Subsection 4, "Near-Earth; UCE=1; Single Pilot."

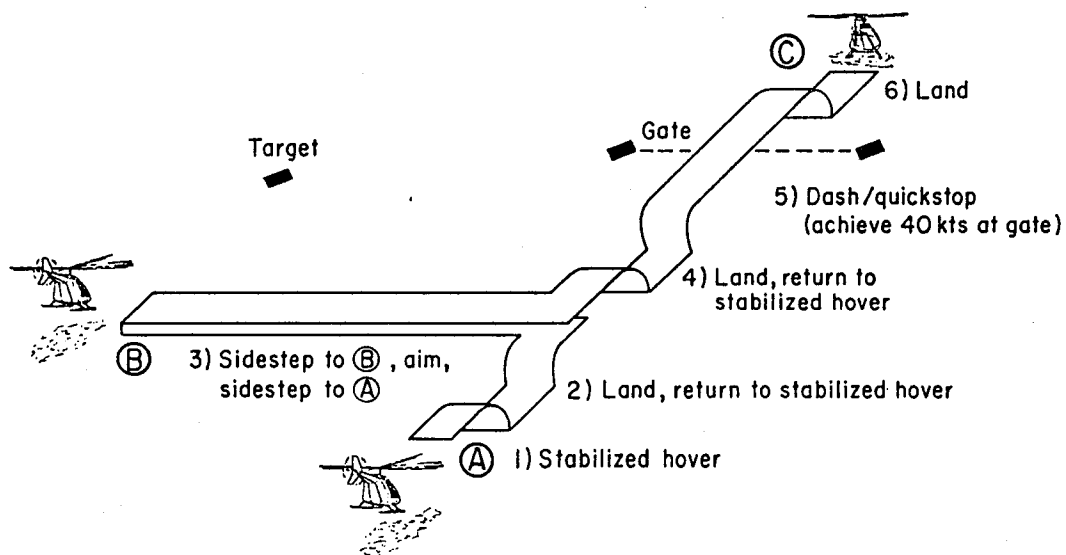


Figure 6(3.2.2). Task Scenario for Flight Tests at NRC (Canada).  
Scenario was Repeated Three Times and Separate Ratings  
Given for Hover, Landing, Quickstop, Sidestep

## UCE=1; Level 2; Rate

Since Rate is the lowest level of Response-Type, it is specified as the failure mode. This is academic since failures in UCE=1 are handled as a degradation in the dynamic response throughout the specification. The specification of a Rate Response-Type as Level 2 for divided attention tasks reflects experience with current day helicopters. That is, divided attention tasks are regularly accomplished, but with an acknowledged high level of workload.

## 2. Near-Earth; UCE=2 and 3

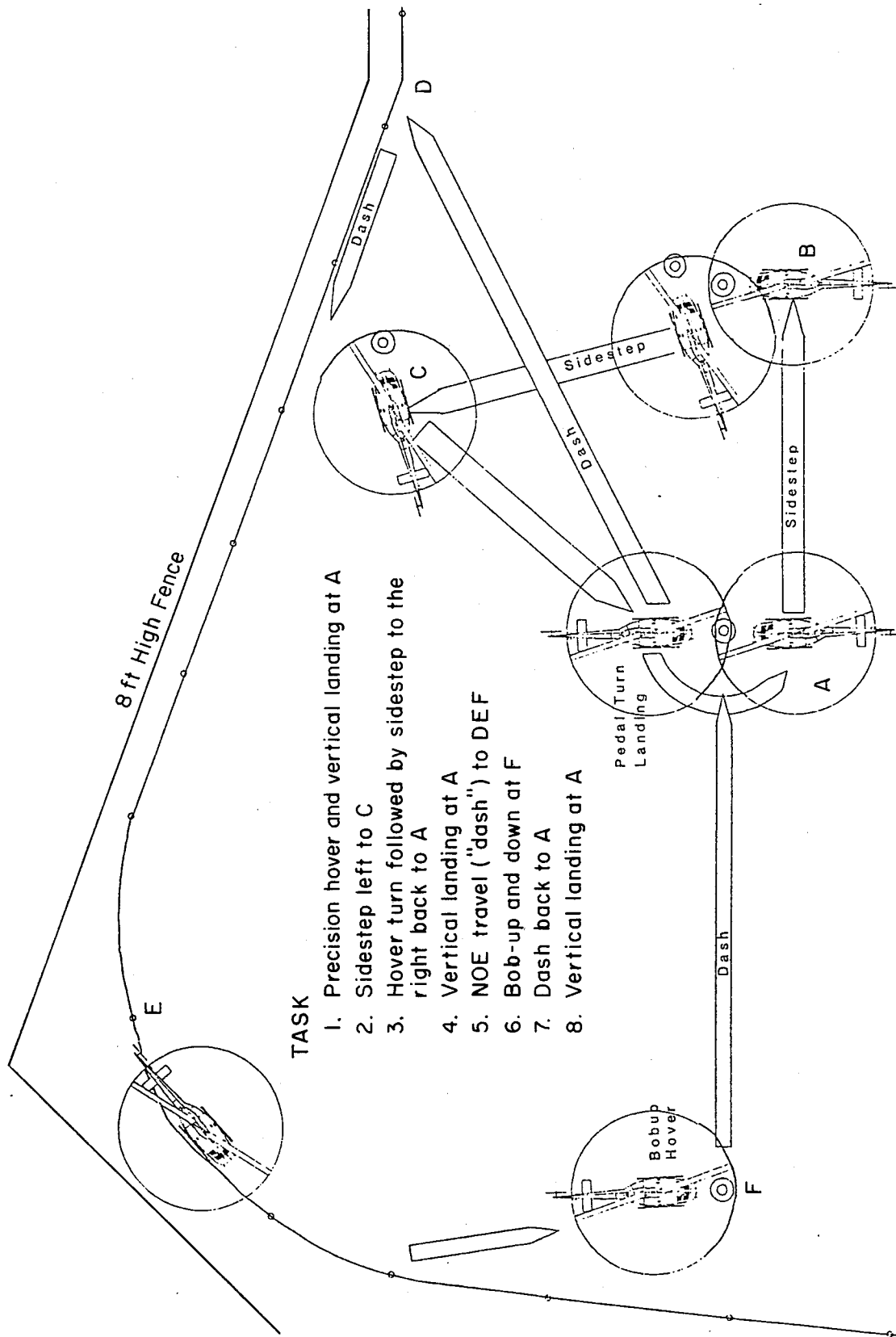
The UCE=2 and UCE=3 columns in Table 1(3.2) are intended to quantify the fact that additional stabilization is required to accomplish hover and low speed tasks when certain visual cues are not available. This phenomenon has been quantified in several flight test and ground-based simulation programs (see Reference 156). The primary facility for the in-flight visual cueing experiments has been the NRC variable stability Bell 205, using night vision goggles with daylight training filters and electronically fogged lenses to vary the visual environment. The ground-based simulation experiments were conducted on the NASA Ames Vertical Motion Simulator. All of these experiments have indicated that there is a quantifiable tradeoff between visual cueing and rotorcraft stabilization.

The methodology used to quantify the tradeoff between stabilization (i.e., "Response-Type") and visual cueing ("UCE") is contained in Table 1(3.2) and Paragraph 3.2.2.1 ("Determination of the Usable Cue Environment"). The supporting data for the boundaries that determine the UCE is discussed subsequently in the supporting data for Paragraph 3.2.2.1. Readers not familiar with the derivation of the UCE boundaries in Figure 2(3.2) of the specification may want to read the supporting data section for Paragraph 3.2.2.1, or Reference 156, before continuing this section.

### 2a. Near Earth; UCE=2 and 3; In-Flight Simulation

These experiments were conducted by the National Research Council Canada, using a variable stability Bell 205A helicopter. The piloting tasks which were given separate Cooper-Harper handling qualities ratings (HQRs) were: precision hover, vertical takeoff and landing, lateral sidestep, an NOE dash, and a bob-up/down (see Figure 7). These tasks were performed three times, with the exception of vertical takeoffs and landings which were performed nine times, in each tested combination of visual environment and Response-Type. The methodology for making determinations of visual cue ratings is described in Reference 156 and Paragraph 3.2.2.1. Seven Response-Types were tested; the primary variations were in the pitch, roll, and heave axes and are summarized below. The yaw axis Response-Type was always Rate Command/ Direction Hold (RCDH).

- Rate -- consisted of simple rate damping in pitch and roll, with no augmentation in heave.



# TASK

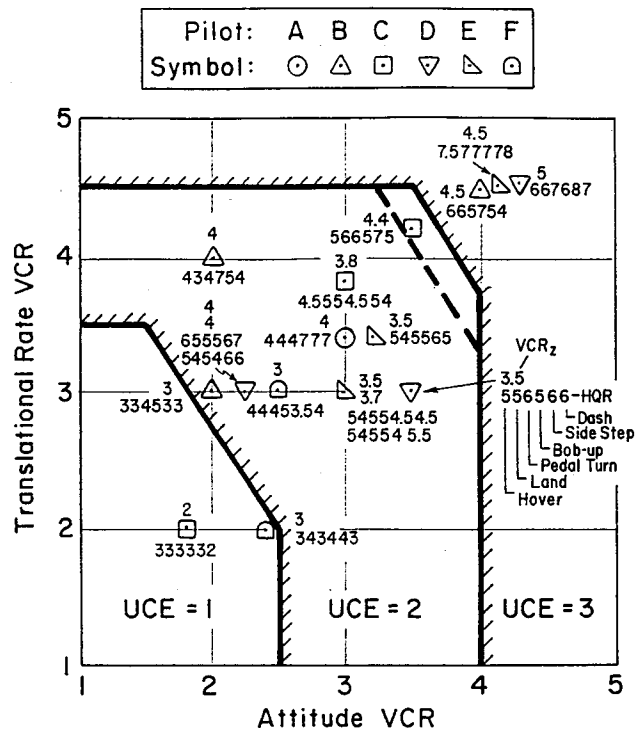
1. Precision hover and vertical landing at A
2. Sidestep left to C
3. Hover turn followed by sidestep to the right back to A
4. Vertical landing at A
5. NOE travel ("dash") to DEF
6. Bob-up and down at F
7. Dash back to A
8. Vertical landing at A

Figure 7(3.2.2). Description of Task in UCE Experiment

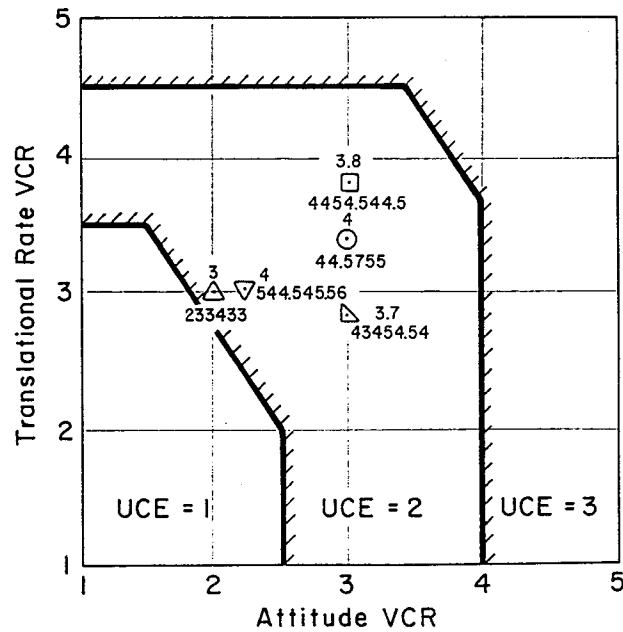
- Attitude Command/Attitude Hold (ACAH) -- ACAH in pitch and roll, and no augmentation in heave.
- Rate plus Position Hold (Rate + PH) -- Rate with the addition of a pilot selectable position hold, blended in over a 5-sec period.
- Rate plus Position Hold plus Rate Command/Height Hold (Rate + PH + RCHH) -- Rate + PH with the addition of pilot selectable Altitude Rate Command with Height-Hold function (RCHH). The RCHH function held altitude with respect to the ground (i.e., radar altitude) as long as the collective controller was in a mechanical detent. When out of the detent, the collective controlled a specified rate-of-climb per inch of displacement. The heave damping was augmented in the RCHH mode.
- ACAH + RCHH -- as noted above.
- ACAH + PH + RCHH -- as noted above.
- Translational Rate Command plus Position Hold (TRC + PH) -- in this mode longitudinal and lateral cyclic stick position commanded a translational rate. The commands were lagged to avoid abruptness in attitude, found to be an undesirable feature of TRC in the simulation of Reference 4. This also had the desirable effect of maintaining the consonance between stick and attitude, a feature demanded by the pilots in Reference 4. Position Hold was automatically armed when the stick was within a prescribed deadband, and activated when the groundspeed was below 5 knots. RCHH was always on in this mode.

The pilot ratings (HQRs) for each task are plotted on the UCE boundaries in Figure 8. Each datapoint includes six ratings, one for each of the tasks in the following order: 1) Precision Hover, 2) Vertical Landings, 3) Pedal Turns, 4) Bob-Up and Down, 5) Sidestep, 6) NOE Dash. Note that the first three ratings involve stationary low hover tasks and the last two involve maneuvering tasks. The use of the term "stationary" is meant to refer to horizontal translations. The bob-up and down is unique in that the translation is vertical. It is grouped with the stationary MTEs rather than create a separate category. The data in Figure 8 are based on the results obtained from the six pilots that participated in the experiment. All were rated helicopter pilots with substantial flight test experience.

A review of Figures 8a through 8f indicates the tradeoff between UCE and Response-Type noted in Reference 156, that is, it is possible to make up for degraded UCE via increased stabilization. Note that the UCE=2 boundary from Reference 156 (dashed line in Figure 8a) has been shifted slightly to the right to allow inclusion of a data point for TRC



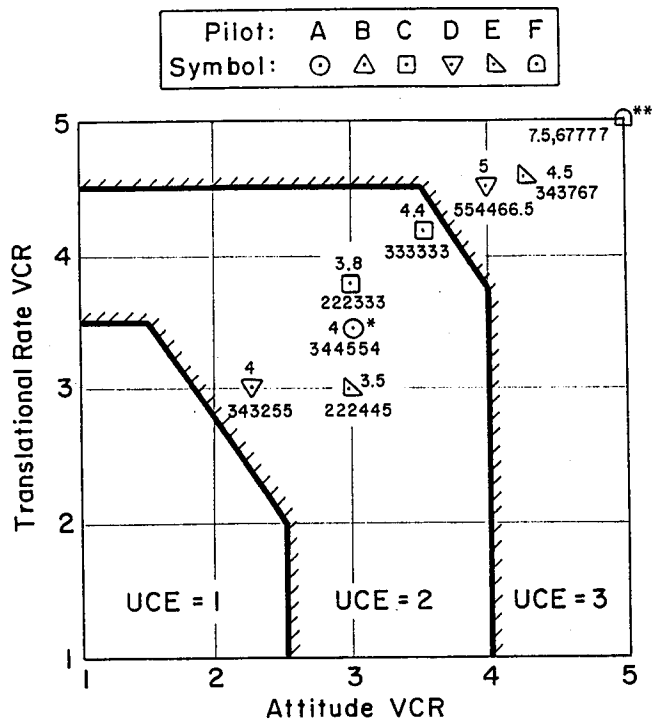
a) Rate + RCDH



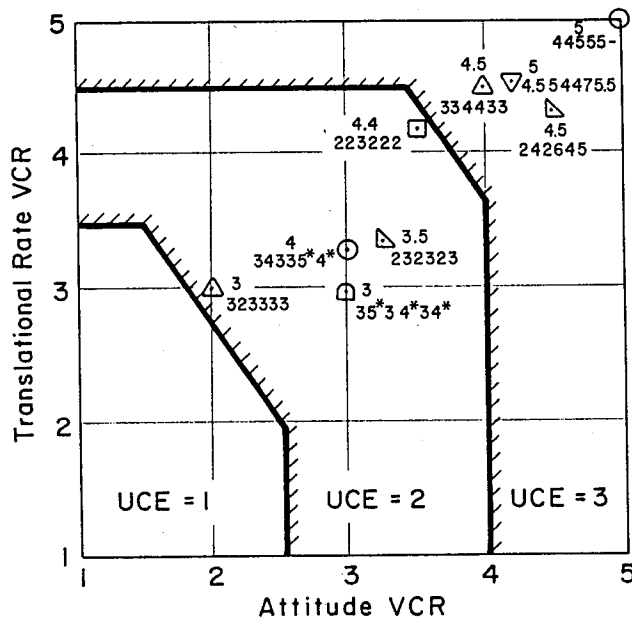
b) ACAH + RCDH

Figure 8(3.2.2). Data From UCE Experiment  
(Inflight Simulation)





e) *ACAH + PH + RCHH + RCDH*



f) *TRC + RCDH + PH + RCHH*

\*RCHH was not operating due to excessively small detent.  
-- data not used in Figure 9

\*\*Sun angle bad "could see nothing in front of aircraft"  
-- did not use this data in Figure 9 as navigation was a primary limitation

\*Problems with RCHH mechanization  
-- data not used in Figure 9

Figure 8(3.2.2). (Concluded)

+ PH + RCDH + RCHH (Figure 8f). Figure 8a shows that this revised boundary also correctly puts that data point (square symbol) in UCE=2 for a Rate Response-Type. This revised boundary is reflected in Figure 2(3.2) of the specification.

The Figure 8 data are the basis for the Response-Type requirements for UCE=2 and 3 in Table 1(3.2). The following discussion contains the data interpretations and rationale which support the Response-Type requirements in Table 1(3.2) for UCE=2 and 3.

The data have been crossplotted on Figures 9a, 9b, and 9c to isolate categories of Mission-Task-Element (MTE) in terms of stationary and translating MTEs. These plots indicate that the improvement in handling qualities ratings (HQRs) afforded by the tested combinations of Response-Types generally occurs in the following order from least improvement to most improvement for UCE=2 or 3.

1. Rate + RCDH -- Baseline Response-Type
2. ACAH + RCDH
3. ACAH + RCDH + RCHH
4. Rate + RCDH + RCHH + PH
5. ACAH + RCDH + RCHH + PH
6. TRC + RCDH + PH + RCHH

The above numerical designations have been retained as a label for these combinations of Response-Types in Figure 9, and in the following discussions.

Rate command of heading with direction hold (RCDH) was included in all the tested configurations, and was not a variable in the experiments. We have taken a conservative approach by requiring RCDH for all cases where UCE=2. This seems intuitively correct as the addition of one more axis to control in poor visibility is easily seen to represent a large workload increment. This was verified in the NASA Ames CH-47 in-flight simulation reported in Reference 185.

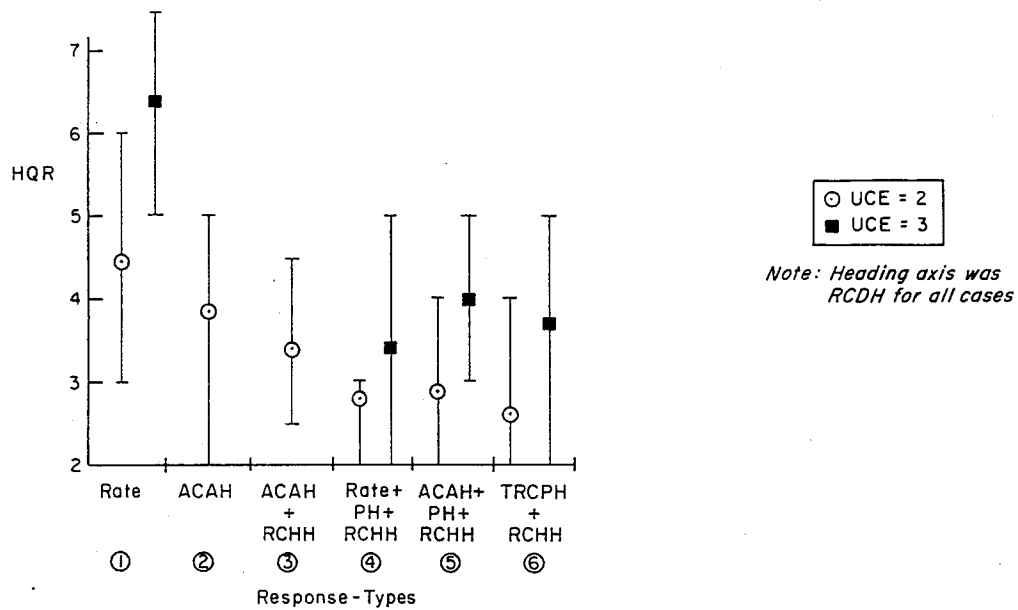
The above ordering of Response-Types (from 1 to 6) is the basis for the allowance to increase the stabilization from that specified in Table 1(3.2), see note 3 below the table.

## 2b. Near-Earth; UCE = 2 and 3; Ground-Based Simulation

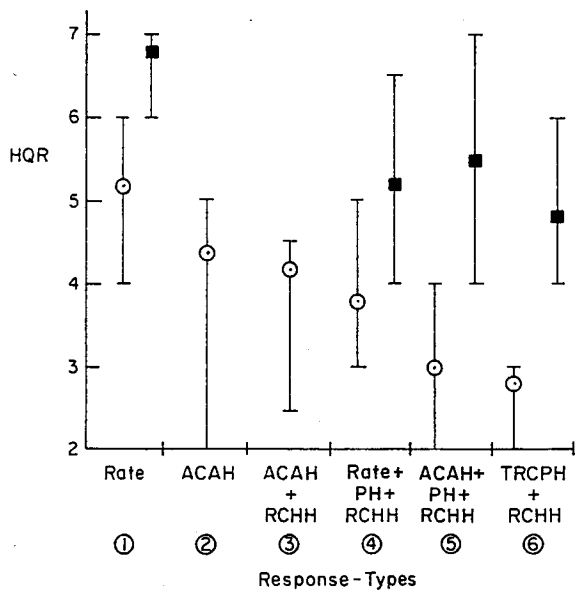
A ground-based piloted simulation was conducted (Reference 198) at NASA-Ames on the Vertical Motion Simulator to assess Response-Types in degraded visual conditions (UCE=2 and 3). Response-Types included a Rate Command (RC), an Attitude Command/Attitude Hold (ACAH), and a Translational Rate Command (TRC). Pilot control was implemented through conventional controls, i.e., a cyclic, collective, and pedals.

The piloting tasks which were given separate Cooper-Harper Handling Qualities Ratings (HQRs) were: a precision hover, a vertical landing, a pirouette, a dash/quickstop, and a sidestep maneuver (see Figure 10). These tasks were flown in sequence and desired and adequate performance

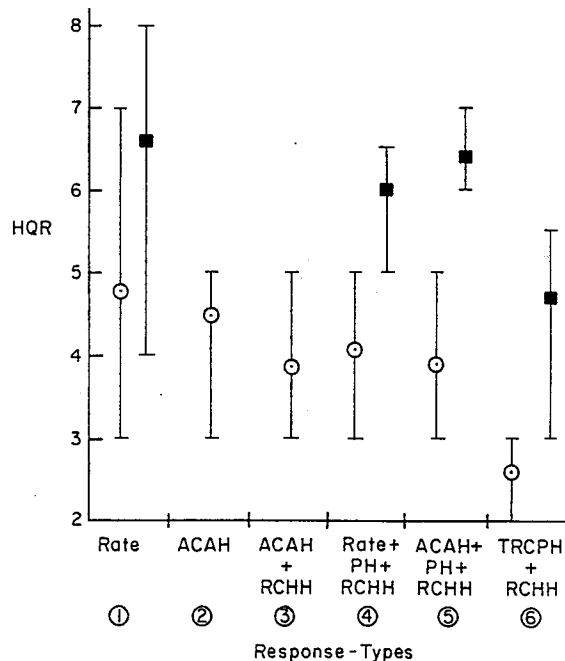




a) Stationary Low Hover Tasks (hover, land, 360° hover turn)



b) Bob-Up and Down Task



c) Maneuvering Tasks (sidestep and dash)

Figure 9(3.2.2). Pilot Rating Data from In-Flight Simulation as a Function of Response-Type and UCE

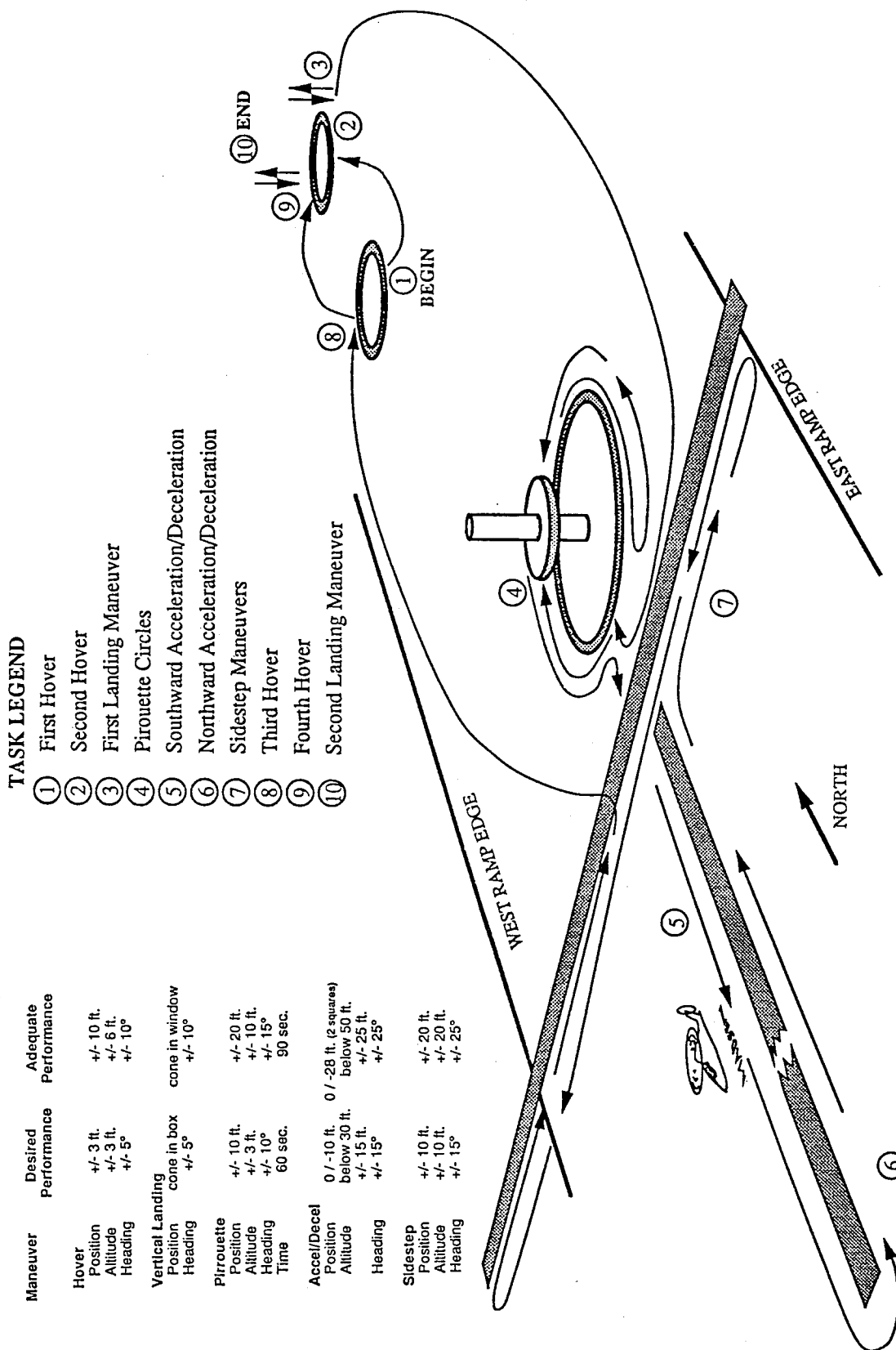


Figure 10(3.2.2). Description of Tasks in Ground-Based UCE Simulation

was specified for each task. One sequence through this course required four precision hovers, two pirouettes (one to the left and one to the right), and two dash/quickstops. The course was flown at least twice before ratings and comments were collected. Visual Cue Ratings (VCRs) with a Rate Command Response-Type were initially collected to determine the UCE. Subsequent evaluations included the ACAH and the TRC. A simple stability derivative mathematical model with known roots and no coupling was used to perform the evaluations. Vehicle pitch and roll response dynamics are listed in Figure 11. The yaw axis Response-Type was always a Rate Command/Direction Hold (RCDH) and the vertical axis Response-Type was always an Altitude Rate Command.

The pilots' HQRs for each task are plotted on the UCE boundaries in Figure 12. The data in Figure 12 are based on the results from the seven pilots that participated in the experiment. The Figure 12 data have been crossplotted on Figure 13 to isolate categories of MTEs in terms of stationary and translating MTEs. The stationary MTEs in Figure 13a include the hover and landing tasks; the translating MTEs in Figure 13c include the dash/quickstop and the sidestep tasks; the pirouette task was plotted separately in Figure 13b. The following subsection discusses these results for the Response-Type requirements in UCE=2 and 3.

#### UCE=2; Level 1

Figure 13 supports the previously presented in-flight results, i.e., an Attitude Command Response-Type is required to achieve Level 1 pilot ratings in UCE=2. Note that this Response-Type (ACAH + RCDH) received Level 1 ratings for the stationary and the maneuvering tasks as well as the pirouette task.

#### UCE=2; Level 2

In UCE=2, a Rate Response-Type is required for Level 2 flying qualities for all MTEs. The HQRs are very consistent, ranging from 4 to 5.5.

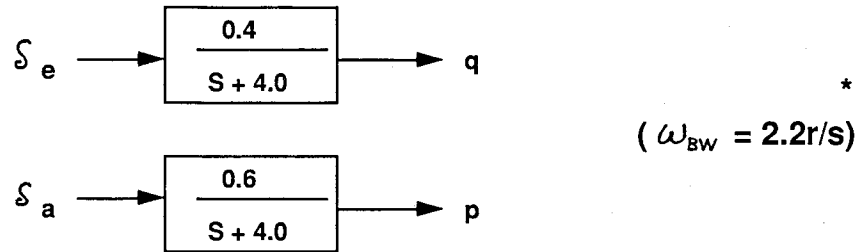
#### UCE=3; Level 1

For the stationary MTEs, TRC is required to obtain consistent Level 1 HQRs. Although some Level 1 ratings were received with the ACAH Response-Type, the majority of the ACAH ratings fell in the Level 2 region, while all the TRC ratings were better than HQR=3. For the Pirouette Task and the Maneuvering MTEs in Figures 13b and 13c, Level 1 HQRs can be achieved with TRC.

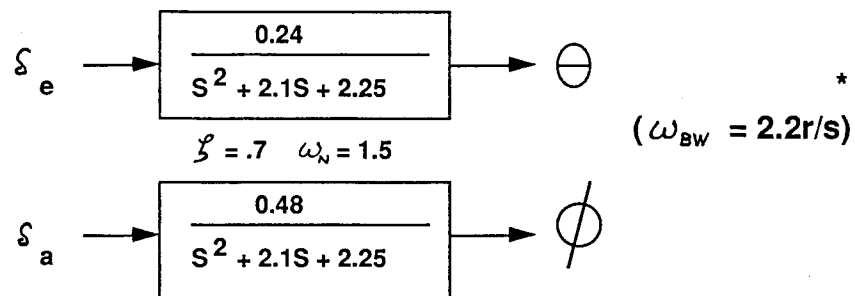
#### UCE=3; Level 2

As with the in-flight data discussed previously, a Rate Response-Type would almost be adequate as a Level 2 requirement in UCE=3.

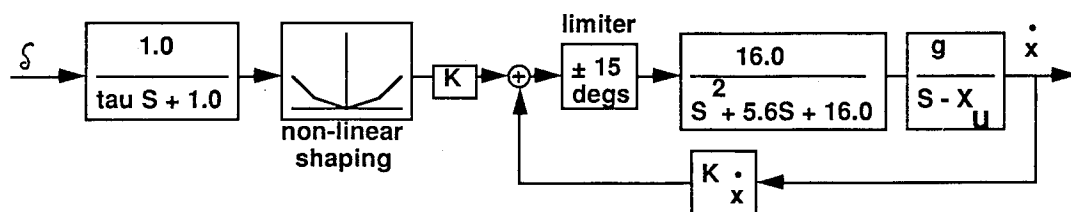
### Rate Command:



### Attitude Command:



### Translational Rate Command:



Pitch:  $\tau = 1.5$ ,  $K = 4.5$ ,  $K \dot{x} = .25$

Roll:  $\tau = 1.25$ ,  $K = 4.5$ ,  $K \dot{x} = -.20$

\* Computed Bandwidth includes 132 msec of pure delay

Figure 11(3.2.2). Configurations for Ground-Based UCE Simulation

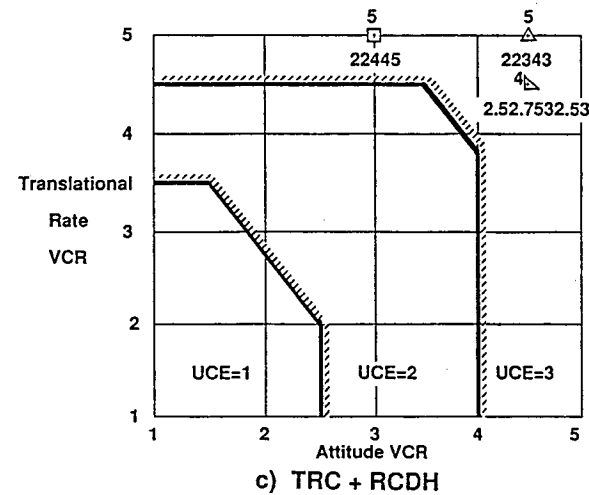
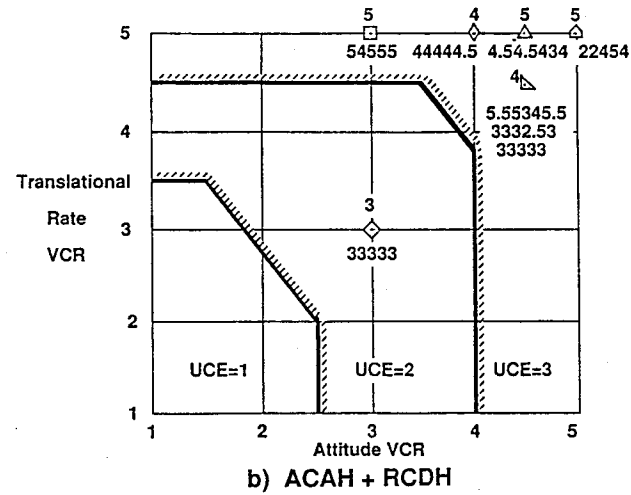
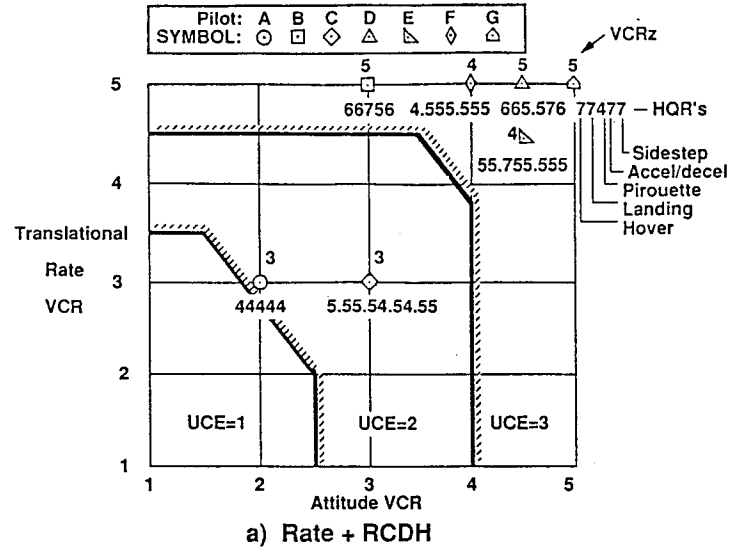
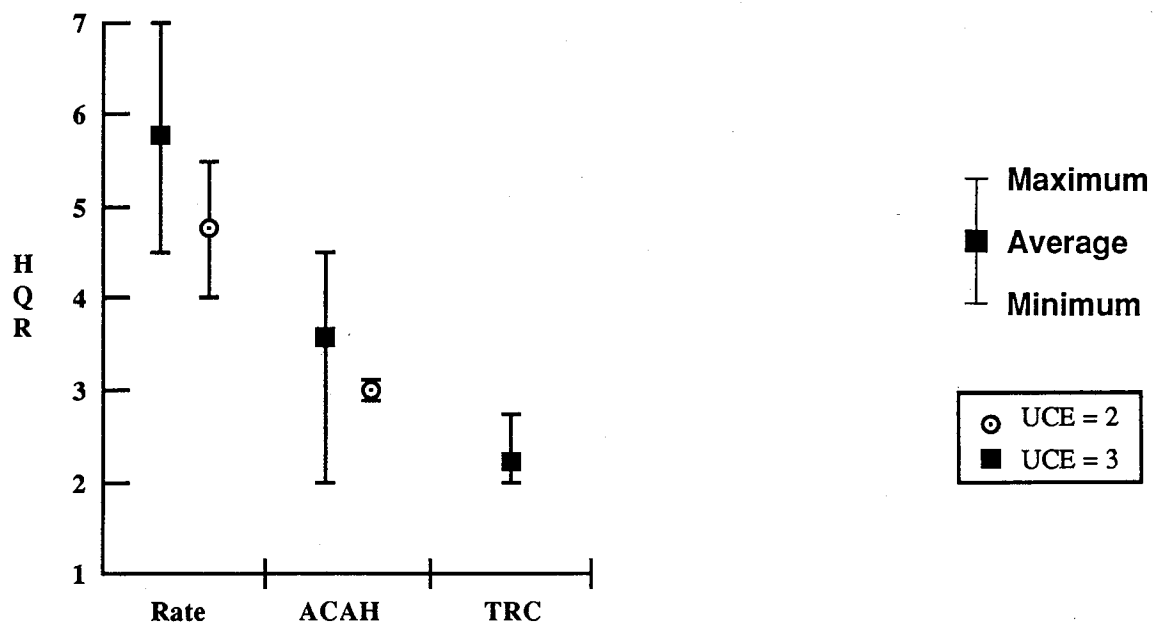
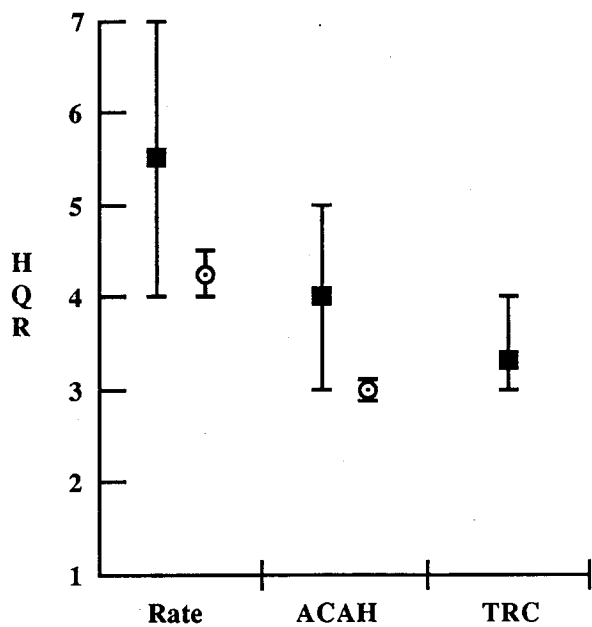


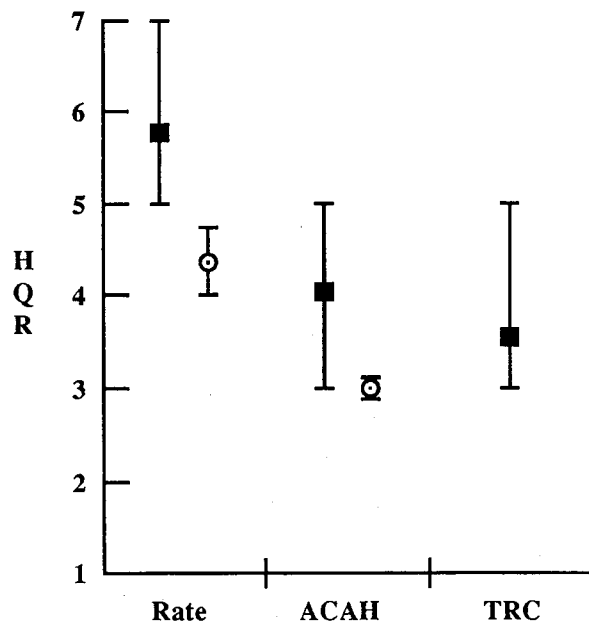
Figure 12(3.2.2). Data from Ground-Based UCE Simulation



a) Stationary MTE's (hover and landing tasks)



b) Pirouette Task



c) Maneuvering MTE's (sidestep, accel/decel)

Figure 13(3.2.2). Pilot Rating Data from Ground-Based Simulation as a Function of Response-Type and UCE

However, the HQRs are marginally Level 2 (and in some cases Level 3) and it is suspected that with any increase in pilot workload, such as with divided attention tasks, these marginally Level 2 results would become Level 3. Therefore, the added stabilization from the ACAH is necessary. The ACAH Response-Type provides consistent Level 2 results, and in some cases Level 1 ratings.

#### 2c. Summary of Justification for UCE=2 and 3 Columns of Table 1(3.2)

The justification for each of the Response-Type requirements for UCE=2 or 3 in Table 1(3.2) is summarized below.

##### UCE=2; Level 1; ACAH+RCDH+RCHH

Figure 9a indicates that for stationary MTEs and UCE=2, ACAH + RCDH + RCHH (Response-Type 3) is required to achieve an average Level 1 pilot rating, and this is reflected in Table 1(3.2). If the hovering MTEs are judged to involve divided attention operation (for example, single pilot), the Response-Type will require the addition of PH, as indicated by row 5 in Table 1 (also see subsections 1b and 4). Hover-taxi and Rapid Slalom are included in this group, since the data indicate that ACAH is required for all maneuvers in UCE=2, but that Position Hold is obviously inappropriate for these tasks. Note that the experiments did not include a slalom task, but that the dash task in Figure 9 is reasonably representative\*. The slope landing, vertical landing, and pull-up/push-over MTEs are grouped in row 3, and delete the requirement for Height Hold for obvious reasons.

##### UCE=2; Level 1; ACAH+RCDH+RCHH+PH

The remainder of the tasks (rows 4, 5, and 6) fall in this category as discussed below.

The pilot rating data in Figure 8 indicated that a distinction should be made between the bob-up/down MTE and the other stationary MTEs. This is revealed from a comparison of Figures 9a and 9b, where it is seen that the bob-up/down MTE requires ACAH + RCDH + RCHH + PH (i.e., add PH to the stationary requirement). The pilot commentary for this maneuver indicated that there was an inherent loss of microtexture with only small increases in altitude. Any drift occurring during the bob-up is potentially hazardous, as it could result in collision with obstacles during the bob-down. It could be argued that this presumes that all displays will suffer a loss in microtexture with increased altitude, thus unfairly penalizing a display/vision aid that does not. Since such a display seems unlikely in the near future, it is judged to be safer to err on the conservative side.

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\*The data in Figure 9c indicate an average HQR of 3.9 which is not Level 1. However, it did not seem reasonable to upgrade to TRC for such a small difference. This is discussed in greater detail below.

The rapid hovering turn was also included in this group based on comments from industry that it is a challenging maneuver, and that it will be difficult to stay oriented during rapid turns in conditions of degraded visual cuing. A significant drift during this maneuver could result in a collision with other rotorcraft, or fixed obstacles.

Table 1(3.2) requires an ACAH Response-Type for Level 1 when UCE=2, and TRC when UCE=3 (in addition to Height Hold and Direction Hold). This is a result of certain data interpretations and decisions. As shown above, the data from the flight experiments tended to be more conservative, indicating a need for TRC to obtain Level 1, for maneuvering tasks when UCE=2 (e.g., see Figure 9). These data also indicate that Level 1 is not possible for UCE=3, even with TRC. The ground-based simulation data indicate that ACAH yields Level 1 HQRs for all tasks in UCE=2, and that TRC results in Level 1 when UCE=3 (see Figure 13). It was decided to specify ACAH for UCE=2 and TRC for UCE=3 for the following reasons.

- In good visual conditions (UCE=1), increased stabilization (i.e., from Rate to ACAH to TRC) results in reduced maneuverability. At the same time, the ability to maneuver aggressively with a Rate Response-Type decreases rapidly as the UCE degrades. This is because the poor visual cues so reduce the pilot's ability to stabilize the vehicle that he loses control if aggressive, large-amplitude (attitude) maneuvers are attempted. In the Reference 198 simulation, it was found that this degradation in maneuver capability of the pilot, in reduced UCE, was greater than the degradation in maneuverability of the vehicle due to increased Response-Type. Thus, in a UCE=3, the pilot could maneuver more aggressively with a TRC than with Rate. This should not be surprising since it is actually implied by the definition of VCR and UCE. It does, however, have two important consequences: (1) Since ACAH and TRC Response-Types inherently have less maneuverability than Rate, it was decided to slightly reduce the performance standards for Level 1 maneuvers in DVE (Sections 4.4 and 4.5) since the aggressive standards appropriate for Rate in UCE=1 are impossible to achieve. (2) Since TRC does inhibit maneuvering capability more than ACAH, it was decided to limit TRC to UCE=3 even though it provides marginally better HQRs than ACAH in UCE=2.
- The in-flight HQRs for ACAH in UCE=2 were not far from the Level 1 boundary, i.e., average HQR $\approx$ 4.
- Current vision aids are expected to achieve UCE=2 in typical Degraded Visual Environments (see support for Paragraph 3.1.1). If this implies a requirement for TRC, the cost impact was judged to be significant. Such a large cost differential did not seem worth improving the average HQR from 4 to 2.5.



In addition, there was concern over the ability to produce a high quality TRC due to sensing problems. As a result, the requirement would have driven manufacturers into the development of a high risk SCAS.

- The ground-based simulation results clearly indicated that ACAH is adequate for Level 1 in UCE=2, and that TRC will result in Level 1 for UCE=3. The TRC in the flight experiment was not ideal due to sensor problems (doppler drift).

#### UCE=3; Level 1; TRC+RCDH+RCHH+PH

The basis for requiring TRC for Level 1 in UCE=3 is discussed directly above. The data in Figure 9 indicate the need for Height Hold and Heading Hold. For some tasks (e.g., slalom) it is desirable to disengage Heading Hold, which is the basis for footnote 2 of Table 1(3.2).

The details of the tested TRC are discussed in the supporting data for Paragraph 3.3.12. However, it is important to emphasize that the correspondence between aircraft groundspeed and stick force was a primary attribute of TRC. If TRC is to be used to fly with respect to a moving target (e.g., a ship), the reference velocity must be with respect to the ship. Inclusion of a "trim function" as a method to allow station-keeping with a moving object without holding a constant force would destroy the all-important consonance between speed and stick force.

#### UCE=2; Level 2; Rate+RCDH

A Rate Response-Type is required for Level 2 flying qualities for all MTEs when the UCE=2. However, based on the data in Figure 9, substantial improvements in the HQRs (albeit still Level 2) can be realized by increasing the stabilization to ACAH + RCHH + RCDH. Level 2 can be maintained with this Response-Type even if the UCE degrades to UCE=3.

#### UCE=3; Level 2; ACAH+RCDH+RCHH

The data in Figure 9 indicate that a Rate Response-Type would almost be adequate as a Level 2 requirement for UCE=3. However, because the HQRs barely support, or in some cases do not support, Level 2, it is necessary to require additional stabilization. Unfortunately, no formal data runs were made in UCE=3 for the next two higher levels of stabilization (ACAH and ACAH + RCHH), so the requirement must be based on inferences drawn from the other data in Figure 9. For stationary tasks, Response-Type 3, ACAH + RCDH + RCHH, represents the Level 2 limit for UCE=3, see Figure 9a.

### UCE=3; Level 2; Rate+RCDH+RCHH+PH

For the bob-up/down (Figure 9b), and all maneuvering MTEs (Figure 9c), the Response-Type Rate + PH + RCHH results in ratings that fall near the Level 2-to-3 boundary, and therefore represents the minimum for Level 2, UCE=3. It is not intuitively obvious why PH is required for translating MTEs, and not for stationary MTEs. Pilot commentary indicates that the need for PH centers about problems related to stabilizing the helicopter during the transition from a translation to a stationary task. The need for PH in the bob-up and down MTE relates to the fact that any drift occurring during the bob-up is potentially hazardous, as it could result in collision with obstacles during the bob-down. The likelihood of such a drift occurring in UCE=3 is high, hence the need for PH, even for Level 2.

### 3. Up-and-Away; IMC; Ground-Based Simulation

The bulk of the supporting data for IMC MTEs in Table 2(3.2) comes from eight sources (References 19-23, and 36-38). Table 2 summarizes the pertinent information concerning tasks, test conditions, SAS types evaluated, etc., for these references, plus a handful of other references (References 3, 24, 25, 26, and 27) that also have information (albeit less useful for one reason or another) to increase the data base. Most of the studies are fairly recent, and were conducted on a ground-based simulator, either the Flight Simulator for Advanced Aircraft (FSAA) or the Vertical Motion Simulator (VMS), both at NASA Ames Research Center.

The possible shortcomings inherent to out-the-window, low-speed maneuvering in ground-based simulation tasks (see subsection 4) are, for the most part, not present for up-and-away IMC operations. For example, the very large time delays discussed in subsection 4 are primarily due to digital image generation (DIG) delays, and for IMC evaluations, no visual scene is used.

A final observation about the references of Table 2 is that most included lateral and directional augmentation, and since the tasks were primarily longitudinal in nature, the effects of lateral/directional characteristics can be ruled out as a major factor in the pilots' evaluations.

#### 3a. Constant Airspeed Approaches -- No Flight Director

The results of an extensive simulation to investigate the effects of control position and force gradients, Response-Types, and rotor configurations, are given in Reference 19 for a constant-speed instrument approach task. Pilot ratings for the study of Response-Types (for which the control position gradients were set at the neutral values) are summarized in Figure 14. The rate damping augmentation was a Rate Response-Type in pitch and roll, involving pitch rate feedback only (i.e., no Attitude Hold). Turn-following augmentation increased directional stiffness and allowed the pilots to perform coordinated turns

TABLE 2(3.2.2). SUMMARY OF EXPERIMENTS PROVIDING FORWARD FLIGHT DATA

REF. NO. (YEAR)	TEST AIRCRAFT OR SIMULATOR	ROTOR TYPE	DESCRIPTION OF TASK(S)	TURBULENCE/WINDS	SAS TYPES
19(1980)	FSAA	Articulated, Teetering, Hingeless	VOR Approach (IMC), 60 kt; Missed Approach (No Breakout) Standard Display. Varied Static Control-Position and Force Gradients, SAS Type. 31 Configurations	Light Turbulence: $\sigma_{vg} = 1.5$ ft/sec, $\sigma_{wg} = 3$ ft/sec. No winds.	Pitch/Roll: Rate, RCAH, ACAH. Yaw: Unaugmented, and Turn-Following. Included Input Decoupling.
20(1981)	UH-1H (Flight)	Teetering	MLS 6° Approach to Missed Approach at 200 ft (No Breakout), 60 kt. Raw Data and 3-Cue Flight Directors Used; Varied Control-Position Gradients. Good Flight Data for Correlation with Simulator Results.	Not Detailed; Pilot Ratings Separated Into "Headwind" and "Tailwind and Turbulence" Categories.	Pitch/Roll: Basic UH-1H (Bar On); Rate and ACAH. Input Decoupling Included.
21(1982)	VMS	Teetering	Primarily Concerned with Decelerating Approaches (60 + 15 kt) in IMC. Tasks: 1) Constant-Speed (60 kt); 2) Constant 0.05g Deceleration; 3) Deceleration as a Function of Range; 4) "Constant Attitude" Deceleration; 5), 6), 7) are Like 2), 3), 4), but are Level, Rather than Descending, Decelerations. Two Flight Director Displays Tested.	Most of Evaluations in Calm Air; Some with $\sigma_{vg} = 4.5$ ft/sec, $\sigma_{wg} = 2.25$ ft/sec, and 10-kt Wind that Sheared 90° in Azimuth and from 10 to 2 kt Near the Surface.	Pitch/Roll: Rate, RCAH, ACAH, AC/Velocity Hold. Included Input Decoupling, Yaw Rate Augmentation.
22(1982)	VMS	Teetering	MLS 6° Approach to Missed Approach at 300 ft, 60 kt. No Flight Director. Variations in Control-Position Gradient, Phugoid Damping, Angle-of-Attack Stability, and Pitch Augmentation.	Calm Air, Plus Two Levels of Turbulence: $\sigma_{vg} = 3$ ft/sec, $\sigma_{wg} = 1.5$ ft/sec; and Wind of 10 kt Sheared from 49° Right to 49° Left to 30° Left in 1200 ft.	Pitch: Rate and ACAH. Roll: RCAH; Yaw: Rate Damping. Included Input Decoupling.

TABLE 2(3.2.2). (CONTINUED)

REF. NO. (YEAR)	TEST AIRCRAFT OR SIMULATOR	ROTOR TYPE	DESCRIPTION OF TASK(S)	TURBULENCE/WINDS	SAS TYPES
23(1982)	FSAA	Teetering	MLS 6° Approach to Either Missed Approach or Continued Approach to Offshore Oil Rig, 60 kt. Dual- and Single-Pilot Operations. Displays: 1) Raw Data; 2) Collective Director; 3) Collective/Lateral Directors; 4) 3-Axis Directors.	$\sigma_{vg} = \sigma_{vg} = 3 \text{ ft/sec}$ ; $\sigma_{wg} = 1.5 \text{ ft/sec}$ . Wind Shear as for Ref. 22.	6 SAS Types: 1) Rate Damping Pitch/Roll/Yaw 2) No. (1) Plus Wing-Leveler 3) No. (2) Plus Input Decoupling 4) No. (3) Plus ACAH 5) No. (3) Plus RCAH 6) No. (4) Plus Velocity Hold, Augmented Vertical Damping.
24(1979)	FSAA	Articulated, Teetering, Hingeless	Basically Identical to Ref. 19 Both VMC and IMC Approaches.	None.	Four Levels: 1) Baseline -- No SAS 2) Rate Damping Pitch/Roll/Yaw 3) No. (2) Plus Collective Decoupled 4) No. (3) Plus ACAH
25(1968)	CH-46 (Flight)	Tandem	IMC Cruise/Climb/Descent/Accel/Decel. No Tracking. Modified Cooper Scale Used for Pilot Ratings.	Ambient Conditions (not given).	Variations in $M_u$ , $M_q$ , $M_a$ , $M_{\delta_e}$ .
26(1973)	X-22A (Flight)	— (V/STOL)	VMC and IMC Approaches to Missed Approach (200 ft), 65 and 80 kt. Transition Controller Held Fixed — i.e., Flown Like a Helicopter.	Ambient; Turbulence Effect Ratings Given.	Variations in $M_u$ , $M_q$ , $M_a$ .

TABLE 2(3.2.2). (CONCLUDED)

REF. NO. (YEAR)	TEST AIRCRAFT OR SIMULATOR	ROTOR TYPE	DESCRIPTION OF TASK(S)	TURBULENCE/WINDS	SAS TYPES
3(1984)	FSAA	- (V/STOL)	Transition Study: Decelerate from 250 to 90 kt, IMC, and from 90 kt to Hover, VMC or in Low Visibility. ADI/HSI with Localizer; Transition Controller Used -- i.e., Flown Like a Tilt-Rotor. Only Good Variation of Pitch Attitude Bandwidths.	Moderate Turbulence, $\sigma_{ug} = 4.5$ ft/sec.	RCAH (Transition to STOL and to Hover); ACAH (Transition to Hover Only), in Pitch/Roll. Turn Coordination (High Speed), RCDH (Low Speed) in Yaw. Varied Heave Altitude Command for Transition to Hover.
27(1983)	UH-1H (Flight)	Teetering	Straight and Curved MLS Approached at Varying Flight Path Angles in IMC, Using 3-Axis Flight Director. 18 Evaluation Pilots. Cooper-Harper Scale Greatly Simplified; May Have Influenced Results (see Text).	Not given.	Basic Bell-Bar.
36(1979)	FSAA	Teetering (UH-1H)	Decelerating ILS Approaches on 6° Glideslope. Constant-Attitude Profile from 60 kt to 31 kt (50 ft Decision Height). Decel Rate from 0.045g at Start to 0.035g at End. 3 Pilots, With and Without Flight Director.	Light Turbulence.	UH-1H Bell-Bar and ACAH in Pitch/Roll. Also Included Auto. Collective Schedule with ACAH System. Vertical RCHH.
37(1977)	CH-46 (Flight)	Tandem	Constant-Speed (60 kt) Turn Onto Approach Course, Decelerating Approach on 6° Glideslope Followed by Stabilized Hover, All IMC. Constant-Attitude Profile (Decel Rate 0.08g + 0.04g). Flight Director Included Moving Map Display.	Generally Light, Some Crosswinds and Tailwinds (to 18 kt).	Pitch/Roll: Rate and Attitude SAS (Both Had Unstable Aperiodic Modes at ~40 kt and Above), and Attitude CAS.
38(1985)	NRC (Canada) Bell 205A (Flight)	Teetering	Decelerating ILS Approaches on 6° Glideslope. 50 ft Decision Height, Both Continued and Missed Approached. 5 Pilots.	Light-to-Moderate. (No Details Available).	Rate Damping and ACAH in Pitch/Roll. Rate Damping and RCDH in Yaw.

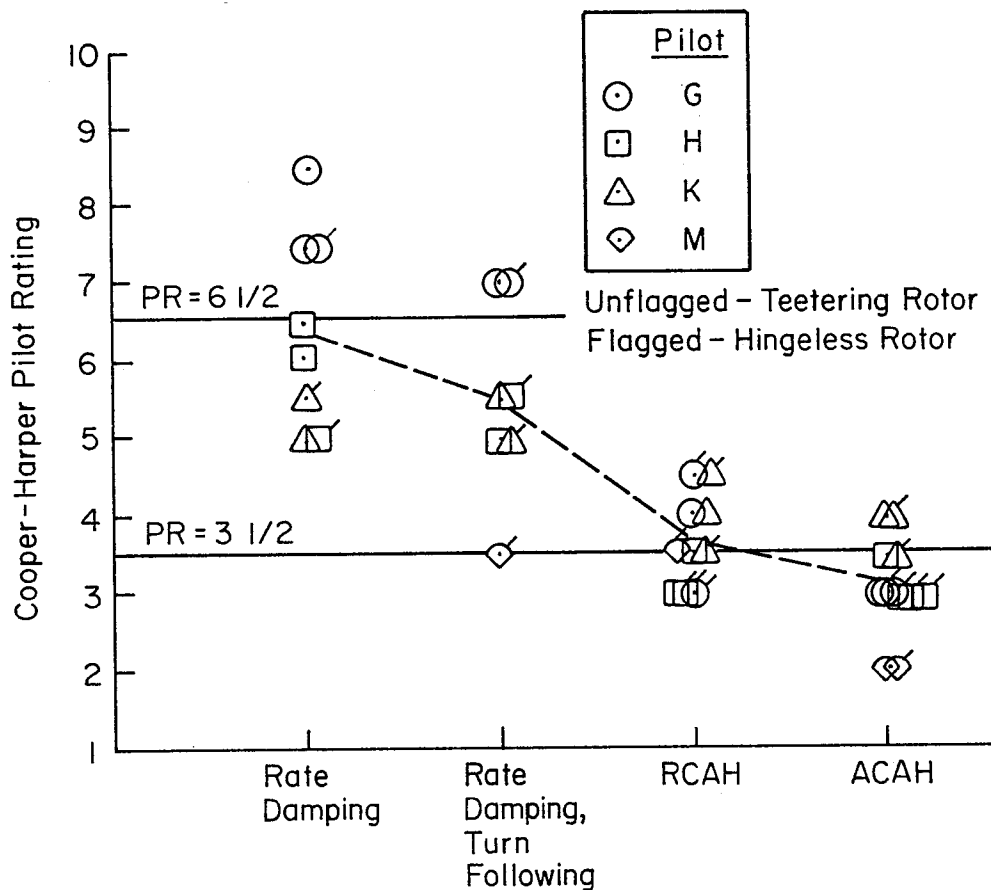


Figure 14(3.2.2). Pilot Ratings from Piloted Simulation of Reference 19. Light Turbulence, No Flight Director

without the use of pedals. The RCAH and ACAH systems were Rate Command/Attitude Hold and Attitude Command/Attitude Hold, respectively, in both pitch and roll. Three rotor types (teetering, hingeless, and articulated) were simulated. The articulated-rotor data are ignored here, however, because of problems in the implementation of that configuration. Specifically, quoting from Reference 19, "the results of the augmentation types for the articulated-rotor configurations were compromised by a nonlinear directional stiffness, which decreased significantly with positive sideslip, even when augmented." Thus only the hingeless- and teetering-rotor data are presented in Figure 14. The rate-damping cases (i.e., no Attitude Hold) are Level 2, and the RCAH system is Level 1, even though the bandwidths are nearly equal. The Reference 19 results indicate that Attitude Hold is necessary for Level 1 operations for IMC approaches. This is reflected in Table 2(3.2).

The in-flight simulation of Reference 20, which utilized the NASA-Army V/STOLAND UH-1H helicopter, provides flight test data for constant speed instrument approaches that can be compared with the ground simulator results of Reference 19. The augmentation and experimental design

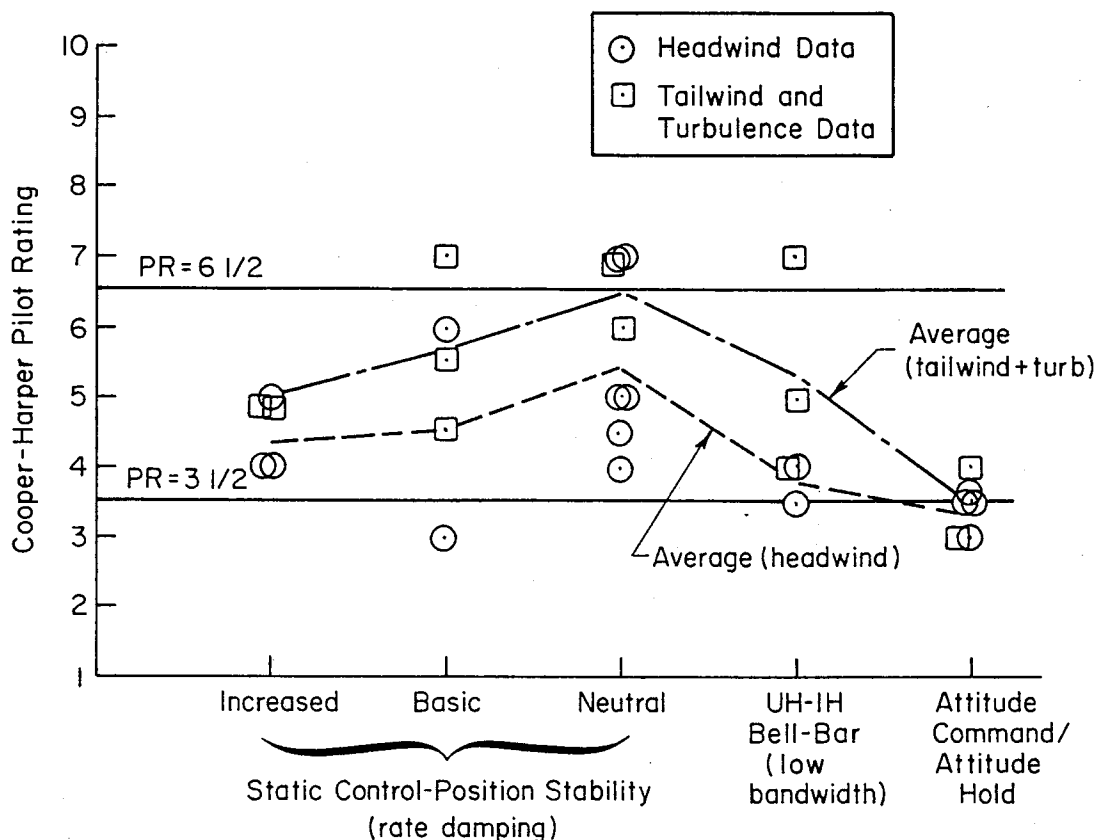


Figure 15(3.2.2). Pilot Ratings from Flight Tests of Reference 20. No Flight Director

were both based on Reference 19. SAS details and transfer functions are not reported in Reference 20, so the rotorcraft responses used later in this discussion are estimates based upon information from Reference 19, plus the dynamics of the augmented UH-1H ("Bell-Bar") from Reference 28. Figure 15 summarizes the pilot ratings for the raw-data (no flight director) approaches. The ratings are separated into headwind and tailwind-and-turbulence runs; for the purposes of this specification, the latter are more interesting. As with the Reference 19 results (Figure 14), there is a clear preference for Attitude Hold (Rate Command/Attitude was not tested in Reference 20).

Reference 22 contains a wealth of data on the effect of static control-position gradient -- i.e., long-term stability. The derivatives  $M_u$ ,  $M_q$ , and  $M_w$  were varied, along with augmentation system types (in pitch only -- Rate Command/Attitude Hold was used in roll for all cases). Figure 16 summarizes the pilot ratings. The baseline rate damping system used an  $M_q$  of -3 rad/sec, resulting in a pitch attitude

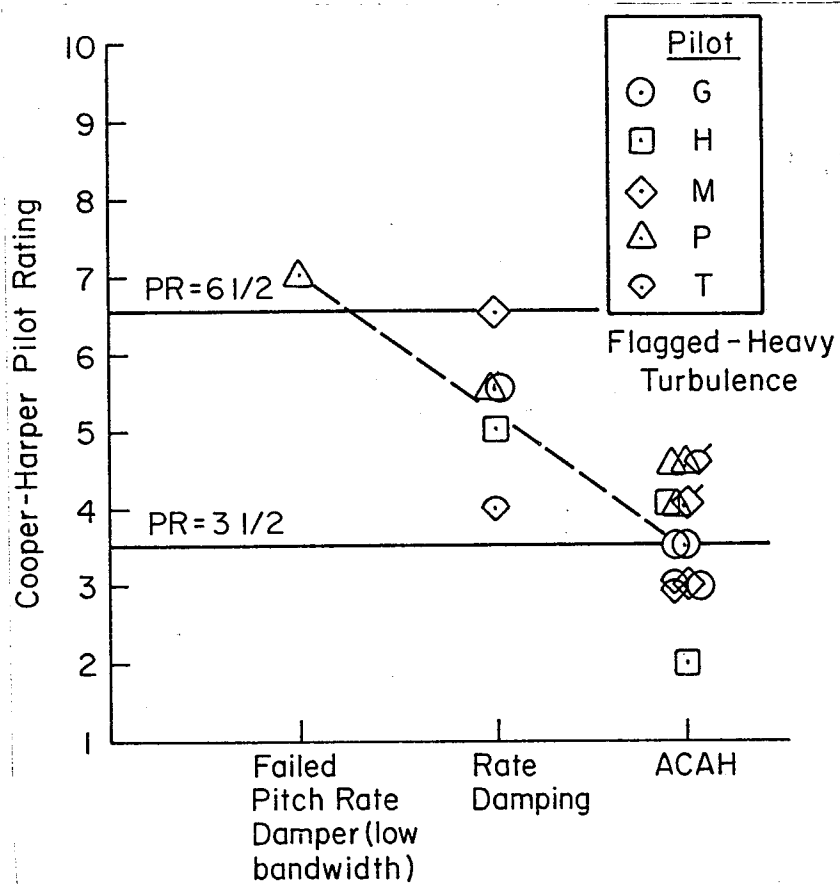


Figure 16(3.2.2). Pilot Rating Results from Piloted Simulation of Reference 22.  $M_{\alpha} = 0$ , Most Neutral Stick Position Gradient (Except ACAH). RCAH in Roll, Light/Moderate Turbulence, No Flight Director

bandwidth of about 3 rad/sec. The single data point in Figure 16 for a "failed pitch rate damper" is for a system with  $M_q$  reduced to -1 rad/sec, for a bandwidth of 1.1 rad/sec. This latter system may be viewed as a low-bandwidth configuration for specification purposes, but other factors may have influenced the pilot's opinion, since controller characteristics were optimized for the baseline system and were not modified for the failed-damper case. In addition, of course, one would generally tend to weigh a single rating very lightly; but the trend shown in Figure 16, which is for the turbulence-plus-wind environment, is also found in the ratings for the calm-air environment. Finally, Figure 16 again shows an ACAH system to be markedly better than rate damping, lending further support for Table 2(3.2).

The critical issue of crew size was examined in the piloted simulation of Reference 23. "Dual-pilot" operation meant that the pilot's



primary concern was controlling the aircraft, assuming that a second crew member would handle ancillary tasks such as communication, radio and transponder frequency switching, copying of clearances, etc. In single-pilot operation, such tasks also had to be performed by the evaluation pilot during a standard approach. In addition, from Reference 23, "communication with other helicopters in the area was simulated; it sometimes interfered with the evaluation pilot's ability to communicate freely with the appropriate controller at the appropriate time, thereby causing additional stress." Missed-approach procedures frequently were changed from the procedures published on the pilots' approach charts to further increase workload. In terms of a realistic operating environment, this simulation most closely approximates what a helicopter pilot would expect in the real world.

Pilot rating data from Reference 23 are shown in Figure 17, for the raw-data (azimuth and elevation errors and DME) display. In general, the single-pilot-with-missed-approach ratings are poorest (highest in value). Pilot ratings improve with increasing augmentation sophistication (the "dip" in the mean-rating lines for the rate damping/wing leveler case should not be weighed too heavily; note that pilot G, who gave the harshest ratings of the four pilots, did not fly this configuration). These data suggest that even an Attitude Response-Type may not be Level 1 in a high workload single-pilot environment. Attitude Hold is clearly necessary.

The simulation of Reference 24 is very similar in detail to that of Reference 19, in that the task involved VOR approaches. The major differences are that the Reference 24 simulation did not include turbulence, and the full transfer functions for the rotorcraft evaluated were not published. (The eigenvalues, or characteristic modes, were included in Reference 24, and thus some conclusions can be reached. However, without the control/response numerators, such details as pitch attitude bandwidth cannot be determined.)

Reference 24 included both VMC and IMC approaches and evaluations. A summary of the pilot ratings is given in Figure 18. The following observations can be made:

- Pilot ratings improve in VMC, though not by much. Since the task is a VOR approach, it is likely that the pilots flew mostly head-down to follow the published procedure.
- The basic (unaugmented) configurations were all rated Level 2-to-3, both VMC and IMC. The bandwidths of these systems were estimated to be something less than 3 rad/sec (short-term natural frequencies ranged from 1.2 to 2.5 rad/sec), and the teetering- and hingeless-rotor configurations had unstable low-frequency (phugoid) modes.
- Rate augmentation in pitch, roll and yaw ("rate damp" in Figure 18) improved the bandwidths and resulted in Level 1-to-2 ratings for the teetering-

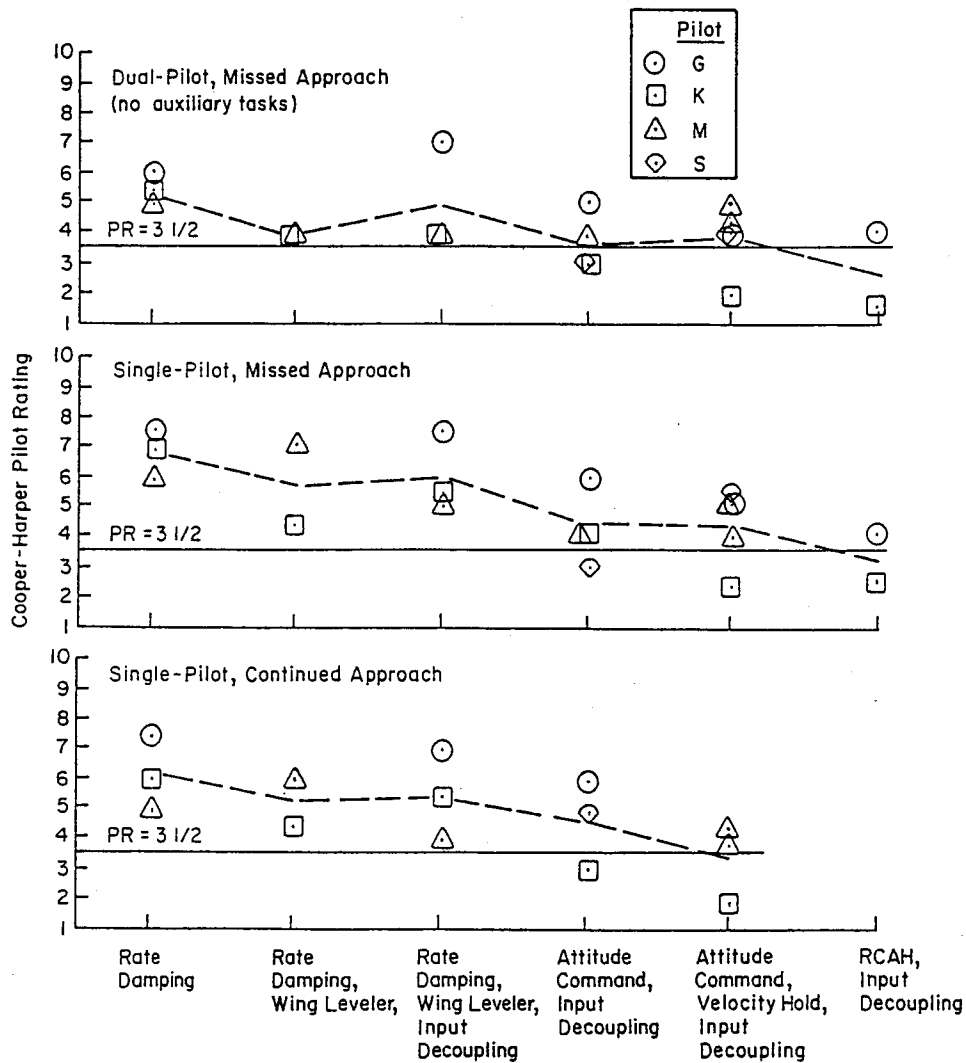


Figure 17(3.2.2). Effect of Crew Size, Response-Type, and Task (Simulation of Reference 23; No Flight Director)

rotor case. Strong pitch-collective coupling kept the ratings for the other two configurations from improving noticeably.

- Collective decoupling ("DCPL" in Figure 18) improved the hingeless-rotor helicopter, but the crossfeeds used for the articulated-rotor case produced an aperiodic divergence so that, overall, there was no improvement in pilot ratings for this configuration.
- Use of ACAH in pitch and roll ("ATT COMM") improved all the configurations to some extent, with the articulated-rotor helicopter showing the most marked improvement.

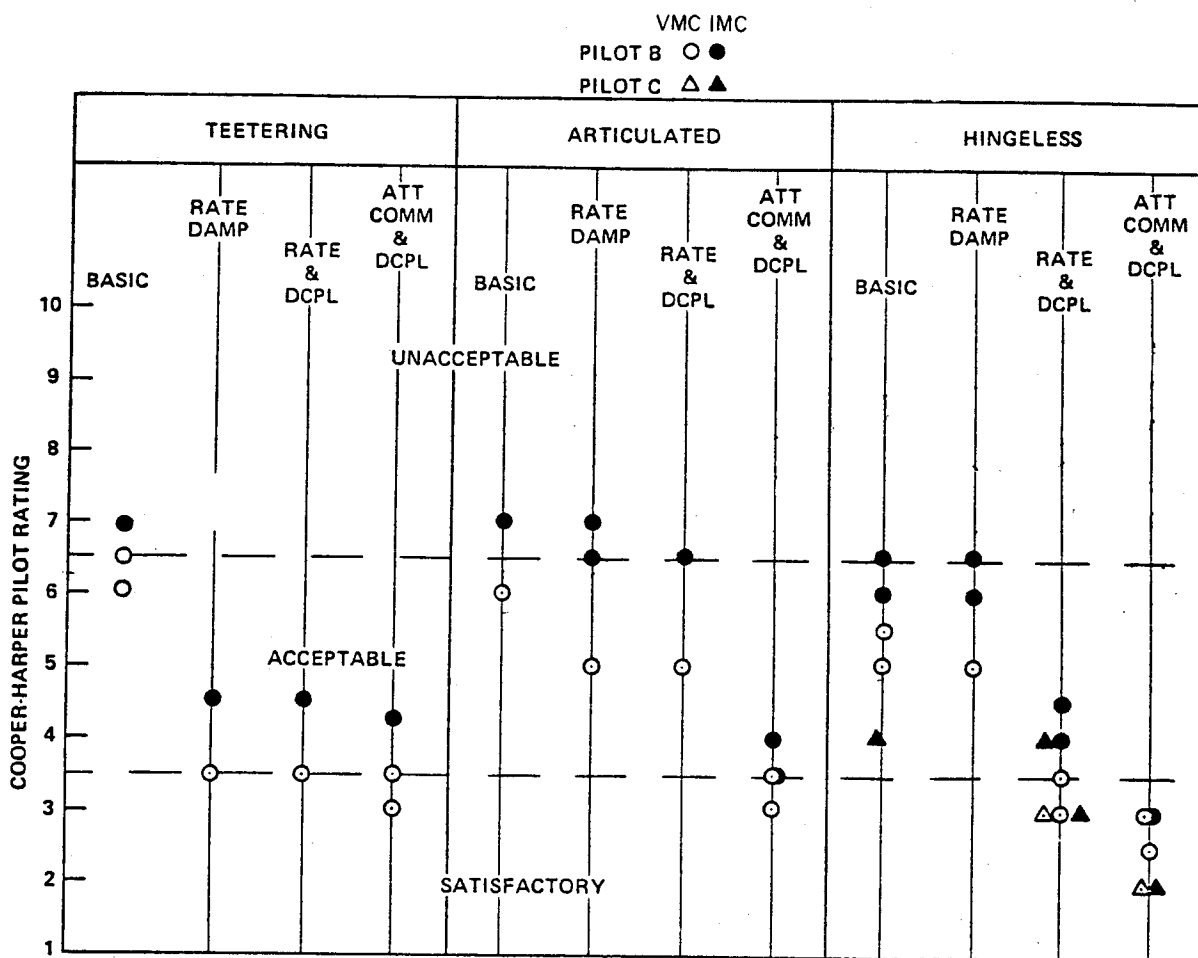


Figure 18(3.2.2). Pilot Rating Summary from Reference 24.  
No Turbulence (No Flight Director)

The above observations are in agreement with the other data discussed above.

Reference 25 reported on a flight investigation of effects of variations in  $M_w$ ,  $M_q$ , and  $M_{\delta_e}$  on flying qualities in IMC. The task involved non-precision maneuvers and thus is considered to be representative of cruise/climb/descent Mission-Task-Elements. Figure 19 shows the pilot ratings on a crossplot of  $M_q$  vs.  $M_{\alpha}$ . It is important to note that these ratings are based on a variation of the old Cooper (or NASA) pilot rating scale, as shown in Table 3, and not on the Cooper-Harper rating scale. Some interpretation is necessary before assigning flying qualities Levels to Table 3. In addition, all the cases of Figure 19 were flown with a fixed value of  $M_{\delta_e}$ . Since the control sensitivity was not optimized for each case, it may have influenced the pilots' opinions. The conclusion to be drawn from Figure 19 is that Level 1 ratings are obtainable for Rate Response-Types (without Attitude Hold) for VMC cruise/climb/descent Mission-Task-Elements. This conclusion is reflected in Table 2(3.2).

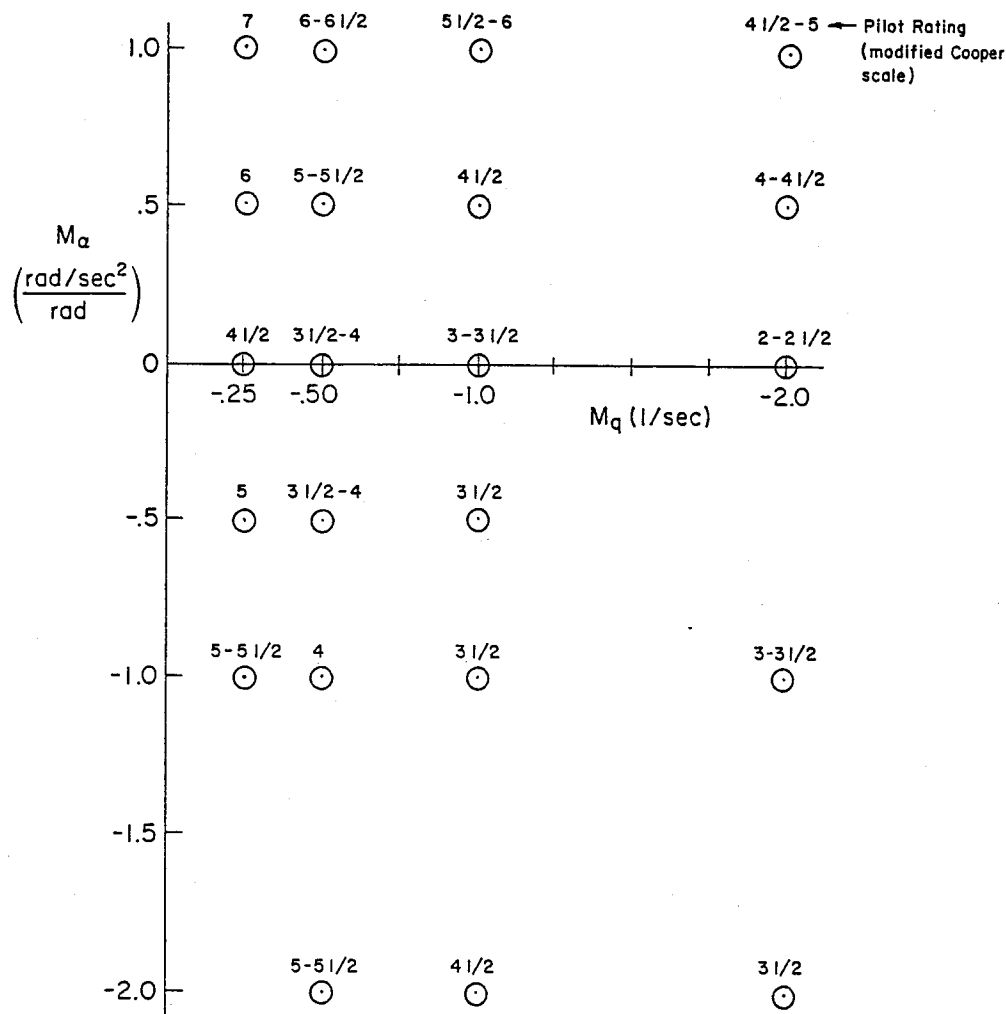


Figure 19(3.2.2). Pilot Ratings from Flight Tests of Reference 25.  
 $M_{\delta_e} = -0.3 \text{ rad/sec}^2/\text{in.}; M_u=0.$

Reference 26 was a flight program using the X-22A variable-stability aircraft to investigate the effects of changes in short-term response. The derivatives  $M_u$ ,  $M_q$ , and  $M_w$  were varied and the pilots were allowed to select an optimum stick sensitivity for each configuration. Approaches were flown at 65 kt, at 6 and 9 deg glidepath angles, and at 80 kt at a glidepath angle of 7 deg (most of the evaluations were for the 65-kt, 9-deg approaches). Results from the 65-kt approaches are given in Figure 20 for those runs that were subjectively considered to be made in moderate turbulence. Letters beside the pilot ratings in Figure 20 are pilot turbulence effect ratings, Table 4.

TABLE 3(3.2.2). EXPANDED COOPER (NASA) PILOT-RATING SYSTEM (FROM REFERENCE 25)

Operating conditions	Adjective rating	Numerical rating	Description	Added descriptive phrases	Task performance	Primary mission accomplished	Can be landed
Normal operation	Satisfactory	1	Excellent, includes optimum	Very stable, disturbances well damped Stable, not easily disturbed	Precise	Yes	Yes
		2	Good, pleasant to fly			Yes	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics		Precise, most of time	Yes	Yes
Emergency operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	Occasional large, but controllable upsets Distractions not tolerable, full-time control required Must concentrate on a single (pitch) axis only	Few errors Difficult to maintain accuracy Poor	Yes	Yes
		5	Unacceptable for normal operation			Doubtful	Yes
		6	Acceptable for emergency condition only <sup>1</sup>			Doubtful	Yes
No operation	Unacceptable	7	Unacceptable even for emergency condition <sup>1</sup>	Difficult to control attitude, with possible loss of control Very easy to lose control (loss of control imminent)	Ignored	No	Doubtful
		8	Unacceptable - dangerous			No	No
		9	Unacceptable - uncontrollable			No	No
	Catastrophic	10	Motions possibly violent enough to prevent pilot escape			No	No

<sup>1</sup> Failure of stability augmenter.

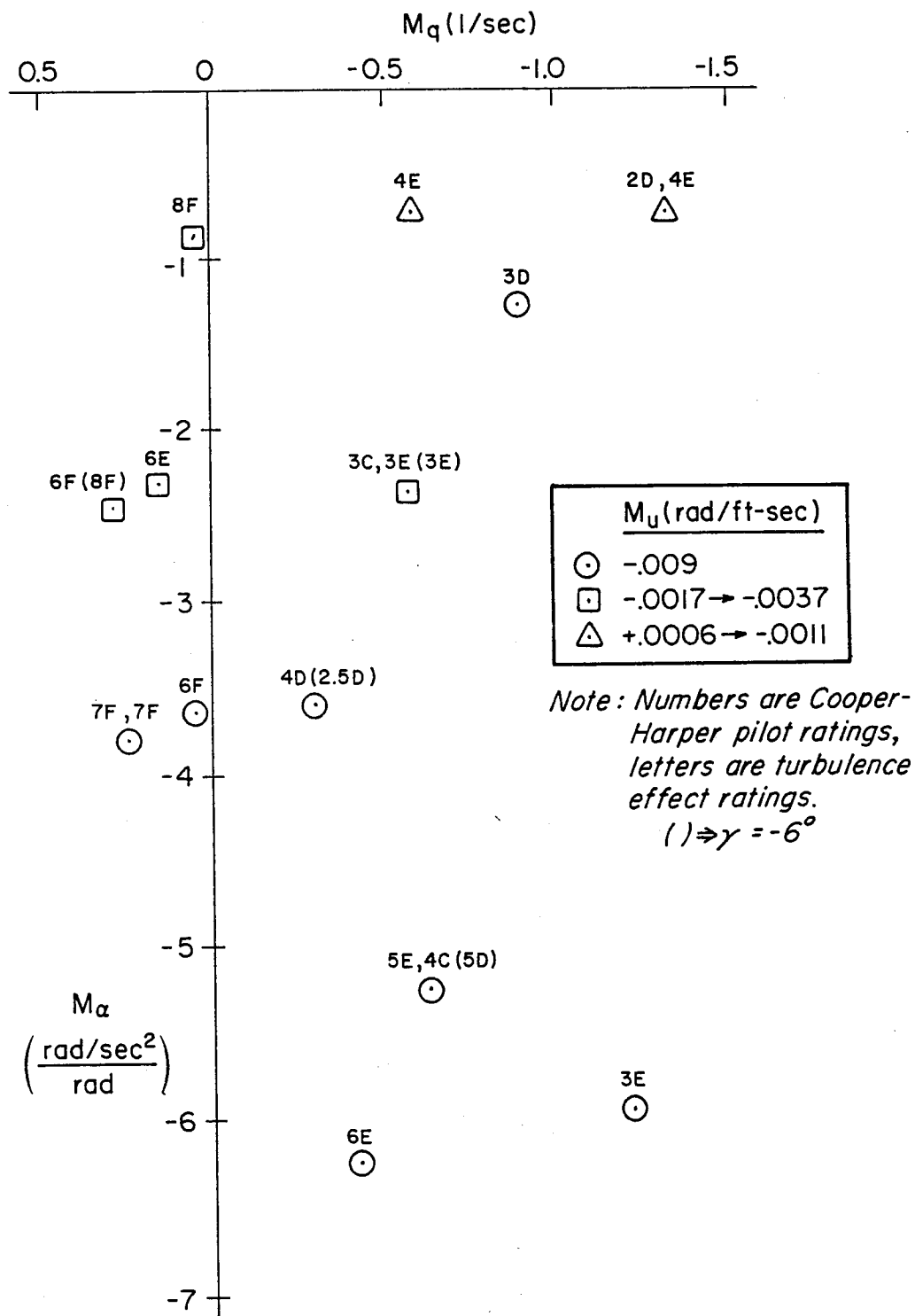


Figure 20(3.2.2). Results of X-22A Study of Reference 26. Moderate Turbulence; 65 kt;  $\gamma = -9$  deg (Except Where Noted Otherwise)

TABLE 4(3.2.2). TURBULENCE EFFECT RATING SCALE  
(FROM REFERENCE 26)

INCREASE OF PILOT EFFORT WITH TURBULENCE	DETERIORATION OF TASK PERFORMANCE WITH TURBULENCE	RATING
NO SIGNIFICANT INCREASE	NO SIGNIFICANT DETERIORATION	A
MORE EFFORT REQUIRED	NO SIGNIFICANT DETERIORATION	B
	MINOR	C
	MODERATE	D
	MODERATE	E
BEST EFFORTS REQUIRED	MAJOR (BUT EVALUATION TASKS CAN STILL BE ACCOMPLISHED)	F
	LARGE (SOME TASKS CANNOT BE PERFORMED)	G
UNABLE TO PERFORM TASKS		H

Use of the data shown in Figure 20 in this specification is limited because the X-22A was flown well on the frontside of the power required-speed curve. Thus it could be flown, in the short term at least, more like a conventional airplane (see, e.g., Reference 2) than like a helicopter (i.e., collective to control flight path and cyclic to control airspeed). A review of pilot comments in Reference 26 supports this conclusion. Figure 20 shows that Level 1 ratings are obtainable for this type of aircraft with a Rate Response-Type.

### 3b. Effect of Flight Directors on Flying Qualities for IMC Approaches

All of the results described thus far for IMC forward flight have not included flight directors. In addition, no data for decelerating IMC flight were reviewed. All references in Table 2 that involved decelerating IMC approaches also involved flight directors, and thus are discussed in the following paragraphs.

#### Constant-Speed Approaches

References 20 and 23 evaluated the effect of flight directors in performing precision 60-kt, 6 deg MLS approaches. Three-axis flight directors (longitudinal and lateral cyclic and collective directors) were integrated into the ADI.

The raw-data (no flight director) pilot ratings from Reference 20 were discussed earlier and summarized in Figure 14; they are replotted in Figure 21 along with the ratings for approaches flown with a three-axis mechanical flight director. There is no consistent improvement in pilot ratings due to the flight director. Likewise, flight directors were found to have a minimal (order of 1 pilot rating or less) influence

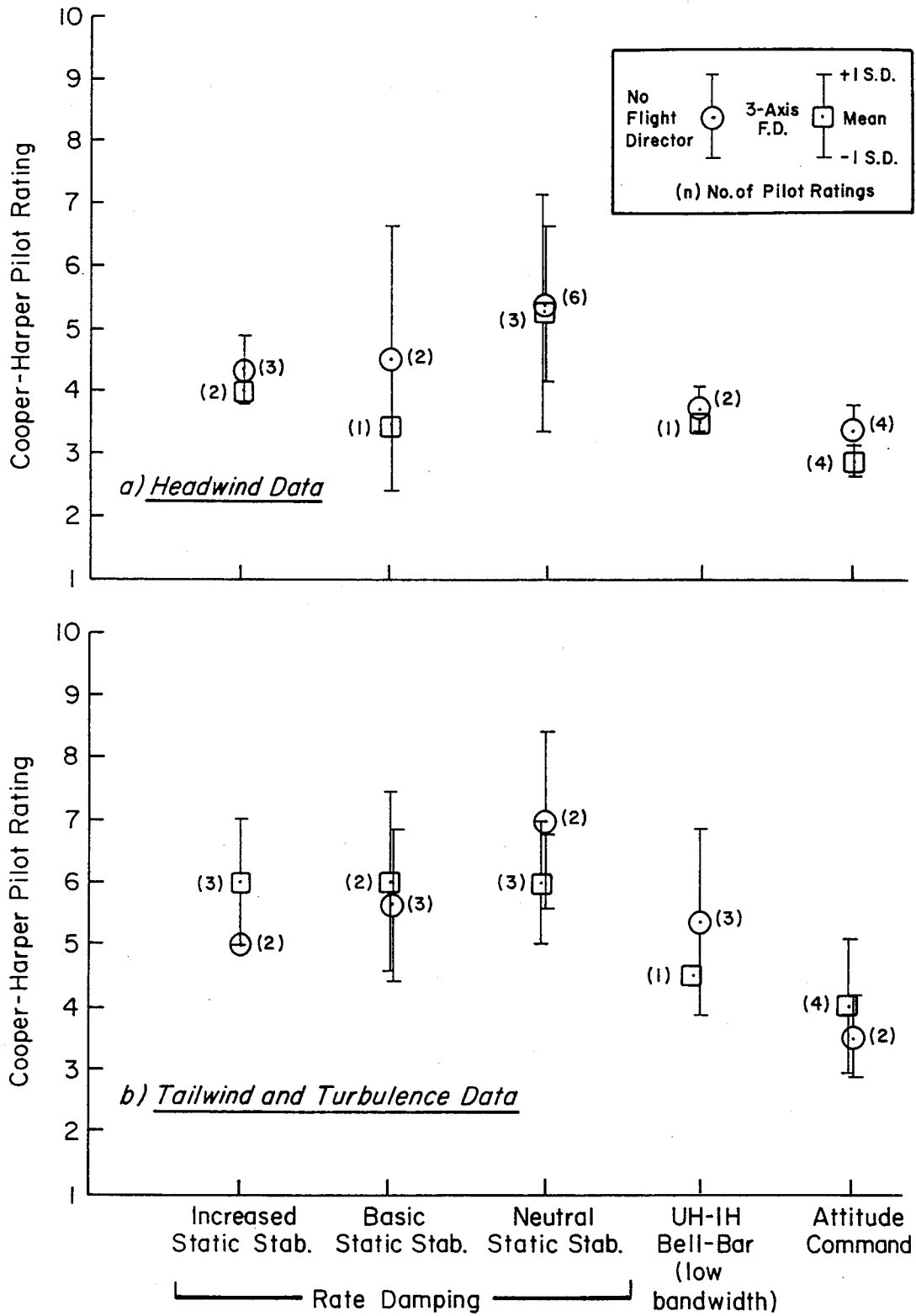


Figure 21(3.2.2). Effect of Three-Cue Flight Director on Pilot Ratings for Constant-Speed MLS Approach (UH-1H Flight Data, Reference 20)



on pilot ratings in the simulation of Reference 23 (Figure 22). The data of Figures 21 and 22 suggest that, for constant-speed approaches, attitude stabilization is considerably more beneficial for obtaining satisfactory flying qualities than a flight director. Therefore, a flight director has not been specified as a requirement for constant-speed IMC approaches in Table 2(3.2), though Attitude Hold is required.

The V/STOLAND UH-1H helicopter was used in Reference 27 to evaluate several different MLS approach profiles in IMC. The data from this experiment are at odds with all of the above discussed results, in that it indicates that the basic unaugmented UH-1H is adequate for IMC approaches (Figure 23). The same basic aircraft (UH-1H Bell Bar) was also evaluated in Reference 20, where it was found to be Level 2 without a flight director (Figure 14).

Examination of the details in Reference 27 suggests some possible reasons for the good ratings. Lack of suitably tight controls on the task (i.e., allowing the pilots to fly at "whatever airspeed they wished") may have resulted in lower workload. In addition, the pilot rating scale used in Reference 27 was not the standard Cooper-Harper scale; in order to "simplify the pilot rating process for the guest pilots," the scale was modified considerably (Figure 24). This scale has lost the performance definitions of the Cooper-Harper scale -- i.e., "desired," "inadequate," etc. -- and has changed the definitions. For example, a rating of 4 is considerably worse on the modified scale than on the Cooper-Harper scale, as shown in the following comparison table.

<u>Cooper-Harper</u> <u>Rating = 4</u>	<u>Modified Scale (Figure 24)</u> <u>Rating = 4-6</u>
Deficiencies warrant improvement	Requires improvement (CHPR = 7-9)
Minor but annoying deficiencies	Moderately objectionable deficiencies (CHPR = 5)
Desired performance requires moderate pilot compensation	Considerable demands on pilot (CHPR = 5 and 8)

This table indicates that the modified scale tends to bias the ratings into the 1-3 range, which probably explains the discrepancy between the data of References 20 and 27. Based on the above discussion, the Reference 27 results are not reflected in Table 2(3.2).

### Decelerating Approaches

For the decelerating-approach study of Reference 21, an electronic ADI (EADI) replaced the mechanical instrument. As Table 2 indicates, the bulk of the evaluations were conducted in calm-air conditions. The data of Figure 25 are for moderate turbulence, with a 10-kt wind that

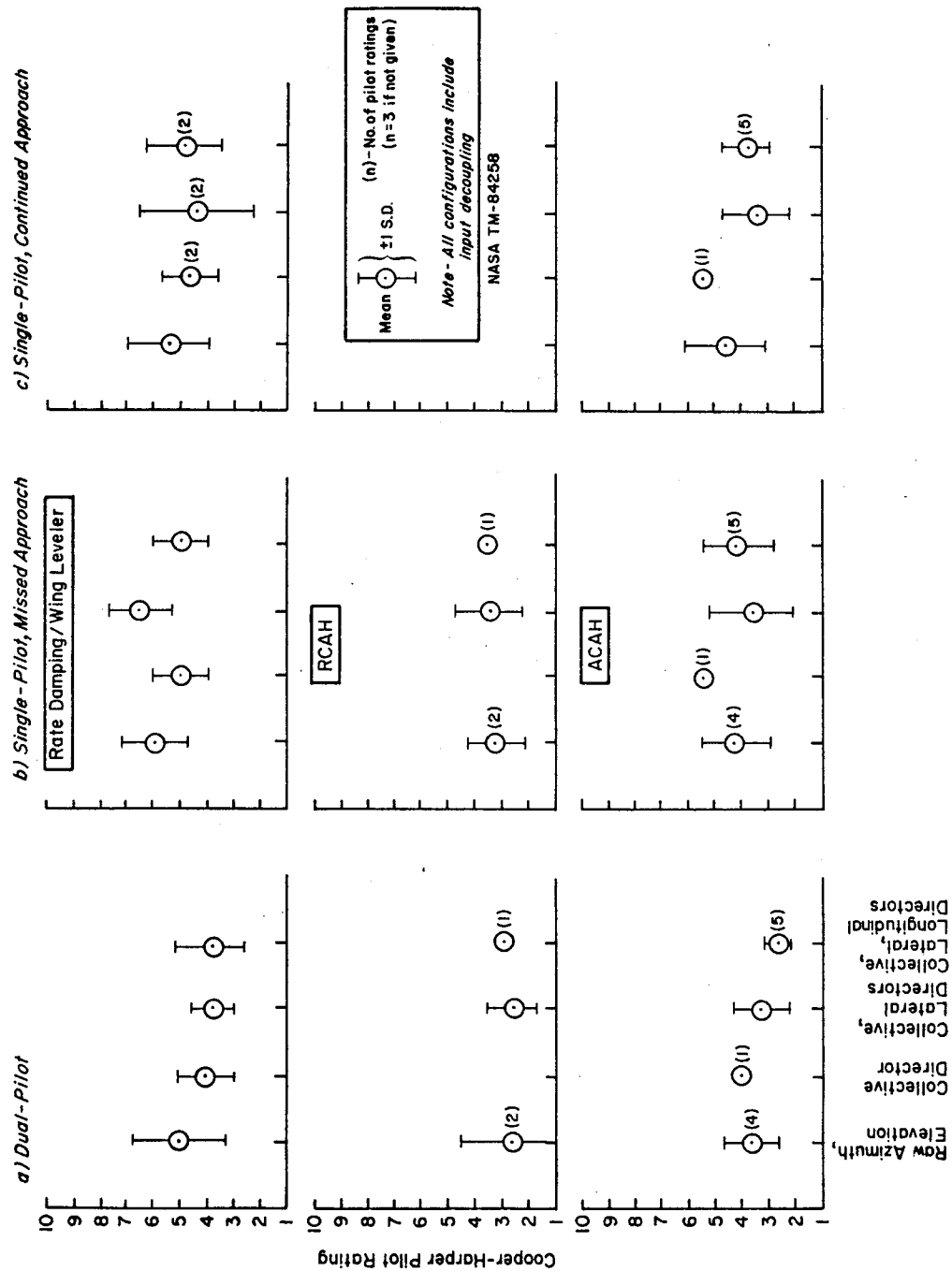


Figure 22(3.2.2). Influence of Flight Directors on Pilot Ratings for Constant-Speed MLS Approach (Simulation Data, Reference 23)

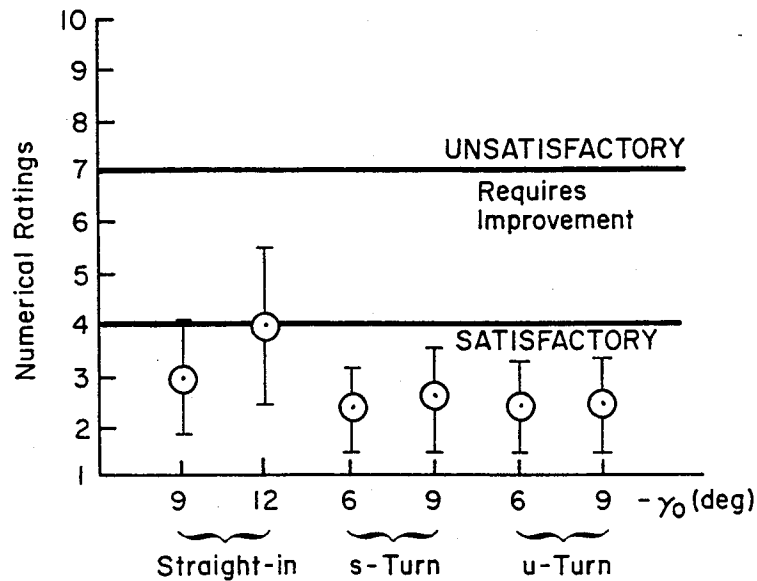


Figure 23(3.2.2). Pilot Acceptability Ratings from UH-1H Flight Test of Reference 27

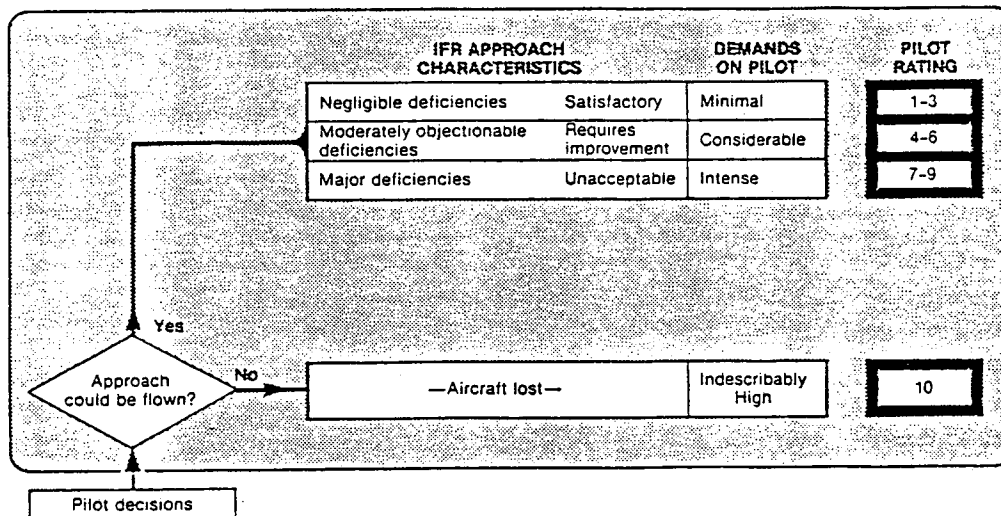


Figure 24(3.2.2). Modified Cooper-Harper Pilot Rating Scale Used in Reference 27

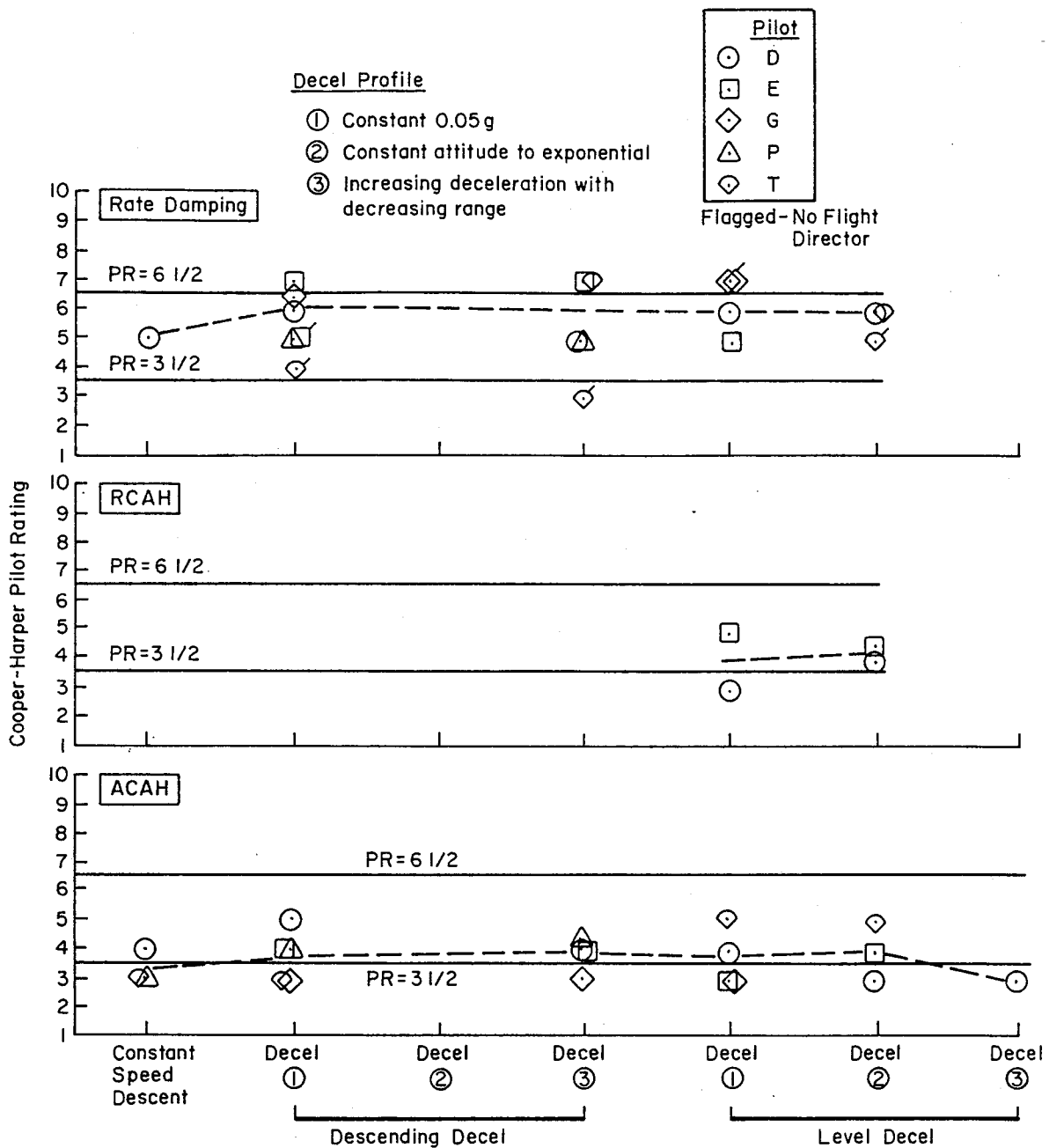


Figure 25(3.2.2). Pilot Rating Results from Simulation of Reference 21. Three-Axis Flight Director with Conventional ADI. (Data Supplied By Author of Reference 21)

sheared 90 deg in azimuth at about 1 nautical mile out and from 10 to 20 kt between 200 ft and 0 ft altitude. These previously unpublished data were supplied by the author of Reference 21. Primary augmentation types were rate damping and ACAH, with a few runs using an RCAH system. The mean lines drawn on Figure 25 show that there was no clear preference for a particular deceleration scheme, and that the ACAH system obtained consistently better pilot ratings.

A few ratings were obtained for the rate-damping system with no flight director (flagged symbols in Figure 25); interestingly, little change in pilot ratings occurs. The changes that do occur show an improvement in pilot rating when the approaches were flown without the flight director. This may, however, have been a result of the design of the director control laws for the rate-damping configuration, which had an unstable phugoid mode. From Reference 21, "most of the evaluations with the rate-damping control system were conducted using a pitch flight director that included low-frequency equalization to stabilize the phugoid. With this design technique, a low-frequency oscillation in speed was evident... and pilot comments noted that the director appeared to be oscillatory and to require significant pitch inputs.... The important point is that the performance of those systems that have stability and control difficulties is much more sensitive to the details of the flight director design than is the performance of benign systems (e.g., attitude command)...." Thus, the flight director is seen to be ineffective for improving flying quality deficiencies. This result was also noted in Reference 50.

Three deceleration profiles, listed on Figure 25, were evaluated in the Reference 21 study, but there is little difference in pilot ratings for the different profiles. Indeed, the profile showing the most marked improvement in pilot rating occurred when the flight director was removed (increasing deceleration with decreasing range; Pilot T rated it a 7 with flight director and 3 without). This profile was based on piloting technique in helicopter visual approaches (see, e.g., References 21 and 39), and it is conceivable that this type of approach is not well-suited for a flight director. In addition, the flight director would have commanded quite high pitch attitudes near hover, and Pilot T may have chosen to fly the situation-only approaches with a lower deceleration rate than the flight director demanded.

The Reference 36 simulation program used the basic dynamics of the UH-1H helicopter with stabilizer bar on (Bell-Bar), and included an Attitude Command/Attitude Hold (ACAH) system. In addition, an automatic collective command system was evaluated in conjunction with the ACAH system. Approaches were flown from 60 kt to approximately 31 kt (corresponding to a decision height of 50 ft) in light turbulence and without winds. The deceleration profile involved a 6 deg approach with an essentially constant-attitude deceleration, resulting in an initial deceleration of about 0.045g and slowly decreasing to 0.035g near hover. Pilot ratings, summarized in Figure 26, show that the basic Bell-Bar was Level 3 (compare this with the constant-speed ratings from flight test, Figure 21, where the Bell-Bar configuration was Level 2) with or without the flight director. A consistent improvement in pilot ratings was obtained for all Response-Types by using the flight director, but there

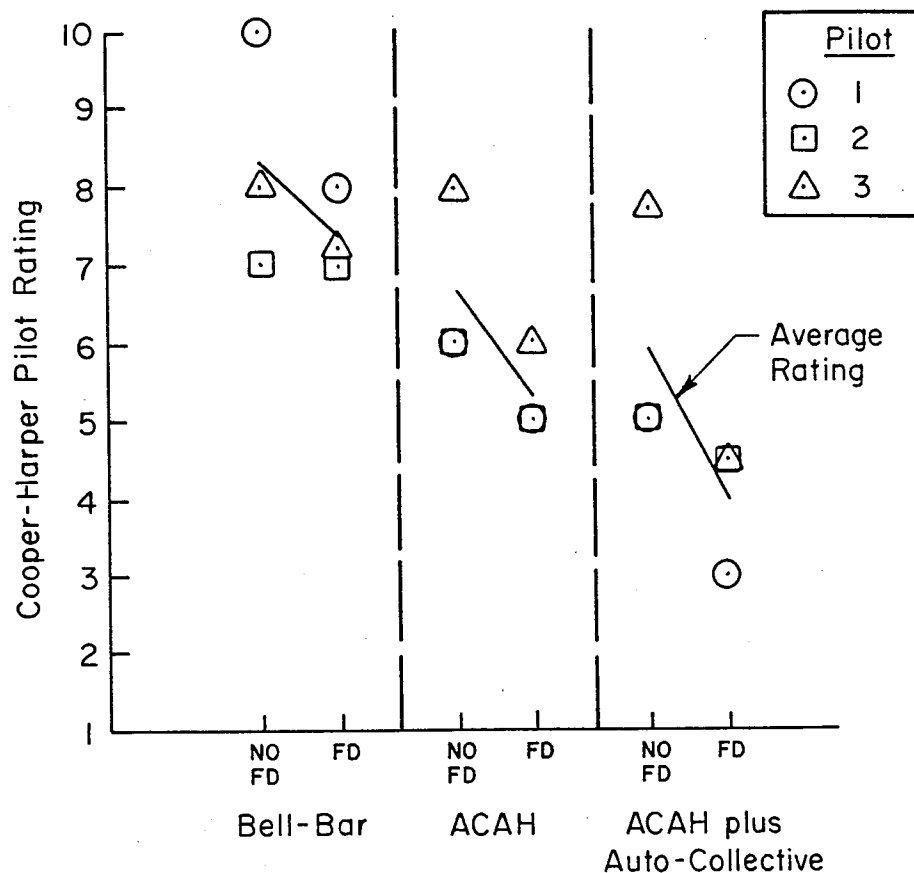
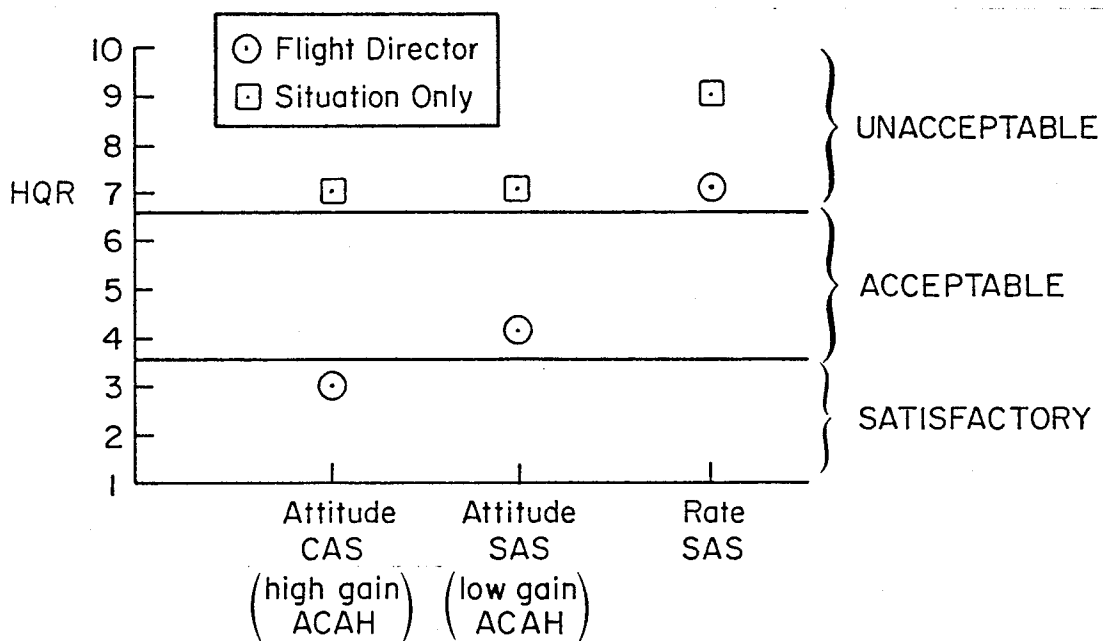


Figure 26(3.2.2). Pilot Ratings for Decelerating Approach (0.04g → 0.035g) from Simulation of Reference 36

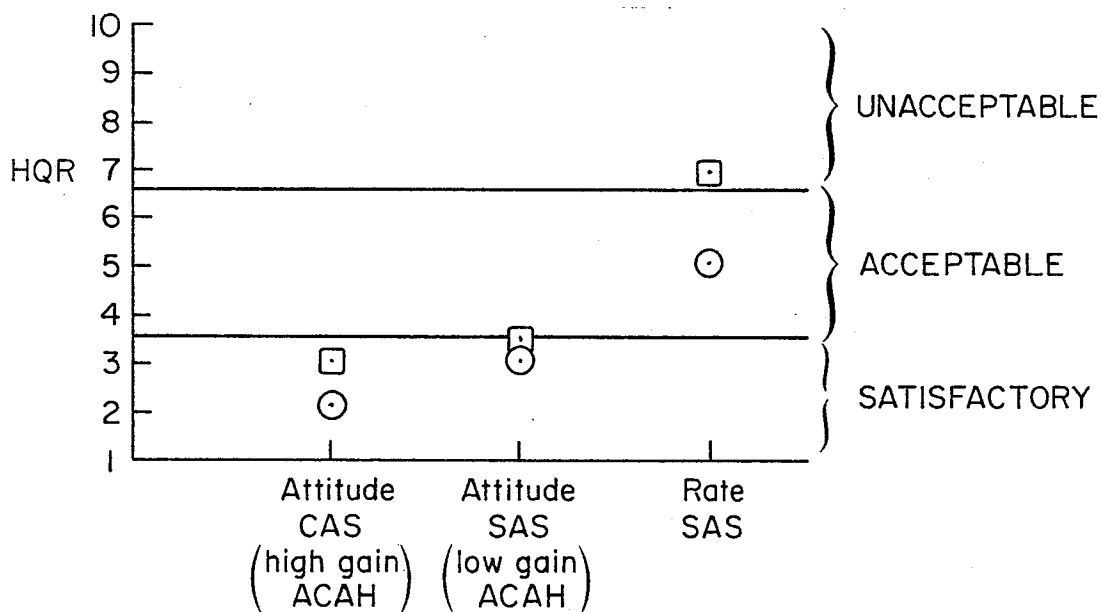
was not a change in flying qualities Level. The dynamics of the basic UH-1H are Level 2 (shown in Paragraph 3.4.1.1) and the dynamics of the ACAH system tested in Reference 36 would also be Level 2 by the requirements of Paragraph 3.4.1.1. It is therefore not surprising that Level 2 pilot ratings were obtained in this experiment, even with the proper Response-Type (i.e., ACAH). Figure 26 shows that automatic collective scheduling improved the pilot ratings somewhat, but not sufficiently to overcome the poor dynamics.

The flight research program of Reference 37, using the NASA Langley variable-stability CH-46, involved decelerations from 50 kt to an instrument hover. The deceleration profile required essentially constant attitude, resulting in deceleration rates varying from approximately 0.08g at the beginning to approximately 0.04g at the end. The flight director was integrated into the ADI, and in addition, a moving-map display showed runway and hover pad outlines. Three augmentation systems were tested: low-gain rate and ACAH systems, and a high-gain model-following ACAH system.

Pilot rating results from Reference 37 are summarized in Figure 27. Separate ratings were obtained for an initial, constant-speed turning task, and for the deceleration and stabilized hover. The rate system



*a) Approach Task Including Deceleration and Hover*



*b) Approach Task Excluding Deceleration and Hover*

Figure 27(3.2.2). Pilot Ratings from Instrument Approach Study of Reference 37 (CH-46; Deceleration Rates of 0.08g → 0.04g, and Instrument Hover)

(Rate SAS) was Level 2 or worse for all phases; a contributing factor was an aperiodic divergence that existed in the longitudinal axis at higher airspeeds. The low-gain attitude system (Attitude SAS) and the high-gain Attitude CAS were Level 1 for the constant-speed portion of the task, and the use of a flight director made little difference (Figure 27b). For the deceleration and hover, operations with situation information only (i.e., without the flight director) were Level 3 -- in fact, Reference 37 reports that "the task could not be completed." Improvements in these systems with a flight director are very dramatic: as Figure 27a shows, pilot ratings improved by 3 or 4 points. This is the only example found where use of a flight director improves pilot rating Levels (e.g., from Level 3 to Level 1 for the Attitude CAS). Unfortunately, it is impossible to measure the influence of the instrument hover portion of the task on the pilot ratings, but the data of Figure 27 strongly support the need for a flight director for decelerating approaches.

The final set of data for decelerating approaches was provided by the National Research Council of Canada's National Aeronautical Establishment (NAE). The data from this flight experiment, using the Bell 205A variable-stability helicopter (Reference 38), are shown in Figure 28. The task involved decelerations on a 6 deg glide slope from 60 kt to approximately 22 kt, and a decision height of 50 ft. The deceleration profile was roughly constant-attitude with a rate of about 0.1g at the beginning. Rate-damping and ACAH systems in pitch and roll were flown; runs were made with and without heading hold (HH) in yaw. Somewhat surprisingly, the heading hold function made little difference on pilot ratings. However, the flight director was worth about 2 pilot rating points, which is generally consistent with Figures 26 and 27.

In addition to Cooper-Harper pilot ratings, the pilots in the NAE study of Reference 38 assigned Certification Level Ratings for civil certification purposes. An example of the comparison between these ratings and Cooper-Harper ratings is shown in Figure 29. No configuration was considered certifiable for single-pilot operations, even if the Cooper-Harper rating assigned was Level 1. The evaluations were flown in a dual-pilot environment, i.e., no secondary tasks.

In summary, for constant-speed approaches (Figures 21 and 22), flight directors have minimal effect, while for decelerating approaches (Figures 25, 26, 27, and 28), a flight director is essential and is worth between 1 and 4 pilot rating points. This conclusion was also reached in Reference 40, which is a comprehensive review of all control/display aspects of helicopter decelerating instrument approaches.

Based on these results, a 3-cue flight director is required in Table 2(3.2) for decelerating IMC approaches.



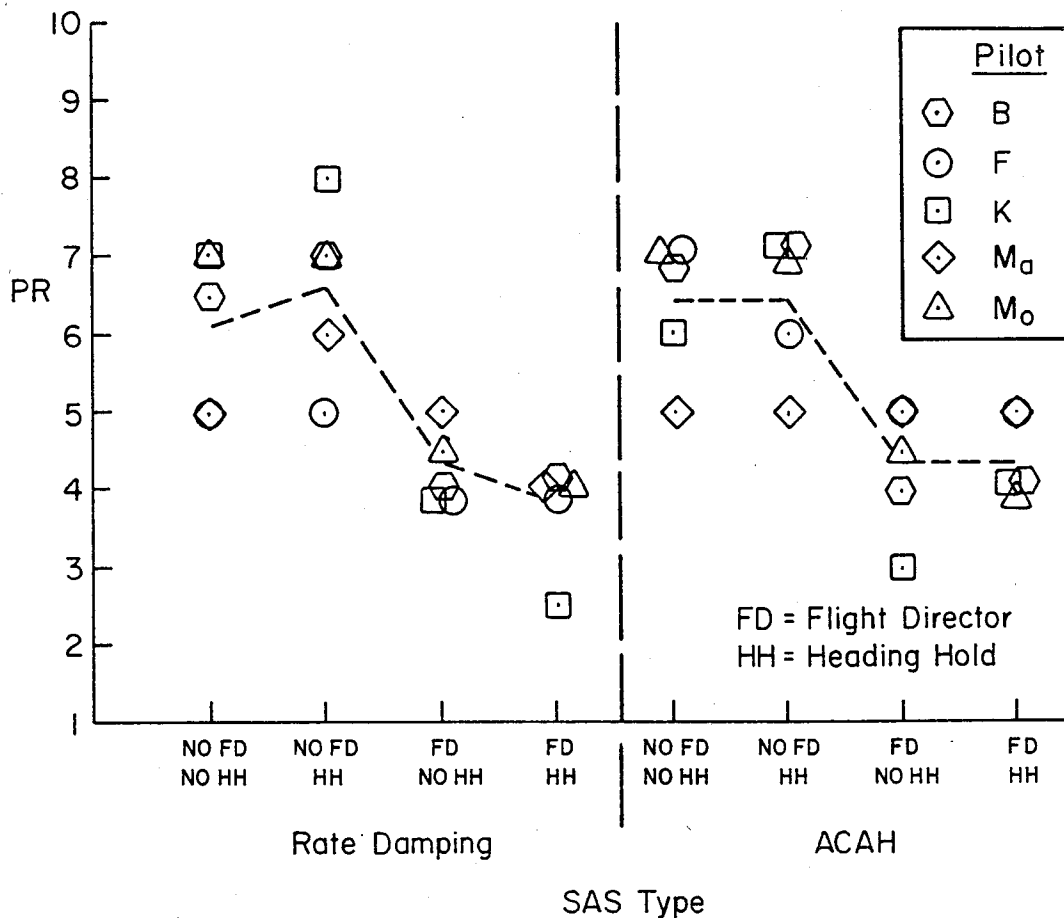


Figure 28(3.2.2). Pilot Ratings from NAE Flight Tests with Bell 205A (Reference 38; Deceleration Rate of 0.1g at Start, Constant-Attitude)

#### 4. Near Earth, UCE=1, Single Pilot, Ground-Based Simulation (References 141 and 155)

The supporting data for "hovering tasks involving divided attention operation" with UCE=1 in Table 1(3.1) is given in this subsection. Two piloted simulations have been conducted by the Army Aeroflightdynamics Directorate to investigate the handling qualities requirements for, and workload involved in, single-pilot NOE combat. The Single Pilot Advanced Cab Engineering Simulations (SPACES) were performed using the Vertical Motion Simulator (VMS) at the NASA Ames Research Center in Moffett Field, California, SPACES I in November-December 1985 and SPACES II in April-May 1986.

#### Experiment

A detailed description of the simulation background and scenario is presented in References 141 and 155; the following will only briefly describe the experiments.

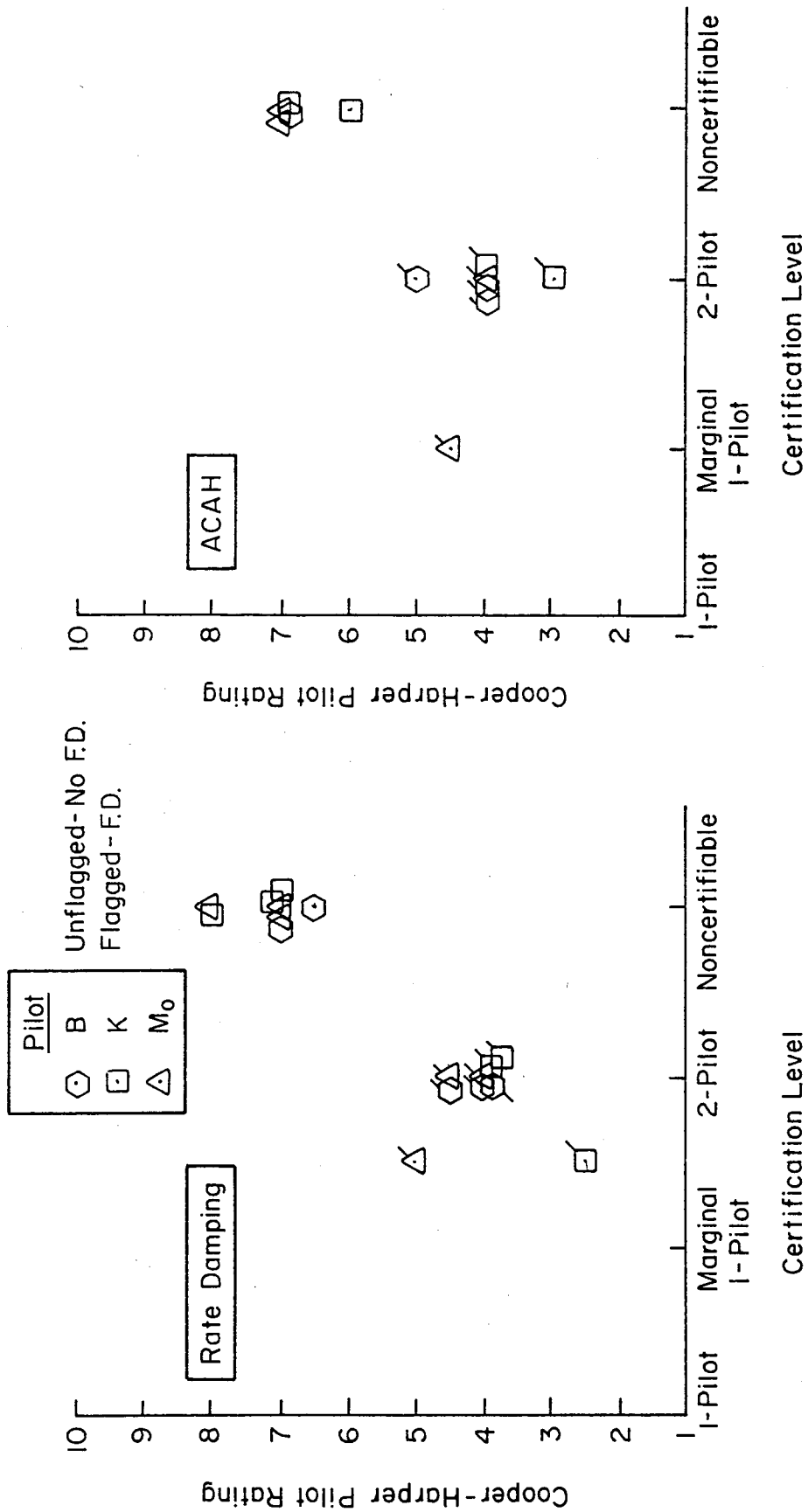


Figure 29(3.2.2). Cooper-Harper Pilot Rating vs. FAA Certification Level Rating for NAE Data of Reference 38

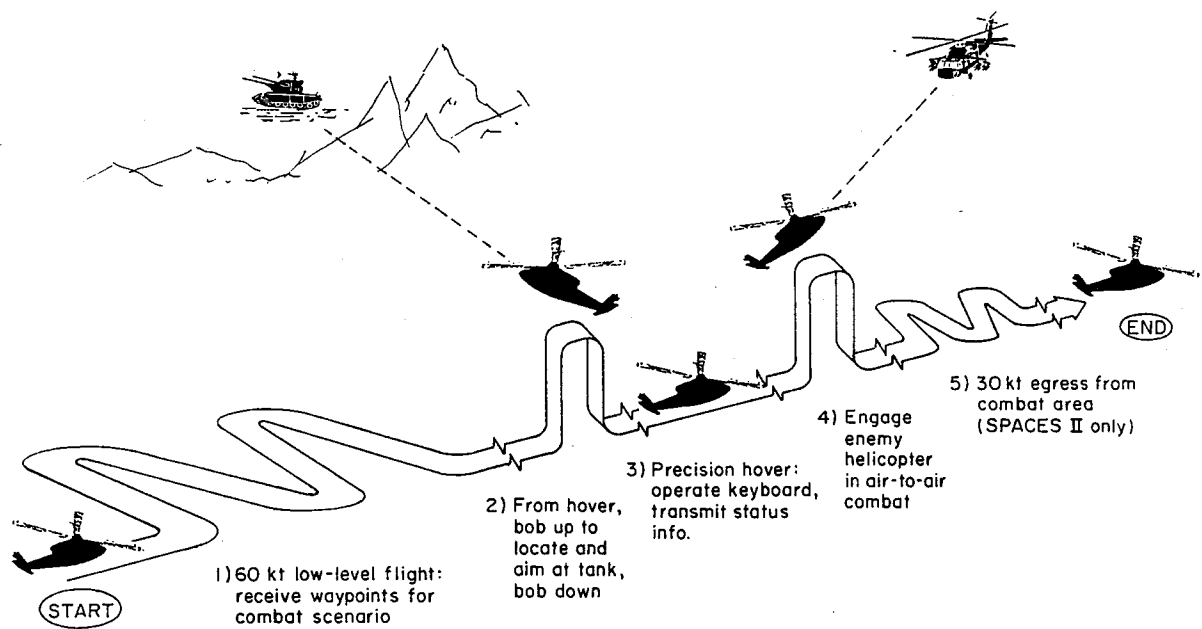


Figure 30(3.2.2). Description of Tasks for Single-Pilot Simulations  
(References 141 and 155)

The tasks are summarized in Figure 30. The experiments were conducted on the NASA Ames VMS 6-degree-of-freedom simulator, with an advanced "glass cockpit" and computer-generated imagery of the Helicopter Air Combat (HAC) data base. The conventional helicopter cockpit displays were replaced by a head-up display (HUD) with analog or digital indications of airspeed, altitude, heading, etc. Head-down displays consisted of a moving map showing position within the data base, and positions of the enemy tank or helicopter. A touchpad screen provided mission and aircraft status information. Cockpit controls consisted of conventional pedals, a left-hand sidestick collective, and a right-hand sidestick cyclic.

Communications were simulated using a single researcher, who followed a script to produce all radio traffic via a voice disguiser. The evaluation pilot was required to communicate with other aircraft and ground controllers, and a realistic level of background communications was maintained. An automated voice system provided indications to the pilot of operation of checklist functions, warnings, and cautions.

The helicopter model was equipped with both missiles and guns for the ground and air attack phases. The computer-controlled enemy tank was capable of firing on the helicopter, and on several occasions successfully shot the helicopter down. Destruction of the helicopter was simulated by drawing cracks on the screen.

Visibility was set at approximately 1-1/4 mile with a 200-ft ceiling to simulate adverse weather and to force the pilot to remain at low level. A 10-kt wind and light-to-moderate level of turbulence were applied to all

evaluations. Five evaluation pilots from the Army Aviation Systems Command (AVSCOM) participated in SPACES II.

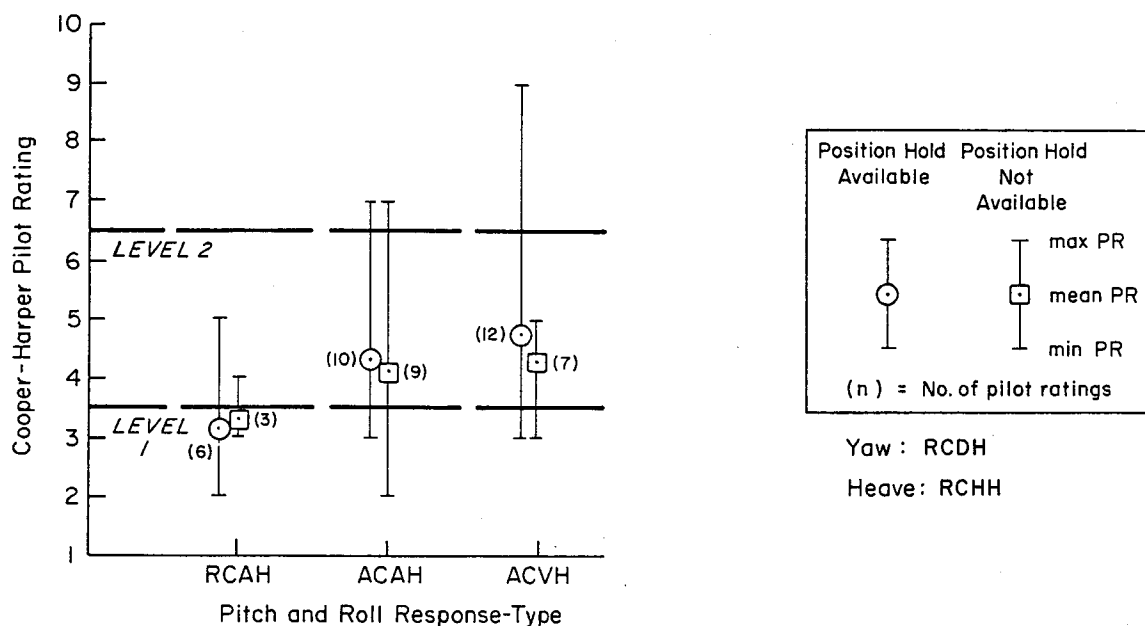
The helicopter mathematical model used for both simulations was the NASA Ames ARMCOF (Reference 145), representing the UH-60 Blackhawk. The control system model developed for the Advanced Digital/Optical Control System (ADOCS) (Reference 14) program served as the baseline system. The ADOCS flight control system uses switching logic to vary Response-Types, and this switching logic was the primary form of Response-Type variation in the first simulation. For the second simulation, several modifications to the basic ADOCS structure were made to produce other Response-Types.

Three primary pitch and roll Response-Types were evaluated: Rate Command/Attitude Hold (RCAH), Attitude Command/Attitude Hold (ACAH), and Attitude Command/Velocity Hold (ACVH). In addition, two "hold" functions -- heading and position -- were evaluated by turning these functions on or off.

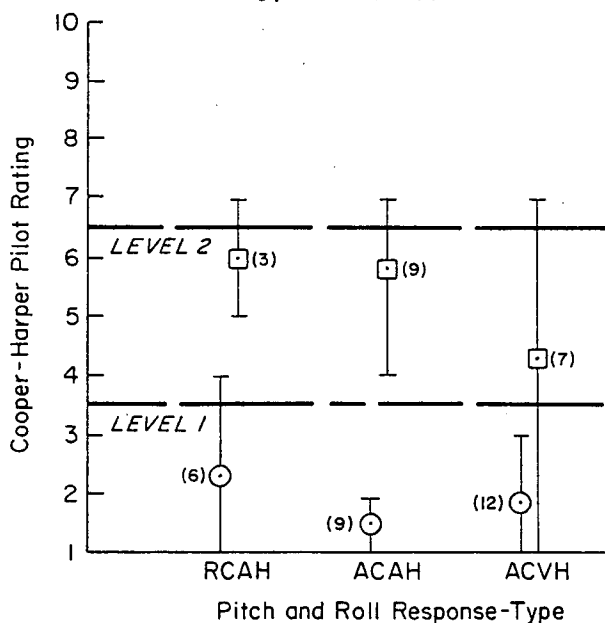
### Summary of Experimental Results

The Cooper-Harper Handling Qualities Ratings (HQRs) for tasks 2,3, and 4 in Figure 30 are summarized in Figure 31. The "ground attack on tank" and "air-to-air combat" tasks have been interpreted as the "target acquisition and tracking" MTE in Table 1(3.2). The similarity of the HQRs between Figures 31a and 31c support the notion that these tasks may be grouped as a single MTE for handling qualities analyses. This is further supported by the pilot commentary which is focused on problems related to centering the pipper on the target in both cases. The "hover and transmit status report" is construed as the "hovering tasks involving divided attention" MTE in Table 1(3.2).

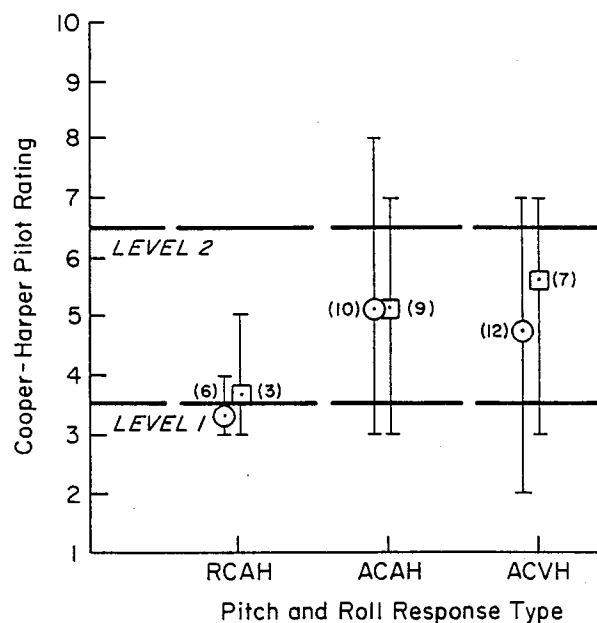
Before making the interpretations that led to the Table 1(3.2) requirement, certain reservations regarding the data should be noted. In all cases, the RCAH Response-Type is predicted to be superior to ACAH or ACVH, a result which is at odds with previous simulator evaluations. For example, in the "ADOCS" simulation (Reference 14), these same configurations were tested with the result that ACAH was the best Response-Type, and RCAH was consistently Level 2. Similar results were obtained in Reference 4, a simulation of V/STOL shipboard landings. A logical explanation would be that the ACAH and ACVH Response-Types lacked sufficient agility for the aggressive MTEs simulated. However,  $q_{pk}/\Delta\theta$  and  $p_{pk}/\Delta\phi$  are within Level 1 limits (as expected since actuator saturation was not simulated, and the bandwidths are on the order of 3.5 rad/sec in pitch and roll). Furthermore, there is no evidence in the pilot commentary that agility, control power, or large control forces were a factor in any of the ratings. In fact, the pilot commentary relating to the degraded configurations was centered almost exclusively on PIO problems in getting, and holding, the pipper on the target. This is surprising since a bandwidth of 3.5 rad/sec is predicted to be Level 1 in Paragraph 3.3.2.1. Simulator visual time delay is not suspected, since RCAH usually suffers the most from such a problem, and the visual delays for the task (pipper on target) probably were not large. The



a) Ground Attack on Tank (target acquisition and tracking, table 1(3.1))



b) Hover and Transmit Status Report (hovering tasks involving divided attention, table 1(3.1))



c) Air-to-Air Combat (one-on-one engagement) (target acquisition and tracking, table 1(3.1))

Figure 31(3.2.2). Summary of Pilot Rating Data -- Single-Pilot Operations; SPACES II Simulation

most likely explanation for this discrepancy is that the sidestick controller characteristics, particularly sensitivity, were not properly optimized for each Response-Type, i.e., it was correct for RCAH, but not for ACAH and ACVH.

The data in Figure 31 support disallowing an upgrade from Rate to increased stabilization, for UCE=1, in Table 1(3.2). However, because of the above factors, and previous experience which indicates that the Response-Type does not effect closed-loop tracking as long as adequate bandwidth, agility, and control power exists, it was decided not to disallow upgrades in the Response-Type, i.e., to ignore the Figure 31 data, at least in that respect.

There is no reason to doubt the validity of the data in Figure 31b which predicts that position hold (PH in Table 1(3.2)) is essential for the "hovering tasks involving divided attention" MTE. Similarly the data in Figure 32 indicate that heading hold is of considerable value for this hovering MTE, and even though the mean rating is on the Level 1 boundary, it seems prudent to require heading hold, i.e., RCDH in Table 1(3.2).

The configurations in Figures 31 and 32 included RCHH, but do not illustrate the effect of removing the height hold function. However, the results of the flight tests conducted by NASA Ames on the variable stability CH-47, reported in Reference 185, indicated that RCHH was the most desirable feature (PH second and RCDH third) for reducing workload. In addition, the pilots commented that tasks involving divided attention require RCHH for operation in near-terrain flight to insure safety. On this basis, it was judged necessary to require RCHH for the "hovering tasks involving divided attention operation" MTE for UCE=1 in Table 1(3.2).

In summary, the impact of the SPACES II ground-based simulation, and the NASA CH-47 in-flight simulation, on Table 1(3.2) is to require Rate + RCDH + RCHH + PH for UCE=1, for the "hovering task requiring divided attention" MTE. This MTE has a significant impact on the flight control system complexity, and is primarily aimed at mission requirements which call for tactical operations with a single pilot. The effect of adding a second pilot is seen to consistently improve the mean HQR to Level 1, regardless of the Response-Type, as shown in Figure 33. The procuring activity should carefully evaluate the tradeoff between FCS complexity and a second pilot. Of course, if the mission and available vision aids call for operations in UCE=2, the complexity of the FCS is defined regardless of the number of pilots.

d. Guidance for Application

None.

e. Related Previous Requirements

None.

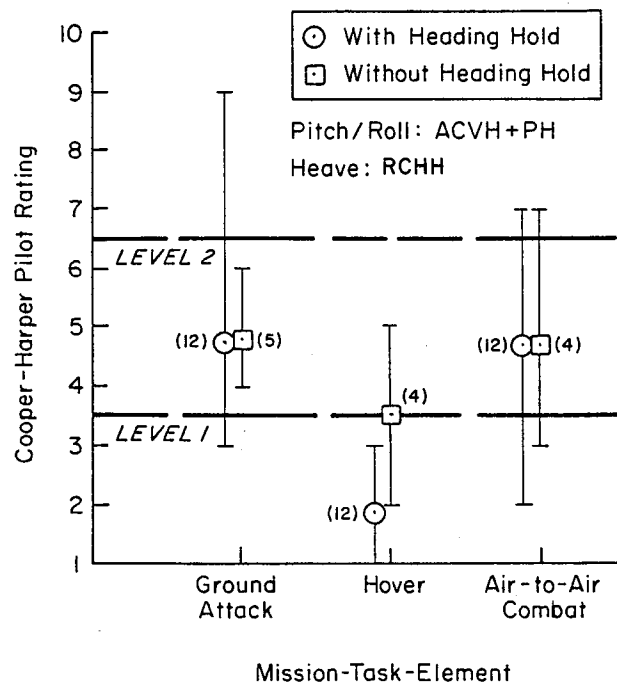


Figure 32(3.2.2). Effect of Heading Hold on Pilot Rating.  
Single-Pilot Operation; SPACES II Simulation

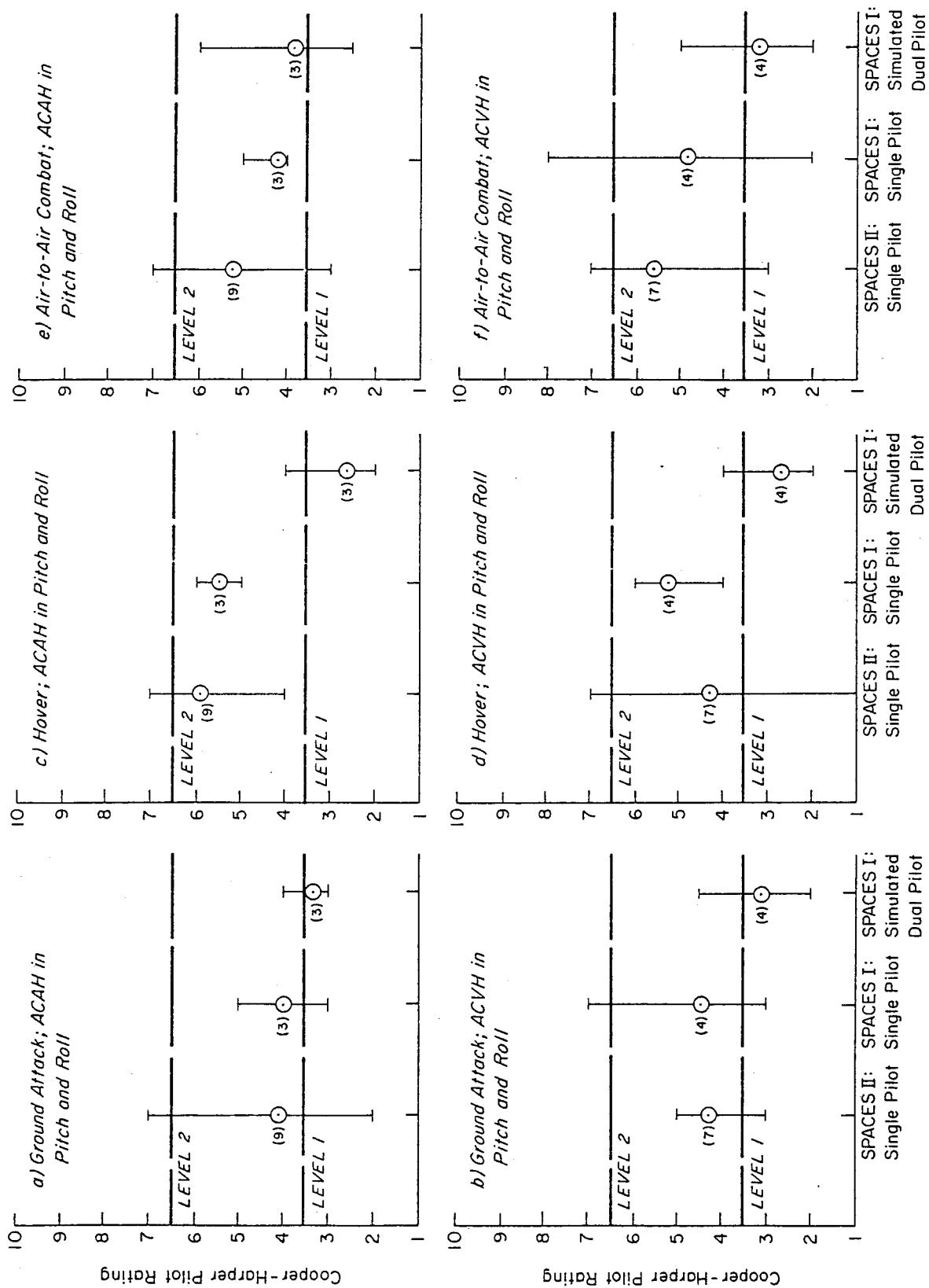


Figure 33(3.2.2). Effect of Crew Size for Hovering and Low-Speed Tasks. Turbulence included in All Runs. Position Hold was not Available



a. Statement of Requirement

3.2.2.1 Determination of the Usable Cue Environment. The displays and vision aids provided to the pilot shall be assessed to determine their effectiveness for stabilization and control. The visual cue ratings shall be determined using all displays and/or vision aids that are expected to be operationally available to the pilot, in the Degraded Visual Environments specified in Paragraph 3.1.1. The usable cue environment (UCE) is defined in Figure 2(3.2) using the visual cue ratings obtained from the Figure 1(3.2) scale during the flight assessments specified below. Points falling on a boundary in Figure 2(3.2) will be considered to lie in the region of numerically higher UCE.

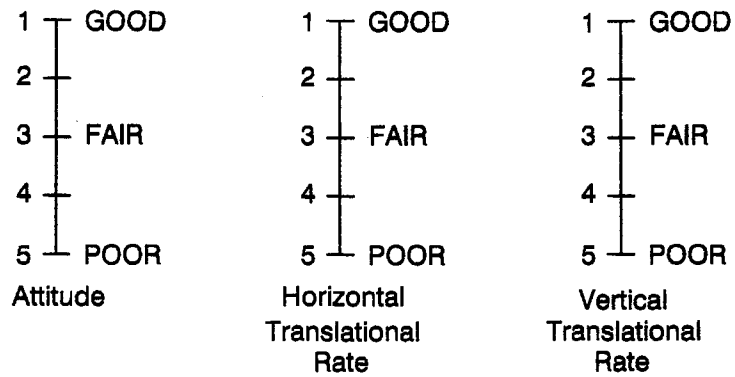
The translational rate visual cue rating to be applied to Figure 2(3.2) is the poorer (higher numerically) of the horizontal and vertical axis ratings obtained from Figure 1(3.2). The visual cue ratings (VCRs) are to be made by at least 3 pilots and the UCE shall be obtained by using the mean VCRs in Figure 2(3.2). The test rotorcraft must meet the requirements for a Rate Response-Type as defined in Paragraph 3.2.5 and have a Level 1 mean pilot rating (Figure 1(2.8) scale) by at least 3 pilots operating without any vision aids in good visual conditions (UCE=1) and negligible turbulence. The following Mission-Task-Elements shall be flown when making the UCE assessments: hover, vertical landing, pirouette, acceleration and deceleration, sidestep, bob-up and bob-down. The task descriptions and performance limits specified in Sections 4.4 and 4.5 for each of these maneuvers shall apply when making the VCR ratings except that the maneuvers may be flown in calm winds.

If the standard deviation of the visual cue ratings among the pilots is greater than 0.75, either additional pilots shall be employed, or the procuring activity shall assign a level of UCE.

b. Rationale for Requirement

Many helicopter missions will include operations at night, and in poor weather, in the nap-of-the-earth (NOE) environment. This gives rise to two critical issues, 1) collision avoidance with fixed or moving objects, and 2) control and stabilization. The Usable Cue Environment (UCE) has been included as a metric to quantify the latter issue, so that adequate stabilization can be specified to compensate for the missing cues. Such additional stabilization is quantified in terms of an upgraded "Response-Type," and is specified in Table 1(3.2). This paragraph provides specific guidance relative to the procedures to be followed to determine the UCE values to be used in Table 1(3.2).

The specification methodology is based on the hypothesis that degraded visual cueing can be compensated for with increased stabilization. That hypothesis was the subject of several inflight experiments, one using a conventional helicopter, and two with the Canadian National Research Council's variable stability Bell 205A. The visual environment was varied by flying over varying terrain, and by the use of electronically fogged lenses, and night vision goggles with daylight training



### DEFINITIONS OF CUES

X = Pitch or roll attitude and lateral, longitudinal, or vertical translational rate.

**Good X Cues:** Can make aggressive and precise X corrections with confidence and precision is good.

**Fair X Cues:** Can make limited X corrections with confidence and precision is only fair.

**Poor X Cues:** Only small and gentle corrections in X are possible, and consistent precision is not attainable.

Figure 1(3.2). Visual Cue Rating (VCR) Scale to be Used When Making UCE Determinations

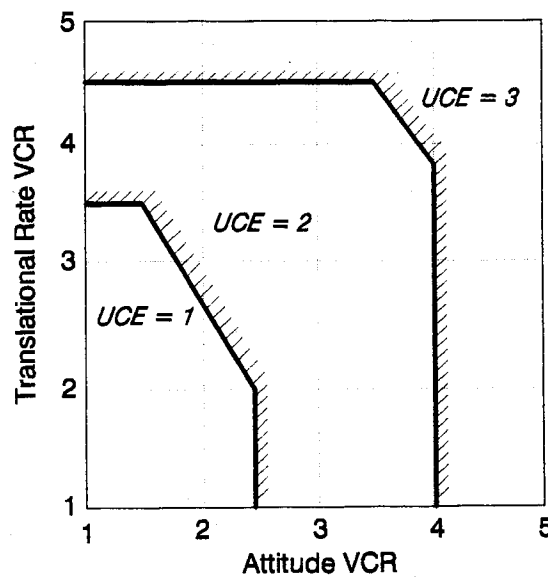


Figure 2(3.2) Definition of Usable Cue Environments

filters. Two of these experiments provided the data that was used to develop the UCE boundaries (Figure 2(3.2)) and supporting VCR scale (Figure 1(3.2)). A complete reporting of these experiments is given in References 156 and 187. Those results are summarized in the supporting data below, with some modifications to the interpretations of the data. The third flight test was specifically oriented toward the determination of the required Response-Types as a function of UCE for use in Table 1(3.2), and those results are discussed in the supporting data for Paragraph 3.2.2.

The visual cue rating scale was developed during the tests described above, as well as during moving base simulations on the NASA Ames Vertical Motion Simulator. In fact, the initial motivation to investigate pilot cues for low speed and hover arose from an observation that it was much more difficult to hover a rate augmented VSTOL or helicopter on a ground based simulator than in the real world (e.g., see Reference 5). A brief flight experiment using an OH-6 and fogged goggles (Reference 6) supported these observations.

During the conduct of the above experiments, it was noted that the pilots' perception of the quality of the available cues when not flying was not reliable. That is, what looked to be an adequate environment before liftoff frequently was not. Therefore, a scale is needed that directly measures the pilot's ability to actually fly in a precise and aggressive manner. Precision and aggressiveness were chosen as metrics because it was found that in poor visual cue environments, pilots tend to be very gentle (nonaggressive) on the controls to avoid losing control, and, as a consequence of such gentle control usage, good precision must be sacrificed. The adjectives "good," "fair," and "poor" were chosen based on the results of Reference 56, which showed that the semantic meanings of these words tend to fall on a linear scale.

The specification does not assume that increased stabilization is a substitute for improved displays, but rather that it is a way to make up for some of the deficiencies in existing displays. The objective of such displays and vision aids is to allow the pilot to see through obscurations and darkness, and as a result, the tradeoffs are generally in favor of distant acuity and field-of view at the expense of fine-grained texture (microtexture). The flight test results of Reference 156 indicate that microtexture is a necessary cue for the pilot to stabilize the helicopter in the low-speed and hover flight regime. The visual cue rating scale in Figure 1(3.2) is intended to provide a measure of the ability of a display or vision aid (or a simulator visual scene) to reproduce the critical cues required for the pilot to stabilize the helicopter. As such, it provides a metric to determine the degree of required stability augmentation over the baseline Rate Response-Type, as well as to evaluate competing displays and vision aids. The UCE boundaries in Figure 2(3.2) represent experimentally derived combinations of VCRs that result in a corresponding Level of flying qualities with a Rate Response-Type (e.g., UCE=2 implies Level 2 for a Rate Response-Type). It should particularly be noticed that the maneuvers to be used in determining UCE are those for Degraded Visual Environment (DVE) (4.4 and 4.5). These have slightly reduced maneuvering performance compared to the maneuvers in good visual

environment, 4.1 and 4.2. The rationale for this is that it will allow a Rate Response-Type with slightly reduced performance in DVE to be accommodated, before imposing the additional complexity of ACAH, etc. The data supporting these boundaries are discussed below.

c. Supporting Data

The supporting data for the Figure 2(3.2) UCE boundaries is based on testing helicopters with a Rate Response-Type (most existing helicopters would be classified as a Rate Response-Type) in varying visual conditions. These variations were obtained by conducting the experiments over test courses with varying amounts of macrotexture (large objects), and by varying the visible microtexture via electronically fogged lenses or night vision goggles with daylight training filters. The pilots were required to give VCR ratings using the Figure 1(3.2) scale, and Cooper Harper handling qualities rating (HQRs). The tasks consisted of precision hovers, precision vertical landings, a sidestep left and right, a dash-quickstop, and a bob-up/-down. These maneuvers were arranged in a test course which was completed three times, after which the pilots assigned a set of VCR ratings and an HQR for each task. The experiments and the resulting data are presented in detail in References 156 and 187 (both contain the same information), and are summarized in Figure 1. A relationship between HQR (for the hover and landing task) and VCR was obtained by performing a linear multiple regression fit on the Rate Response-Type data with the result, shown in Figure 1. The estimated ratings from this fit are plotted vs. the actual ratings in Figure 2 where it is seen that the data spread about the line of perfect correlation is reasonable up to HQRs of about 7. Beyond this value, the linear fit is nonconservative. However, the complexity of a multiple nonlinear regression seems unwarranted, since only the data up to a rating of 6.5 is used in the criterion development. The boundaries in Figure 1 represent a reasonable separation between regions of pilot ratings of 1-3 (Level 1) and 4-5. Ratings of 6 and greater are defined as UCE=3. These have been slightly modified to develop the final criterion boundaries as shown by the solid lines with hash marks in Figure 1. It was felt that the UCE=1 region should not terminate at a point, and that the upper limits (defined by  $VCR_x^* = 4.5$  and  $VCR_\theta = 4.0$  in Figure 1) are a logical extension of the existing data. Without such an interpretation, the dashed linear regression line would indicate that poor attitude cues ( $VCR_\theta = 5$ ) could be compensated with good translation cues ( $VCR_x^* = 1.8$ ), which does not seem likely. Finally, the 6.5 boundary was moved slightly to the left of the HQR=6.5 regression line to provide some protection from a region of very degraded handling qualities.

The required stabilization (Response-Types) to obtain Levels 1 and 2 for each region of UCE defined in Figure 1, was derived from the results of a third variable stability flight test and a ground-based simulation, described in the supporting data for Paragraph 3.2.2 (e.g., see Figures 9(3.2.2) and 13(3.2.2)).



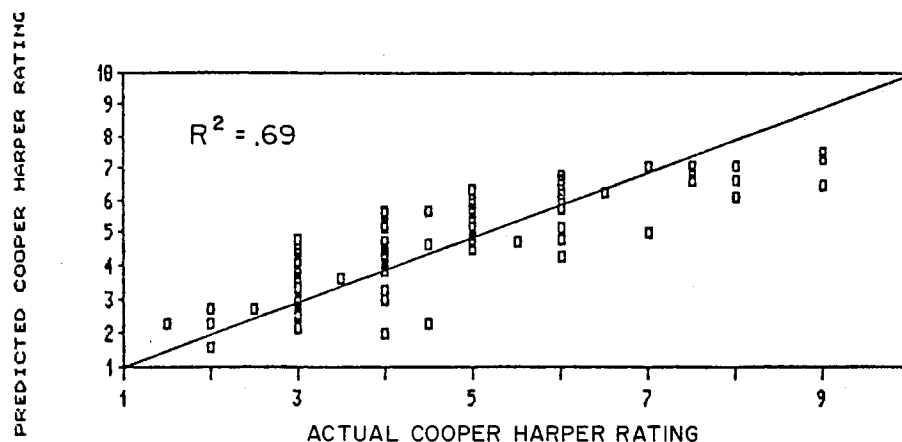


Figure 2(3.2.2.1). Linear Multiple Regression for all Cases Without ACAH (from Both Experiments)  $R^2 = 0.69$ , 89 Observations, RMS Error of Cooper-Harper Rating = 1.0

d. Guidance for Application

The VCR and UCE values must be obtained for each proposed display and/or vision aid in environments expected for the required operational missions as defined by the "Degraded Visual Environments" (DVE) specified in Paragraph 3.1.1. For example, if the mission calls for NOE operations at night, and in poor weather, some type of display/vision aid will be required. The UCE for the proposed display/vision aid should be tested in the worst case conditions, which are defined by environments with the least microtexture. For a FLIR this could consist of flight over a cold-soaked grass field, whereas for night vision goggles, very low illumination over a similar field may represent the worst case. Experience has shown that low speed and hover over water can produce a poor UCE, even in broad daylight VMC conditions. A representative DVE for the LHX helicopter is given in the discussion of Paragraph 3.1.1.

The test site should include prominent objects (macrotexture, such as tires on the ground) so that the pilot has a reference upon which to evaluate his or her precision. Pilots sometimes complain that they cannot judge their precision because of the poor visual environment. However, as long as it is possible to be precise within the context of the perceived visual field, the ability to stabilize and control is probably not compromised. That is, if the pilot can keep the hover reference point exactly where he wants it, control is not an issue, even though the exact location may be off. Since the macrotexture is, by definition, always visible (Paragraph 3.2.4), a capability to see objects in the visual field is theoretically assured.

It is not necessary that the display or vision aid be tested in the proposed new rotorcraft; a surrogate helicopter is an acceptable test bed, as long as the Response-Type is Rate, and it is Level 1 in conditions of UCE=1. A significant difference between the fields of views of the two aircraft would be cause for some concern but would not necessarily be a limiting factor. This is shown in Figures 3 and 4, taken from Reference 156.

The VCR scale is intended to be linear, and the pilot subjects should be encouraged to mark a point on each scale, as opposed to giving numerical ratings. These marked points may be converted to VCR ratings by "eyeball interpolation" after each flight. An average of VCR rating which falls on a UCE boundary should be classified as the highest UCE.

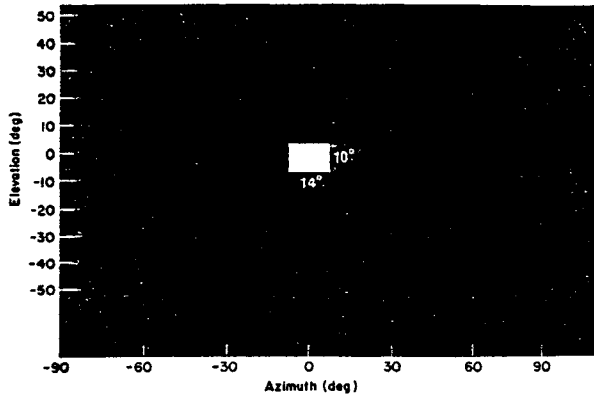
It is desirable to obtain a separate VCR evaluation for each of the required maneuvers. The UCE should be based on the worst VCR rating set across all tasks.

Experience with using the VCR scale has shown that the rating variability is usually less than 0.75. For example, References 156 and 187 show that the standard deviation was less than 0.75 for over 80% of 200 separate evaluations. However, occasionally a pilot subject who is not experienced with subjective rating scales, or one with strong opinions about the display or vision aid, will produce ratings that are "out of line." This can be minimized by emphasizing the importance of using the adjectives on the scale, and slavish adherence to ratings based on actual aggressiveness and perceived precision, as opposed to an evaluation of the perceived cues. That is, the pilot should be asked to rate his ability to perform the tasks with aggressiveness and precision, in the same way he evaluates his ability to perform a defined task when assigning a Cooper-Harper Handling Qualities Rating. He should not try to assess the goodness of the visual cues by combining his ability to achieve a desired level of aggressiveness and precision with some preconceived notion of how well he would do with the same configuration in good visibility.

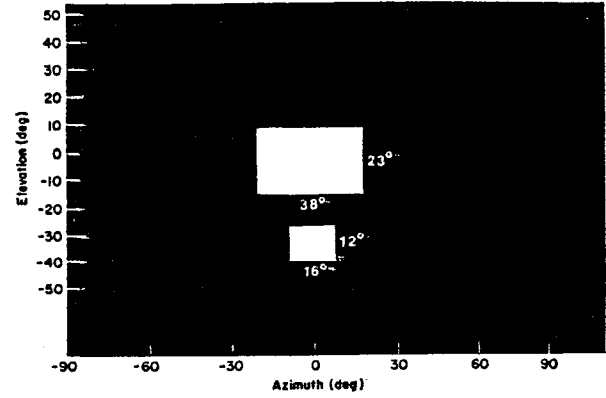
In some cases, it may be necessary to remove one pilot's ratings from consideration, and this is allowed for in the requirement. However, it is important that an adequate number of subjects are in agreement (at least 3), before making such a decision.

e. Related Previous Requirement

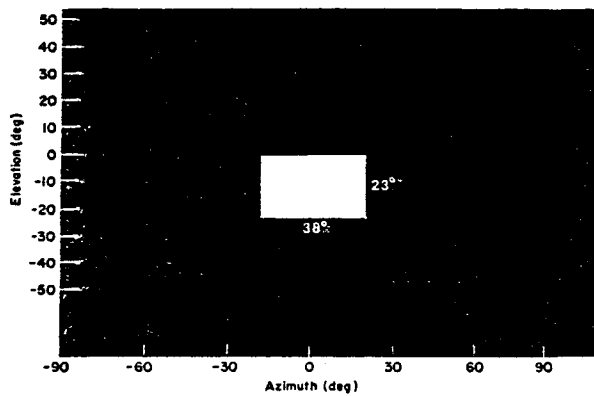
None.



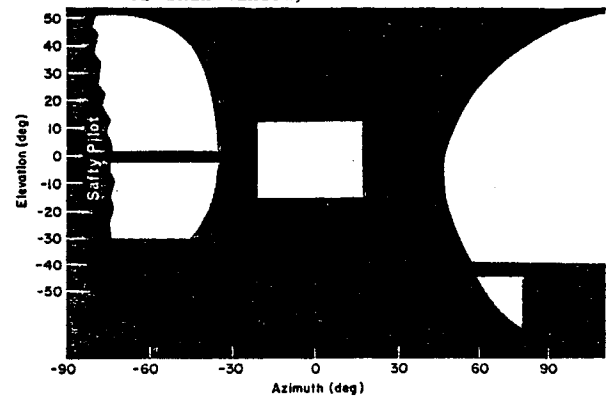
a) Variations in field-of-view -- configuration 1 (narrow upper front to evaluate potential displays)



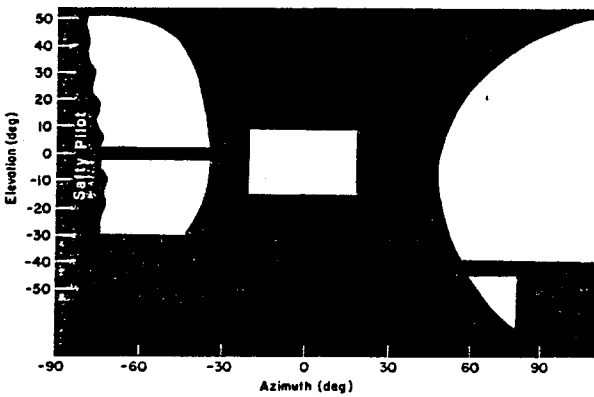
d) Variations in field-of-view -- configuration 3 (nominal upper front plus lower front to investigate effect of thin window)



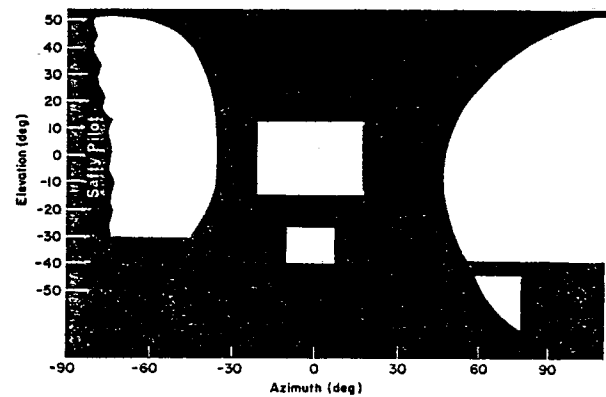
b) Variations in field-of-view -- configuration 2 (nominal upper front to simulate NASA Ames VMS front monitor)



e) Variations in field-of-view -- configuration 7 (wide upper front plus sides)



c) Variations in field-of-view -- configuration 6 (nominal upper front plus sides to investigate effect of thin window)



f) Variations in field-of-view -- configuration 8 (wide upper front + lower front + sides)

Figure 3(3.2.2.1). Field-of-View Variations (from Reference 156)



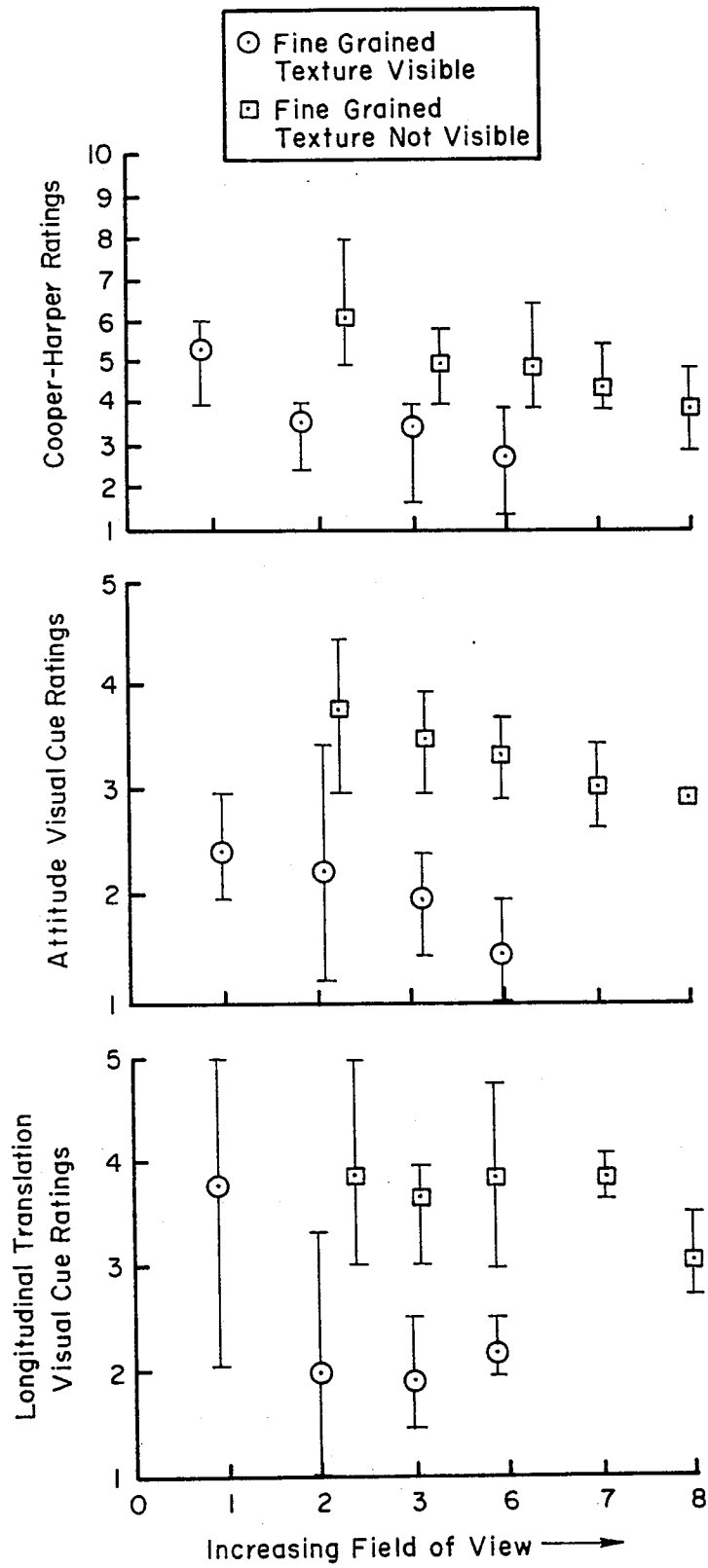


Figure 4(3.2.2.1). Effect of Field-of-View and Microtexture Variations (from Reference 156)

a. Statement of Requirement

3.2.3 Combinations of Degraded Response-Type and UCE. Table 1(3.2) allows a reduction in Response-Type for Level 2 for some selected cases. If the dynamics of the particular Response-Type are Level 2 or 3 based on one or more of the requirements of Section 3.3, the combination of degraded Response-Type, dynamics, and UCE may result in flying qualities Levels that are worse than for any single degradation alone. Table 3(3.2) defines the Levels corresponding to these combinations.

TABLE 3(3.2). LEVELS FOR COMBINATIONS OF DEGRADED RESPONSE-TYPE, DYNAMICS, AND UCE\*

UCE	LEVEL FOR RESPONSE-TYPE (TABLE 1(3.2))	LEVEL FOR DYNAMICS (SECTION 3.3)	OVERALL LEVEL
2	1	2	2
2	2	1	2
2	2	2	3
3	1	2	3
3	2	1	2
3	2	2	3

b. Rationale for Requirement

This requirement is included to insure that the specification minimizes the probability of unsafe flying qualities in conditions of poor visual cueing following a failure of one or more elements of the flight control system. Consider the following scenario. A failure in the flight control system occurs while operating NOE at night, which results in a transition from ACAH to a Rate Response-Type. The data presented as support for Paragraph 3.2.2 indicate that a Rate Response-Type will be adequate for Level 2 provided that the dynamic requirements of Paragraph 3.3 are met. If, on the other hand, the failure is such that the backup flight control system has Level 2 dynamics, it is likely that the combination of reduced stabilization and degraded dynamics will be Level 3.

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\*This table differs from the current version of the specification (ADS 33C) in that it includes a row for UCE=3 with a Level 1 Response-Type (fourth row of the table). This row was removed when it was determined that Level 1 was not possible for UCE=3. However, later ground-based simulation has shown that a TRC Response-Type yields Level 1 in UCE=3. This version of the Table is therefore more correct and the specification will be accordingly revised at the next update.

c. Supporting Data

The data and rationale behind the values used in Table 3(3.2) are summarized below.

- For UCE=2, a degradation in Response-Type or dynamics to Level 2 is considered to result in overall Level 2 flying qualities. (Rows 1 and 2 of Table 3(3.2)).
- For UCE=2, a degradation in Response-Type and dynamics to Level 2 is considered to result in overall Level 3 flying qualities. (Row 3 of Table 3(3.2)).
- For UCE=3, a degradation in dynamics to Level 2 is considered to result in Level 3 flying qualities. The rationale for this is based on the UCE=3 data in Figure 9(3.2.2), which shows that the flying qualities are solid Level 2 for maneuvering tasks without a failure. This, combined with the lack of visual cues needed to stabilize degraded dynamics (microtexture), is judged to be Level 3. (Row 4 of Table 3(3.2)).
- For UCE=3, a degradation in Response-Type to a backup flight control system with Level 1 dynamics is considered to be Level 2. This is based on the data in Figure 9(3.2.2) which shows that a Rate Response-Type is Level 2 (albeit marginal) in UCE=3. (Row 5 of Table 3(3.2)).
- For UCE=3, a degradation in Response-Type and in dynamics to Level 2 is considered to be Level 3. However, it would not be surprising if such a combination would result in loss of control. Needless to say, the probability of such an occurrence should be very low. (Row 6 of Table 3(3.2)).

d. Guidance for Application

The probability of occurrence of a failure in elevated UCE is significantly lower than those calculated based on a total flight hour basis. It will therefore be necessary to estimate the percent of time that the proposed rotorcraft will spend in UCE=2 and UCE=3.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.2.4 Rotorcraft Guidance. For near-earth operations at night and in poor weather, sufficient visual cues shall be provided to allow the pilot to navigate over the terrain, and to maneuver the rotorcraft to avoid obstacles while accomplishing the Mission-Task-Elements. The contractor shall make calculations which determine three-dimensional maneuvering envelopes defined by the required Mission-Task-Elements (Paragraph 3.1.1), and indicate on these envelopes the maneuvering limits imposed by the visual field of the available displays and/or vision aids. The detailed assumptions regarding limiting performance (pilot delays, reaction time, and rotorcraft limits) must be approved by the procuring activity.

b. Discussion

The usable cue environment (UCE) methodology is provided to ensure adequate handling qualities in the presence of severely degraded visibility. Specifically, the degradations considered in the UCE methodology involve cues required by the pilot to stabilize the rotorcraft in attitude and/or linear translation. The ability to maneuver without colliding with fixed objects represents another dimension to the problem which must be addressed in order to successfully accomplish the Mission-Task-Elements. A UCE of 1 does not guarantee that the pilot can see well enough to avoid obstacles. While the handling qualities specification is primarily aimed at ensuring satisfactory aircraft control, it is inappropriate to ignore the closely related navigation function required to accomplish the task.

There is considerable evidence that the fine-grained texture in the foreground is a primary cue for stabilization of attitude and path in the low speed and hover flight regime. The loss of this cue sometimes represents extremely low visibility (zero/zero) to the point where only very slow linear translations would be attempted. With the incorporation of displays and/or vision aids, it is possible that the macrotecture (large objects) would be restored and the microtexture (blades of grass, etc.) would not. This loss of microtexture has been noted with the use of night vision goggles, and to an even greater extent with forward looking infrared displays (FLIR) when the temperature gradient is small or zero. In such cases Paragraph 3.3.2 ensures that a higher level of augmentation is required for compliance. The purpose of Paragraph 3.2.4 is to provide a separate metric to set a minimum standard on the effectiveness of the display for seeing and avoiding fixed objects at speeds required to perform the Mission-Task-Elements.

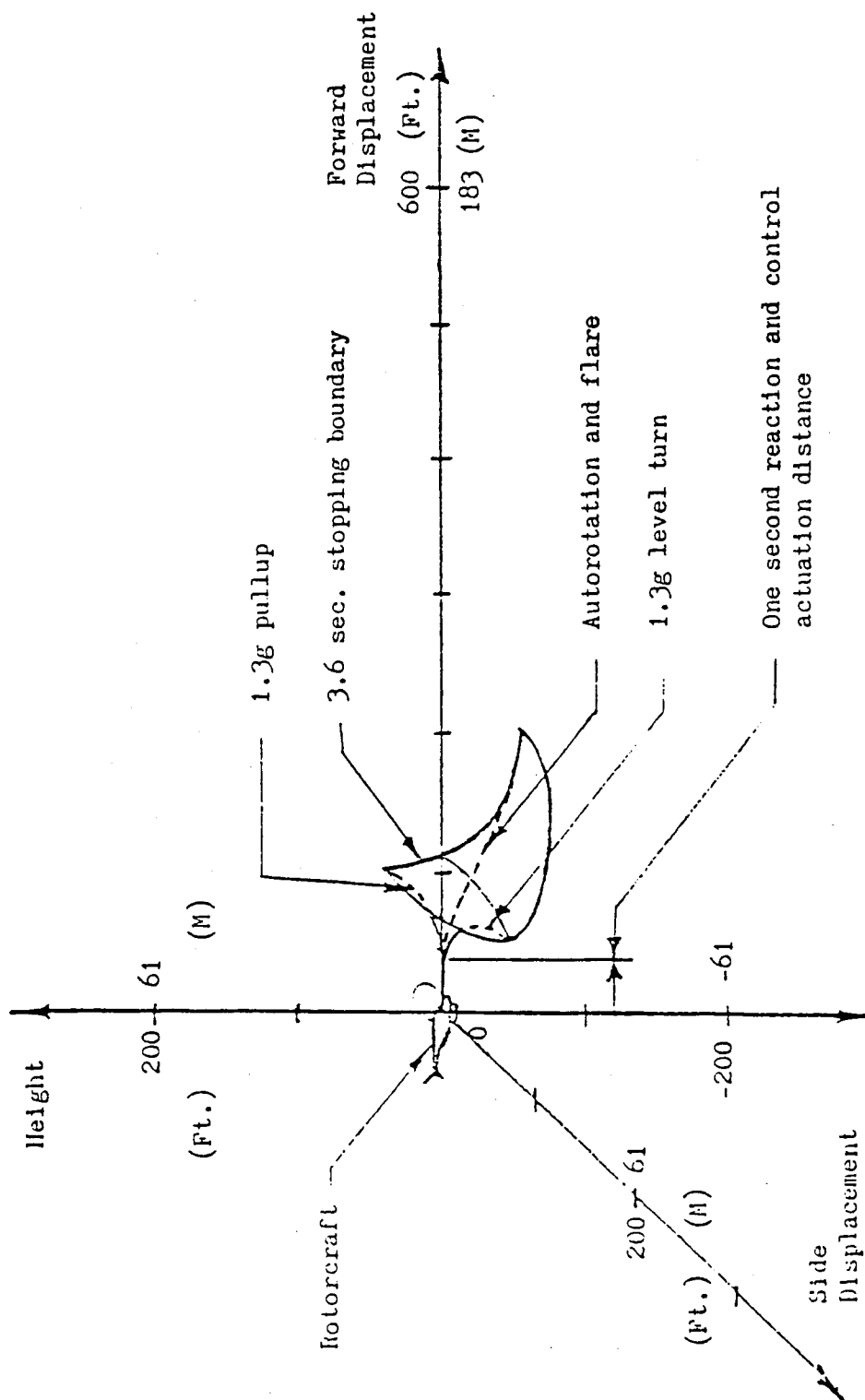
Insurance against collisions with fixed or moving objects while maneuvering in the NOE environment requires that the display contain information regarding the visual field which represents the performance potential of the rotorcraft. Three-dimensional boundaries representing a typical helicopter were developed in Reference 104 in terms of speed, deceleration capability, and ability to turn the flight path vector in the vertical and lateral planes (e.g., normal acceleration and turn radius). These boundaries are shown for three speeds in Figure 1. The

boundaries are fixed to the helicopter in a non-rolling reference frame, and project mostly forward in the direction of motion. The kinematic assumptions used in the calculations are noted in Figure 1. A pilot reaction time of one second was assumed.

The significance of the boundaries in Figure 1 is that the space within them must be visible to the pilot to eliminate the possibility of a collision. As would be expected, the boundaries grow rapidly as speed is increased. The boundaries are strongly dependent on the aircraft performance limits, as well as the limit of the ability of the pilot to maneuver aggressively in the NOE environment. However, for the purpose of defining the required display capability in the specification, it is overly stringent to require that the envelopes include more area than is necessary to accomplish the Mission-Task-Elements. Therefore, the requirement specifies that the boundaries are to be based on the Mission-Task-Elements where possible, and otherwise on helicopter limits.

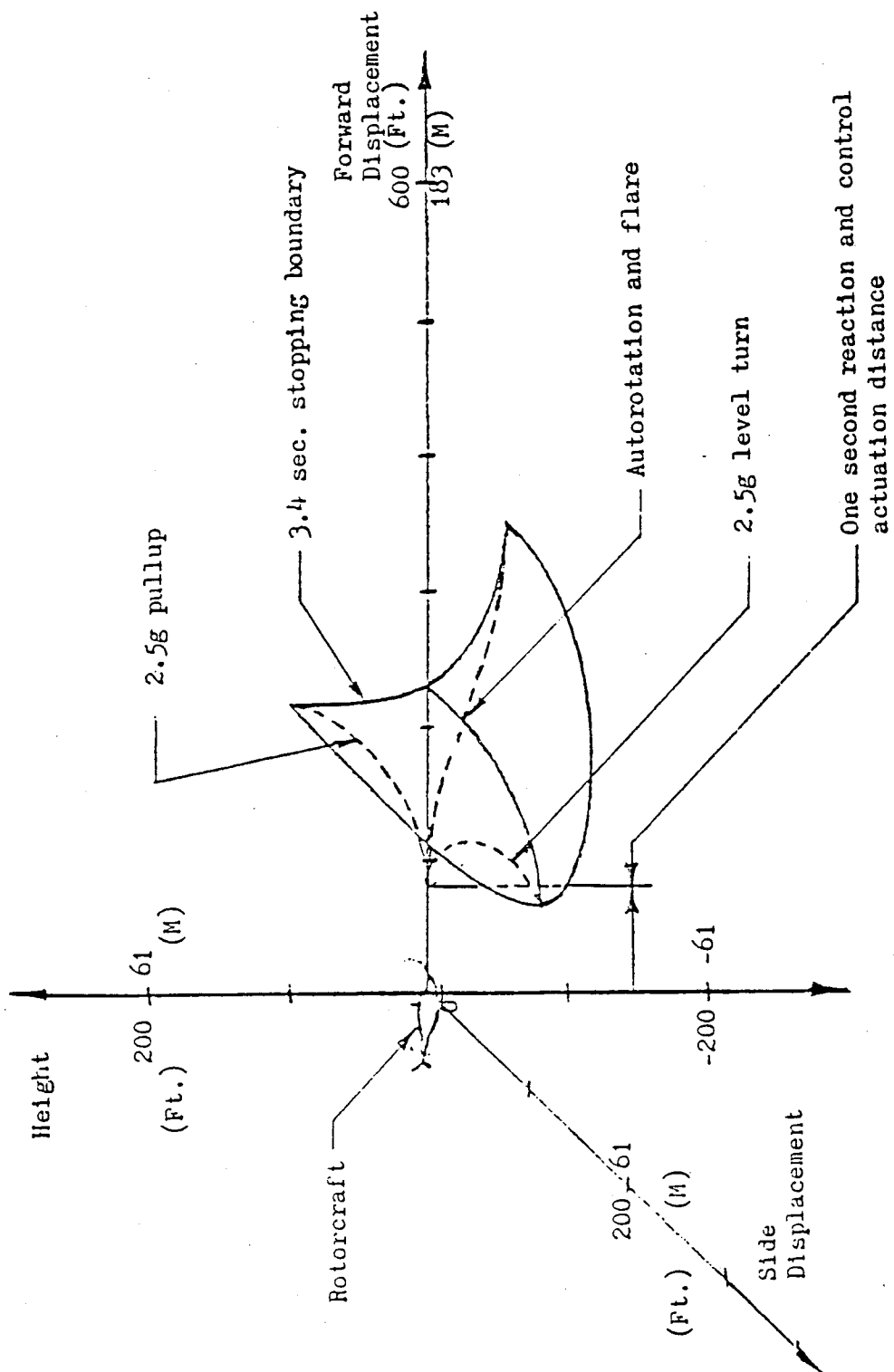
Specific limits are not included in the specification. Instead, the contractor is required to develop boundaries such as those shown in Figure 1, and to compare those boundaries with the capabilities of the available displays. If the information on such displays is deficient in terms of stabilization (e.g., UCE of 2 or 3), the rotorcraft performance used to compute the Figure 1 boundaries must include the effects of the required augmentation [upgraded Response-Type from Table 1(3.2)].

The boundaries constructed for compliance with the paragraph may, as a practical matter, take into account the limitations of existing displays and vision aids. For example, it may not be practical to require the same level of aggressiveness for day VMC as for NOE operations at night and in poor weather, and hence, the definitions of some MTEs may be modified for a degraded usable cue environment (for example, see Paragraphs 4.4 and 4.5).



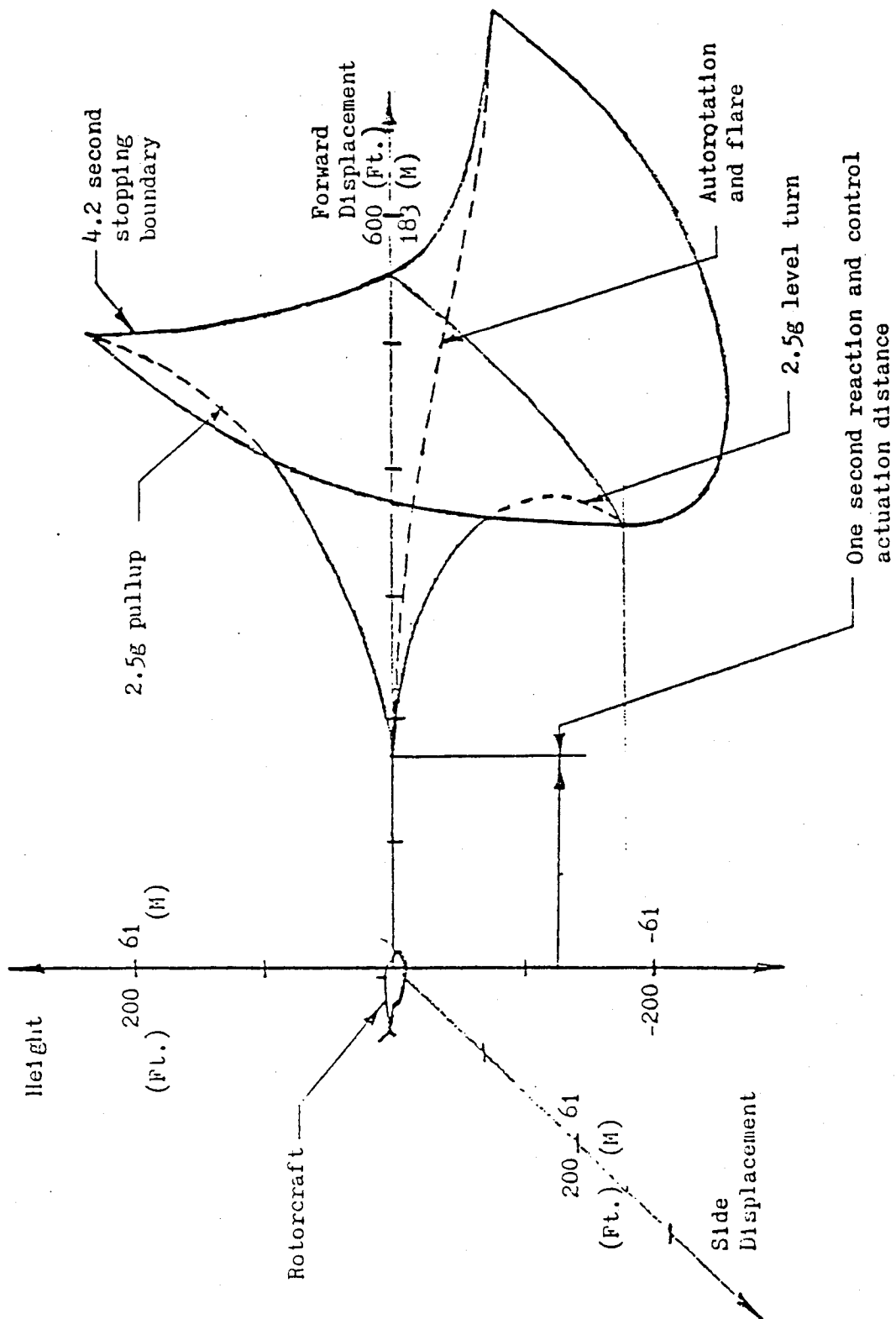
*a) Airspeed of 25 kts*

Figure 1(3.2.4). Typical Rotorcraft Performance Boundaries  
(Right side only) From Reference 104



*b) Airspeed of 50 kts*

Figure 1(3.2.4). (Continued)



*c) Airspeed of 100 kts*

Figure 1(3.2.4). (Concluded)



a. Statement of Requirement

3.2.5 Character of Rate Response-Types. A response which meets the Bandwidth requirements of Sections 3.3 and 3.4 (Paragraphs 3.3.2.1, 3.3.5.1, 3.4.1.1, 3.4.5.1, and 3.4.7.1), may be classified as a Rate Response-Type. No requirement on the specific shape of the response to control inputs is specified, except that the initial and final cockpit control force required to change from one steady attitude to another shall not be of opposite sign.

b. Rationale for Requirement

The Rate Response-Type represents the lowest level of augmentation defined in the specification. Its dominant feature, in terms of flying qualities, is that the frequency response is K/s in the region of piloted crossover (see Reference 147). That is, the slope of the Bode plot of  $q/\delta$  and  $q/F$  is approximately -20 dB/decade over a range of frequencies important for piloted closed loop control, say between 1 rad/sec and  $\omega_{BW}$ . The name "Rate" Response-Type is based on this K/s characteristic. It should be noted that an approximate K/s response is simply another way of saying that a system has reasonable phase margin, and therefore can be controlled without significant equalization in the frequency range of interest. Hence, the Rate Response-Type is the minimum system that can be controlled without undue pilot equalization, i.e., with Cooper-Harper Handling Quality Ratings (HQRs) between 1 and 3.

A more subtle, and yet extremely important, characteristic of a Rate Response-Type is that it requires the pilot to close an attitude loop to stabilize in hover\*. This is illustrated analytically in Paragraph 3.2.7, and supported with pilot rating data in Paragraph 3.2.2, which defines the requirements for Response-Types as a function of Mission-Task-Element, Usable Cue Environment, and division of attention. Pilot ratings and commentary indicate that the requirement to close an attitude loop to achieve a stable hover is not a problem in conditions of good visual cueing, but results in a significant increase in workload when cueing is degraded (discussed in detail in Paragraphs 3.2.2 and 3.2.2.1). Hence, Rate Response-Types are specified for conditions of good visual cueing (UCE=1), and higher levels of augmentation are specified when the visual cueing is degraded (UCE=2 or 3), see Table 1(3.2). Higher levels of augmentation are also found to be necessary for conditions of divided attention, e.g., where the demands of the mission tasks do not allow the pilot to devote a

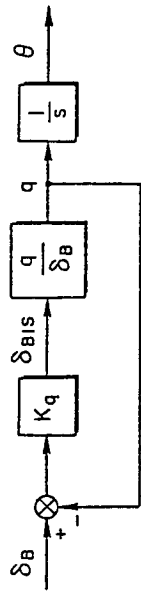
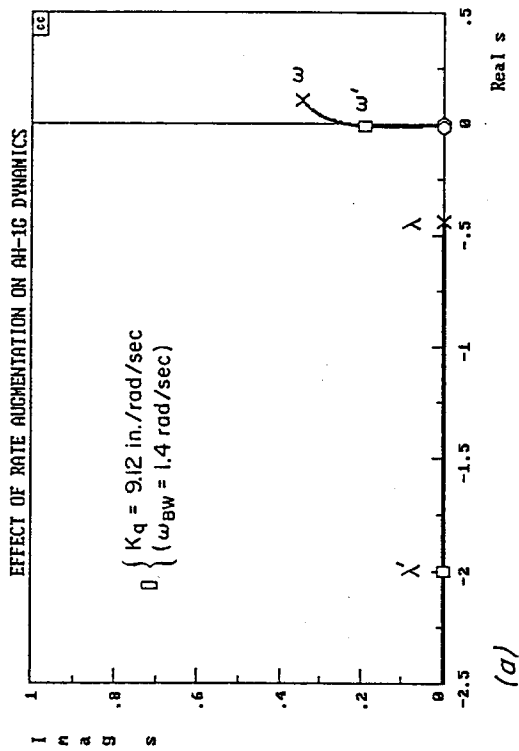
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\*The requirement for closing an attitude loop is based on closed loop analysis and pilot commentary. However, it is possible that the pilot actually uses linear acceleration, or a combination of linear acceleration and attitude. Fortunately, these details do not matter, a point that is further discussed in Paragraph 3.2.7. For the sake of simplicity and physical insight, the inner loop stabilization is generally considered to be an attitude loop in this discussion.

high level of concentration on the closure of the attitude loop at moderate frequencies (1 to 3 rad/ sec).

The Rate Response-Type category is intended to include rotorcraft which are either unaugmented or rate damped. The rate damping may consist of a low-gain, low-authority angular-rate feedback, or a high-gain, full-authority RCAH augments. The following examples illustrate stability augmentation systems that would be classified as Rate Response-Types.

- Example 1. Pure Rate Augmentation. The generic characteristics of pure rate augmentation are illustrated by investigating the effects of pitch rate feedback on the dynamics of an AH-1G and an OH-6A in the hovering flight condition, as shown in Figure 1. The basic AH-1G and OH-6A dynamics in hover were taken from Reference 28. The primary characteristics of interest are as follows.
  - The region of K/s (Bode amplitude slope of -20 dB/decade) is increased by decreasing  $\omega$  and increasing  $\lambda$ . If the Bandwidth is very low (i.e.,  $\lambda'$  is small), the region of K/s could become nonexistent. The lower limits of Band-width in Sections 3.3 and 3.4 prevent this for Level 1.
  - The pitch attitude time response to a step controller input exhibits a low frequency oscillation which is less pronounced with increasing feedback gain (because  $\omega'$  is decreased, and  $\zeta'$  is increased). The lightly damped oscillation persists for all practical values of feedback gain,  $K_q$ , and in the short term appears as a "droop" since the attitude nods back towards zero instead of monotonically increasing as in an "ideal" rate response.
  - The frequency of the oscillation varies widely with rotor type, as shown in the comparison between the OH-6A and AH-1G with varying degrees of augmentation (see Figure 1c).
- Example 2. Rate Command/Attitude Hold (RCAH). There are a number of ways to mechanize an RCAH flight control system, one of which is to simply integrate the input to an attitude command system such as shown in Figure 2. Some proportional input ( $K_\delta$ ) is also necessary to maintain an adequate Bandwidth, or said another way, to quicken the initial response. There are other ways to achieve RCAH, but the generic features of the resulting system are essentially the same as shown in Figure 2, unless the attitude hold loop is switched in and out of the system as a function of stick position or force (sometimes referred to as Rate Command/Attitude Retention



$$\frac{\theta}{\delta_B} = \frac{-146(.016)e^{-.15s}}{(.44)[-29, .36]} \quad (\text{AH-1G})$$

$$\frac{\theta}{\delta_B} = \frac{-74(.016)e^{-.15s}}{(2)[0, 0.40]} \quad (\text{OH-6A})$$

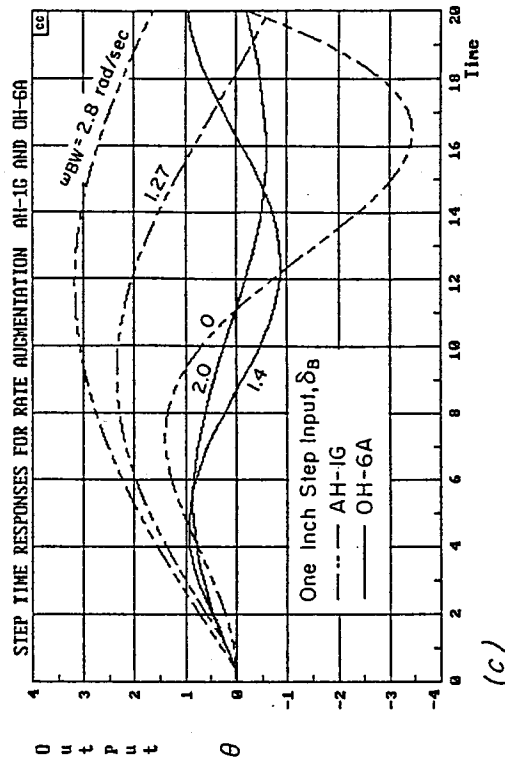
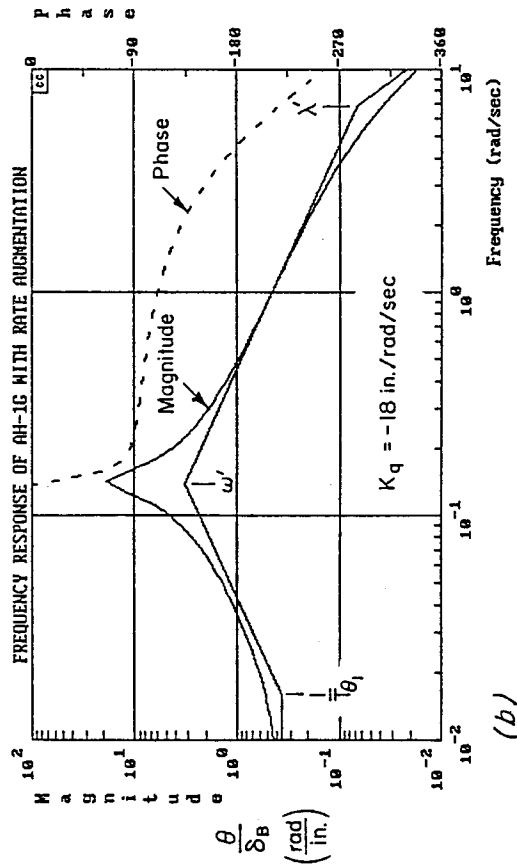


Figure 1(3.2.5). Generic Characteristics of Rate Augmentation for AH-1G and OH-6A

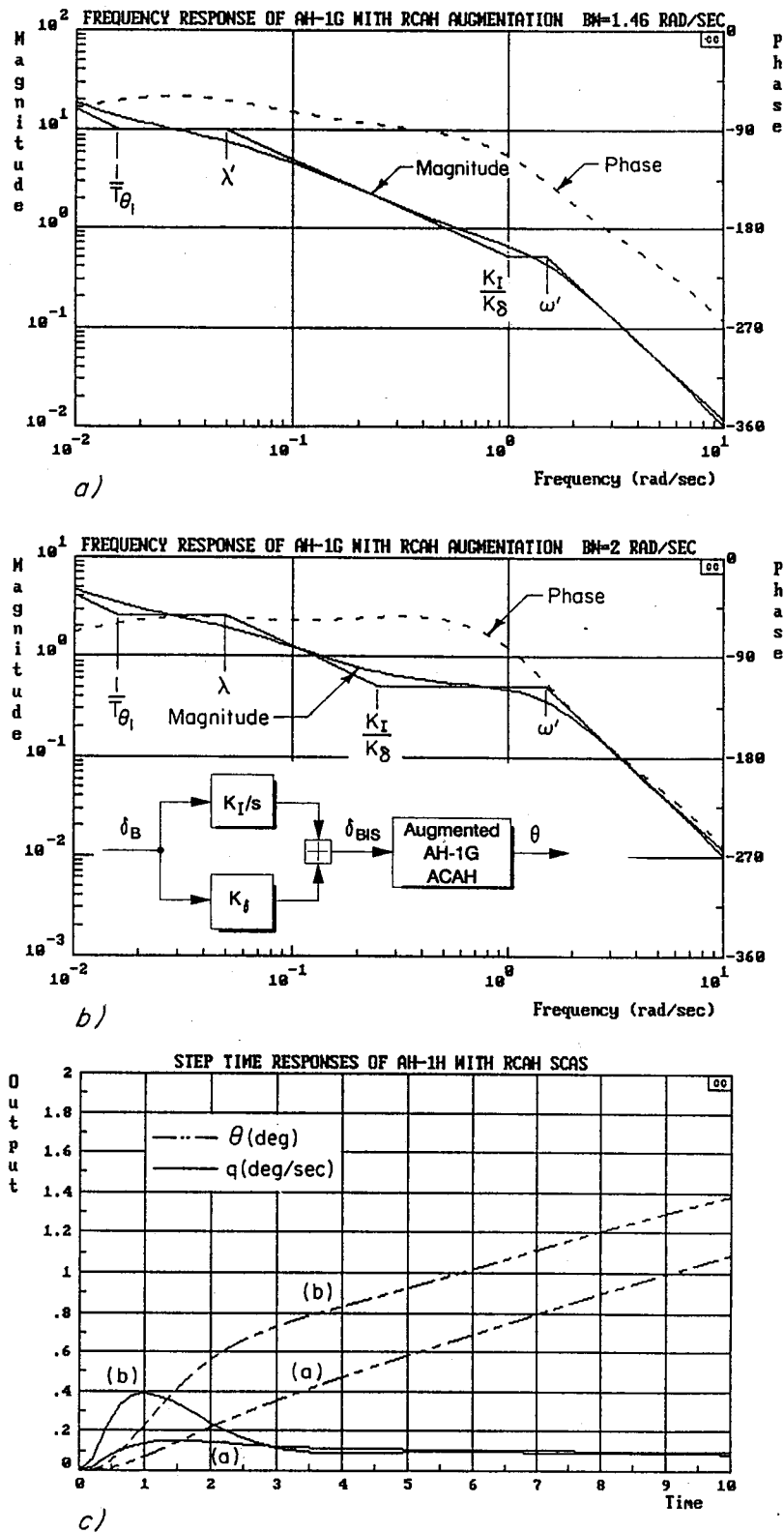


Figure 2(3.2.5). Generic Characteristics of RCAH Augmentation

(RCAR)). For RCAR systems, the command/response characteristics are, of course, identical to Rate systems. Both mechanizations are classified as Rate Response-Types plus Attitude Hold Response-Types in the specification. The characteristics of RCAH augmentation (as distinct from RCAR) are summarized below.

- With RCAH, the low frequency mode defined by  $\omega'$  is stabilized by the attitude hold loop (see Figure 1(3.2.7)), thereby eliminating the low frequency oscillation which shows up as a droop in the time domain step response for a rate or RCAR augments.
- With RCAH the pitch rate time response to a step input may exhibit some initial overshoot (attitude quickening due to proportional input path,  $K_\delta$ ) followed by a steady rate, i.e., no droop. Two cases are shown in Figure 2, to illustrate that increasing the Bandwidth tends to increase the initial pitch rate overshoot. The implications of large pitch rate overshoot on handling qualities is discussed in "Guidance for Application."
- With RCAH the overall frequency response is reasonably well approximated by a K/s shape. This includes a Bode magnitude slope of approximately -20 dB/decade, and a phase in the vicinity of -90°.
- The Rate or RCAR augments (Figure 1) allows higher Bandwidth without the need for pitch rate overshoot associated with RCAH. (The pitch rate overshoot is due to the proportional path which is required for Level 1 Bandwidth with an RCAH augments.)
- The RCAH augments is easily distinguished from a Rate or RCAR system by the step time response in that it always exhibits an essentially linear increase in attitude without low frequency oscillations, reversals, etc.

There is no requirement in the specification for conventional stick free static stability, i.e., stick force per knot. However, the last sentence in this requirement serves to insure that the basic intent of the stick free static stability requirement is retained. Specifically, the stick force required to pitch from one steady attitude to another should not reverse sign. This will allow Rate Command/Attitude Hold where the stick force returns to zero, and will eliminate statically unstable configurations. The requirement is stated in terms of pitch attitude instead of airspeed since airspeed is of less importance for rotorcraft than for fixed-wing flying qualities.

#### c. Supporting Data

During the in-flight variable stability rotorcraft handling qualities experiment discussed as supporting data for Paragraph 3.3.2.1, it was noted that rate damped configurations were well liked by the pilots (HQR = 2-3 by all of the five subject pilots). The addition of attitude retention to the basic rate system (RCAR) also resulted in HQR = 2-3 by all five pilots, and the associated commentary indicated that the pilots did not notice any benefits, or shortcomings, due to the attitude retention feature. Likewise, RCAH augmentation was also found to have a negligible effect on the rotorcraft flying qualities for fully attended NOE tasks in conditions of good visibility\*. That is to say, all of the response shapes in Figures 1 and 2 were found to be acceptable, as long as the Bandwidth was high enough (see Paragraph 3.3.2.1 for additional details). In fact, even Attitude Command/Attitude Hold augmentation was not found to have any influence on pilot opinion for the fully attended tasks in UCE = 1. Summarizing these results, the augmentation type was not found to be a factor in the flying qualities evaluations for precision, aggressive NOE maneuvering. On that basis, the specification does not require a specific response shape for fully attended maneuvering tasks in conditions of good visibility (i.e., UCE = 1).

There is a considerable body of pilot rating data which indicates that the shape of the response (i.e., Response-Type) is extremely important for operations in conditions of degraded visual cueing (UCE > 1) and/or high pilot workload (see Paragraph 3.2.2.1). This data indicates conclusively that Rate Response-Types are not desirable (i.e., are Level 2 or worse) for flying NOE in conditions of degraded visual cueing (UCE > 1), and/or where divided attention requirements are high. For additional detail, see Paragraph 3.2.2, subsection 2, especially Figure 9(3.2.2). The pilot rating data in Figure 9(3.2.2) indicates that a Rate Response-Type yields average Cooper-Harper Handling Qualities Ratings (HQRs) of about 5 for UCE = 2 and 6.5 for UCE = 3. The reason for this result can be identified from closed-loop pilot vehicle analysis. Such an analysis is presented in the Supporting Data for Attitude Response-Types (Paragraph 3.2.7).

#### d. Guidance for Application

Since the Rate Response-Type is the lowest level of augmentation allowed by the specification, there is no reason to conduct a specific test to identify it. That is, a rotorcraft which has been designed for operation in UCE = 1, and for mission scenarios where the pilot can devote essentially full time to aircraft control, the required Response-Type, from Table 1(3.2), is Rate. In such a case, compliance involves

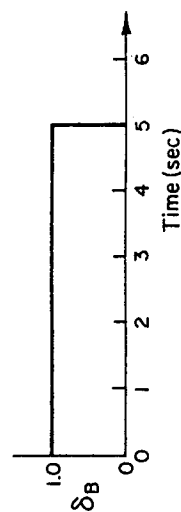
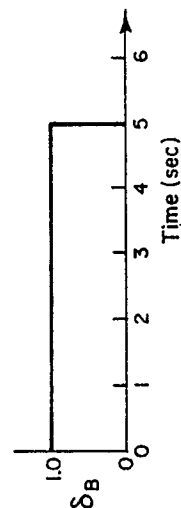
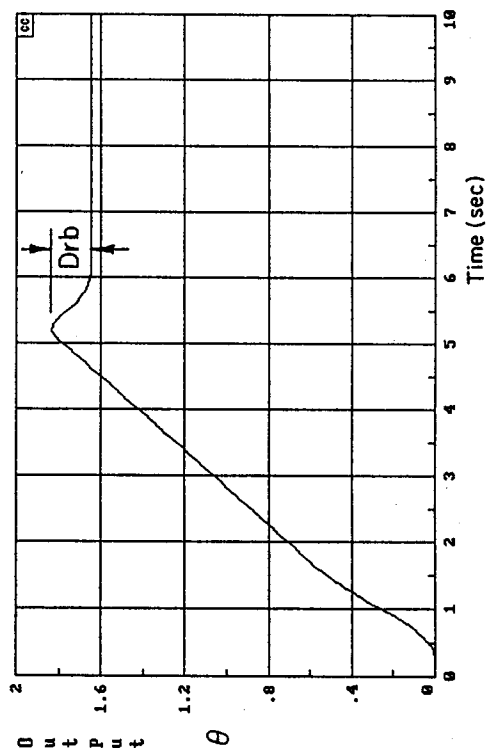
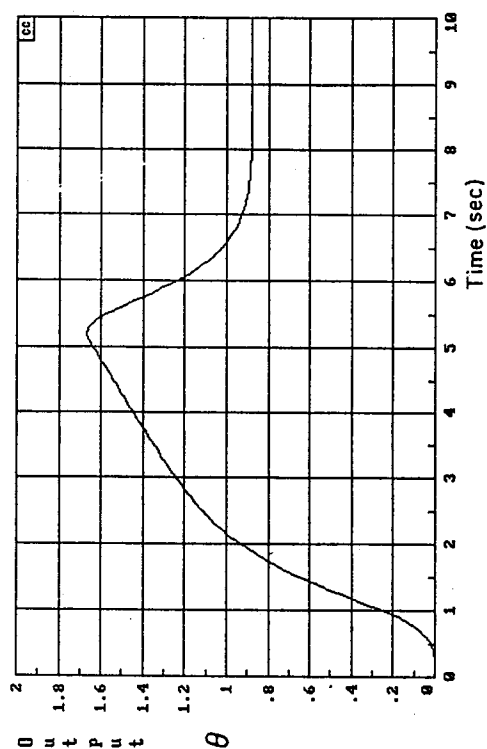
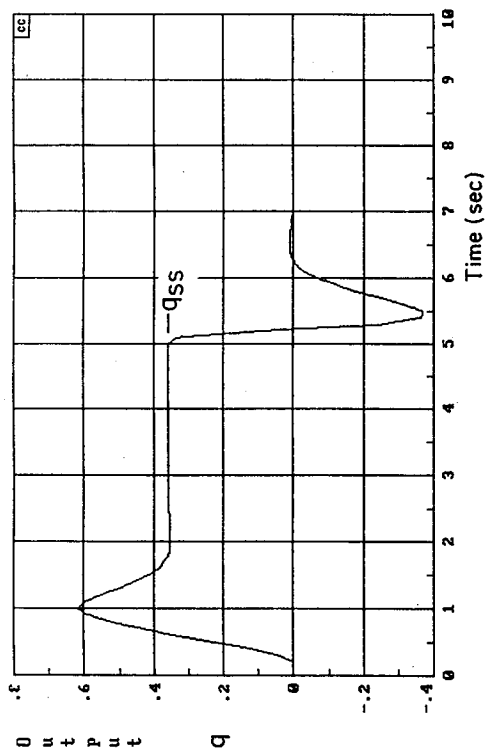
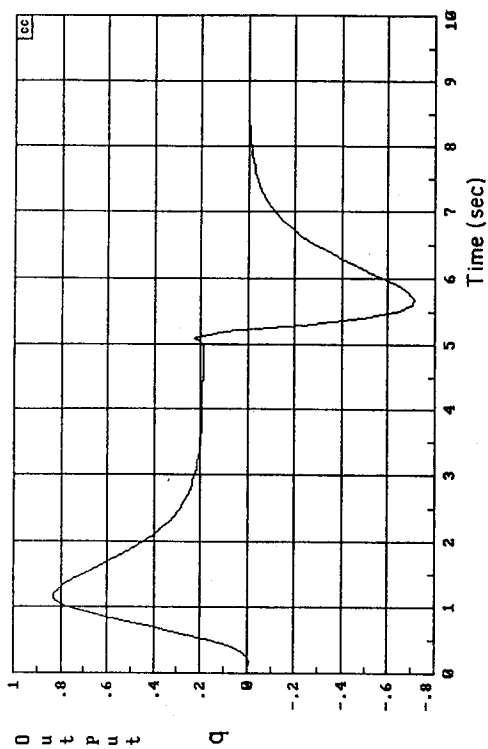
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\*However, it was found that RCAH augmentation is susceptible to gain-margin-limiting, resulting in a Bandwidth which is considerably less than would be indicated from the 45° phase margin frequency. This is discussed in detail in the Rationale for Paragraph 3.3.2.1 (subsection 4).

only meeting the dynamic response criteria of Sections 3.3 and 3.4 without further consideration of Response-Type.

Rate and RCAH Response-Types with large angular rate overshoot can have a potentially undesirable "drop-back" tendency which results in excessive abruptness, and degraded ability to control attitude or heading. The drop-back parameters are illustrated in Figure 3, and the criterion is defined in Reference 10 as  $Drb/q_{ss}$ . The best maneuver to test for excessive drop-back is to rapidly change from one steady attitude to another using a rectangular shaped control input as shown in Figure 3. A good rule-of-thumb, based on fixed-wing experience for precision attitude control (unpublished specification for European Fighter Aircraft (EFA)), is that the drop-back should not exceed 1.0 sec, and for aggressive tracking, values less than 0.25 sec are desirable. Until data more directly related to rotorcraft are available, these drop-back limits shall remain in the category of recommended only. Excessive drop-back is most common for systems where the force applied to the cockpit controller is approximately zero for zero angular rate (i.e., the trim gradient of control force with attitude is zero). From Figure 3 it can be seen that drop-back is a result of large pitch rate overshoot, and occurs when the control input is removed. The effect is therefore minimized for systems where the cockpit controller input is not removed to achieve a steady attitude, i.e., ACAH or rotorcraft with moderate-to-large stick-fixed static stability. Since drop-back results directly from pitch rate overshoot, it is most likely to occur with the RCAH type augmentation discussed in Example 2 in the "Rationale for Requirement" section.

Some examples of drop-back are shown in Figure 3. Figure 3a is the response of roll angle and roll rate to a boxcar input to Case 7 from the variable stability rotorcraft flight tests discussed in Paragraph 3.3.2.1 (also see Reference 157). Case 7 was specifically derived to expose any problems that might arise due to a large value of pitch and roll rate overshoot. The pilot ratings for Case 7 are summarized in Supporting Data for Paragraph 3.3.2.1. There it is noted that the ratings were in the Level 2 region for the more aggressive maneuvers, and that all five of the evaluation pilots complained of excessive abruptness. The value of drop-back is seen to be twice the desired value of 0.25. The drop-back calculated for the Ref 170 ADOCS demonstrator is shown in Figure 3b, and is seen to be considerably higher than the recommended maximum of 1. Since the attitude control of the ADOCS was considered to be acceptable by a large number of pilots, it is doubtful if drop-back is as significant for rotorcraft as it is for fixed-wing aircraft. The reason for this is not known at this time. However, it would seem safest to maintain drop-back to values below 0.25 to insure good flying qualities. This would be especially important for rotorcraft designed to accomplish aggressive tracking tasks (e.g., scout/attack aircraft).



b) ADOCS Model from Ref. 170 ( $Drb/q_{ss} = 3.75$ )

a) Case 7 from Para. 3.3.2.1 ( $Drb/q_{ss} = 0.57$ )

Figure 3(3.2.5). Definition and Examples of Dropback



The pitch rate overshoot following a step controller input is an alternate measure of attitude abruptness\*. On the basis of the results discussed above for Case 7 (Figure 3a), it is recommended that the angular rate overshoot be kept at values well below 2.0, i.e.,  $q_{pk}/q_{ss} < 2$ .

e. Related Previous Requirement

MIL-H-8501A (Reference 31) dictated specific requirements on the gradients of stick force and position with airspeed. These were contained in Paragraph 3.2.9 of Reference 31 and are summarized below.

- The rotorcraft shall possess positive static longitudinal control force and control position stability with respect to speed, except as follows.
- In the speed range between 15 and 50 kts forward, and 10 to 30 kts rearward, a moderate degree of instability is allowed. The magnitude of the change in the unstable direction shall not exceed 0.5 inches for stick position, or 1 lb of stick force.

As a result of this requirement, many helicopter flying qualities investigations placed considerable emphasis (and expended many flight hours) on the determination of stick force and position gradients with airspeed. Invariably, these gradients were very small, and the difference between a "pass" and a "fail" have little to do with the rotorcraft flying qualities. This was shown in References 21 through 23 where pilots were relatively insensitive to static stability, and found the addition of Attitude Hold to be of significantly more benefit for instrument approach tasks. However, it is always desirable to be able to change pitch and roll attitude without reversals in the stick force. Hence the present requirement regulates against stick force reversals.

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\*The removal of the step input can be considered as a step in the opposite direction. The pitch rate overshoot tends to be more severe when the input is removed, and therefore the parameter should be measured at that point (e.g., see Figure 3).

a. Statement of Requirement

3.2.6 Character of Attitude Hold and Heading Hold Response-Types. If Attitude Hold or Heading (Direction) Hold is specified as a required Response-Type in Paragraph 3.2.2, the attitude, or heading, shall return to within  $\pm 10$  percent of the peak excursion, following a pulse input, in less than 20 seconds for  $UCE=1$ , and in less than 10 seconds for  $UCE>1$ , as illustrated in Figure 3(3.2). Roll attitude and heading shall always return to within 10 percent of peak in less than 10 seconds. The peak attitude excursions for this test shall vary from barely perceptible to at least 10 degrees. The attitude or heading shall remain within the specified 10 percent for at least 30 seconds for Level 1. The pulse input shall be inserted directly into the control actuator, unless it can be demonstrated that a pulse cockpit controller input will produce the same response.

For Heading Hold, following a release of the directional controller the rotorcraft shall capture the reference heading within 10 percent of the yaw rate at release. In no case shall a divergence result due to activation of the Heading Hold mode.

b. Rationale for Requirement

Attitude hold functions provide regulation against atmospheric disturbances and interaxis coupling. The quality of such a function is logically based on its ability to return the rotorcraft to its trimmed attitude following a disturbance. On this basis, the specification is based on the time required to return to trim following a disturbance. The disturbance is simulated by a test input injected directly into the control actuator, unless the manufacturer can show that this is identical to pulsing the cockpit controller. The rationale for this is given below.

The generic relationships between the response of the rotorcraft to cockpit controller inputs and to external disturbances are shown in Figure 1. For high loop gains, the response to controller inputs,  $\theta/F_s$ ,

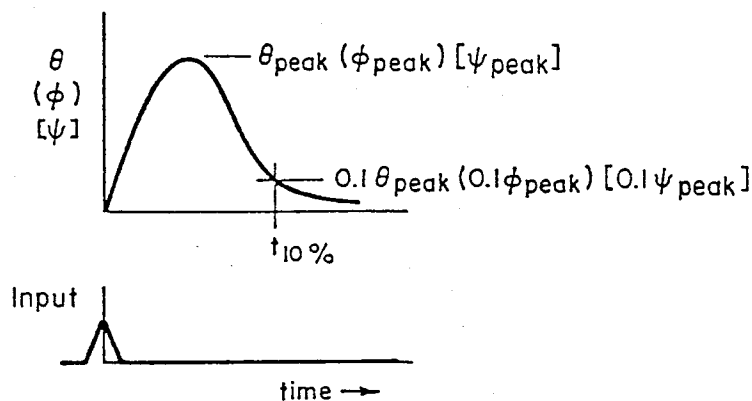


Figure 3(3.2). Illustration of Pulse Response Required for Attitude Hold and Direction Hold Response-Types

depends on the ratio of the input to feedback equalization,  $G_i/G_f$ . The response to external disturbances,  $\theta/\eta$ , depends inversely on the product of the feedforward and feedback equalization,  $G_a G_f$ , but not on the input shaping,  $G_i$ . The output/input signals used to check the disturbance regulation characteristics of an augmented rotorcraft should have the same transfer function as  $\theta/\eta$ . This is well approximated by injecting the test signal directly into the actuator. Such a test signal is shown in Figure 1, and is used for compliance with several requirements in this specification, all having to do with disturbance rejection. These are listed below:

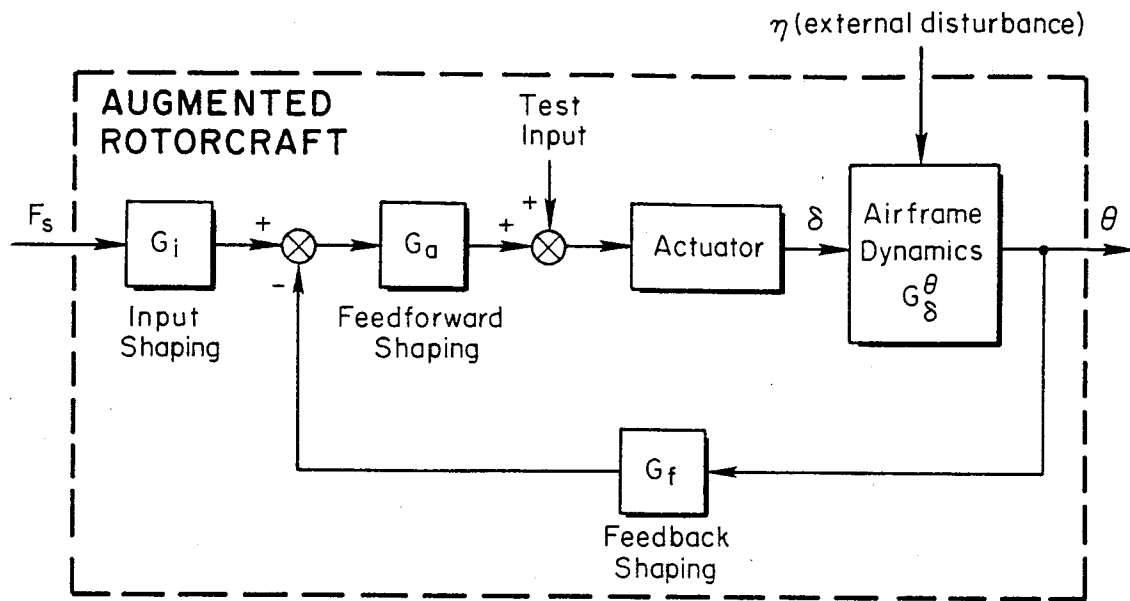
- Paragraph 3.2.9.1 -- Character of Altitude Rate Command with Altitude Hold
- Paragraph 3.2.12 -- Requirements for Inputs to Control Actuator
- Paragraphs 3.3.2.3 and 3.3.7 -- Short Term Pitch, Roll, and Yaw Responses to Disturbance Inputs -- Hover and Low Speed
- Paragraph 3.4.10 -- Short Term Pitch, Roll, and Yaw Responses to Disturbance Inputs -- Forward Flight

For some augmentation schemes, the disturbance can be entered directly through the cockpit controller, by showing analytically that the input and feedforward shaping will not affect the settling time of this requirement. This will usually be the case for Rate and ACAH systems since  $G_i$  and  $G_a$  typically will not modify the pulse input. However, Rate Command/Attitude Hold systems require integration in the input and/or forward loop shaping networks, and this has a long term effect on the response after the input is removed. The specification user is allowed to use cockpit control inputs to satisfy this requirement if it can be shown that  $G_i$  and  $G_a$  will not affect the settling times of Figure 3(3.2).

The requirement is worded so that it is necessary to show compliance for a range of peak attitude excursions from barely perceptible up to 10 degrees. The former is included to insure that there are no unacceptable deadbands in the flight control system, and the latter to eliminate the possibility of poor disturbance regulation due to servo-actuator saturation, or other nonlinear effects.

#### c. Supporting Data

The values of settling time used as the Level 1 limit are based on the bandwidth limits for the command/response requirement of Paragraph 3.3.2.1. The rationale for this is that the pilot rating used to support the bandwidth requirements were based on configurations where the gust rejection bandwidth and the command/response bandwidth were identical. Therefore, it is not possible to determine whether the experimentally derived flying qualities limits are based on gust rejection or response to controls. Until data becomes available, it is safe to conclude that



- Response to control input

$$\frac{\theta}{F_s} = \frac{G_i G_a G_{\delta}^{\theta}}{1 + G_a G_f G_{\delta}^{\theta}}$$

$$\text{High gains} \Rightarrow 1 + G_a G_f G_{\delta}^{\theta} \doteq G_a G_f G_{\delta}^{\theta}$$

$$\boxed{\frac{\theta}{F_s} \doteq \frac{G_i}{G_f}}$$

- Response to external disturbance

$$\frac{\theta}{\eta} = \frac{G_{\eta}^{\theta}}{1 + G_a G_f G_{\delta}^{\theta}} \Rightarrow \frac{G_{\eta}^{\theta}}{G_a G_f G_{\delta}^{\theta}}$$

Figure 1(3.2.6). Generic Relationships Between the Response to Cockpit Controller and Disturbance Inputs

the minimum disturbance rejection bandwidth should be at least equal to the command/response bandwidth.

A representative family of ideal ACAH time responses was calculated and developed to show the effect of damping ratio. For an ideal attitude response (i.e., no time delay), Reference 52 shows that:

$$\frac{\omega_{BW}}{\omega_n} = \zeta + \sqrt{\zeta^2 + 1}$$

That is,  $\omega_{BW}$  is a function of  $\omega_n$  and  $\zeta$ .

As an example, Figure 2 shows the impulse time responses for several values of  $\zeta$  and  $\omega_n$ , all yielding constant bandwidth of 3 rad/sec. The most sluggish system in terms of time to return to within 10 percent of peak response is dictated by the combination of high damping and low natural frequency. A reasonable upper limit on damping ratio was chosen as  $\zeta = 1.3$ . Though it is not shown here, if time delay is included, the effective natural frequency of the attitude hold feature must increase considerably to hold bandwidth constant. It follows that for a constant bandwidth, time delay has the effect of quickening the impulse response, and therefore the most sluggish response (longest settling time) is one for which  $\tau_p = 0$ .

Using the equation given above, and the bandwidths defined in Paragraph 3.3.2.1, impulse responses can be generated for specifying time response limits for  $\zeta = 1.3$ . Such time responses are shown in Figure 3. The numbers given for Level 1 in this paragraph were obtained directly from these plots. A pitch attitude settling time of 10 seconds was selected for UCE > 1 since a bandwidth of 2 rad/sec is required. For UCE = 1, the pitch attitude bandwidth requirement is relaxed to 1 rad/sec, and hence a settling time of 20 seconds is specified (see Figure 3). The minimum required bandwidth for roll and heading is 2 rad/sec regardless of UCE and hence a 10 second settling time is specified.

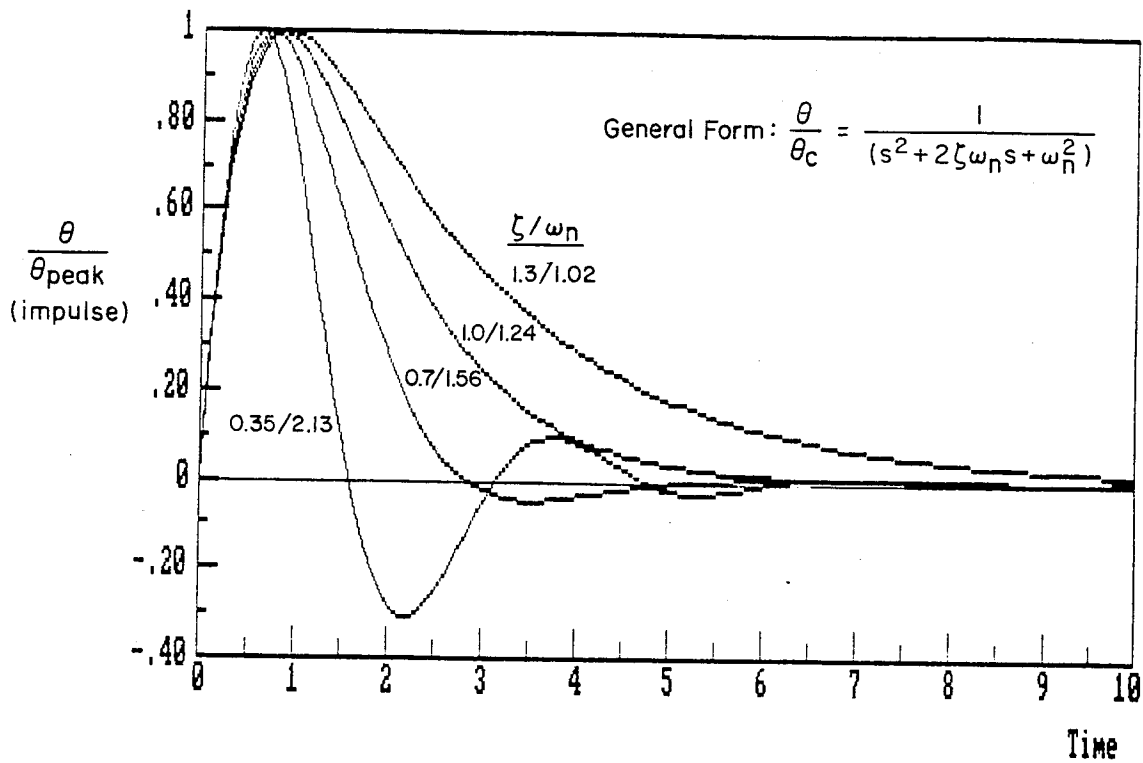


Figure 2(3.2.6). Typical Ideal Impulse Responses of Attitude Hold Systems with Constant  $\omega_{BW\theta} = 3 \text{ rad/sec}$

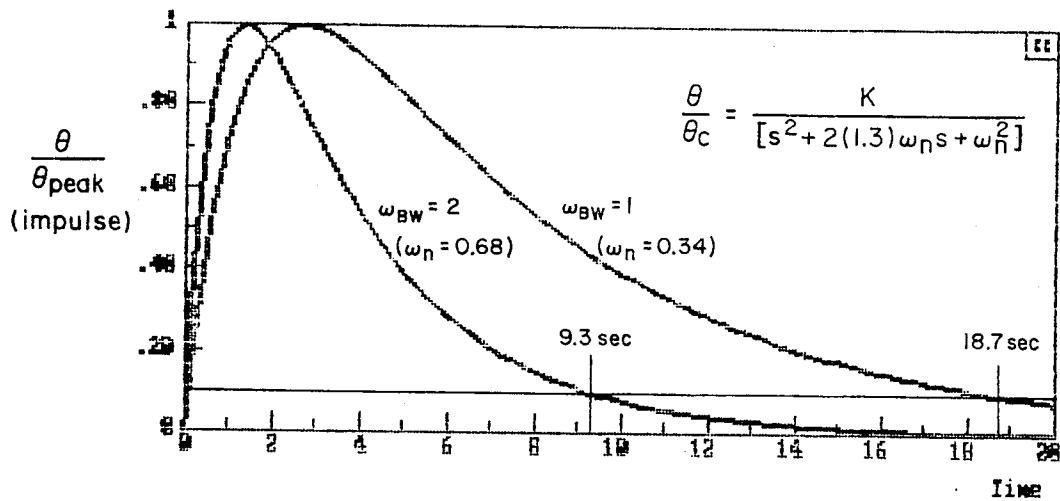


Figure 3(3.2.6). Limiting Cases for Attitude Hold

a. Statement of Requirement

3.2.7 Character of Attitude Command Response-Types. If Attitude Command is specified as a required Response-Type in Paragraph 3.2.2, a step cockpit pitch (roll) controller force input shall produce a proportional pitch (roll) attitude change within 6 seconds. The attitude shall remain essentially constant between 6 and 12 seconds following the step input. However, the pitch (roll) attitude may vary between 6 and 12 seconds following the input, if the resulting ground-referenced translational longitudinal (lateral) acceleration is constant, or its absolute value is asymptotically decreasing towards a constant. A separate trim control must be supplied to allow the pilot to null the cockpit controller forces at any achievable steady attitude.

b. Rationale for Requirement

The fundamental benefit of the Attitude Command Response-Type is that it allows a stabilized hover, with minimum pilot workload ( $HQR \leq 3$ ), in conditions of degraded visual cueing ( $UCE = 2$ ) and/or significant divided attention. Experimental evidence, supported by analysis, is given below.

The supporting analysis shows that the rotorcraft position response should be no more than two integrations from the cockpit controller input, at path mode frequencies, to realize the necessary workload relief in  $UCE = 2$ . An alternative acceleration criterion has been included as a more general (albeit more difficult to test) criterion, that directly satisfies this fundamental requirement. The attitude criterion is retained as primary, because it is easily tested, and also satisfies the first-principle requirement. However, it is possible to fail the attitude requirement and still meet the requirement for rotorcraft position equal to or less than two integrations from the controller input at mid-to-low frequencies. The acceleration criterion would be invoked for such cases. The 6 second time period was selected as representative of the "mid-to-low" frequency range, in that any attitude transients will be over by that time for attitude systems with adequate Bandwidth for Level 1 flying qualities in conditions of degraded visibility,  $UCE = 2$  or 3 (i.e., 2 rad/sec). A total of 12 seconds of time history is required to insure that the response is really essentially constant, and not simply at, or approaching, an inflection point.

c. Supporting Data

1. Pilot Rating Data and Commentary

The need for ACAH in this specification is based on experimental data which shows that attitude stabilization is necessary to achieve Level 1 flying qualities for low speed and hover operations in conditions of degraded visual cueing ( $UCE > 2$ ), and when division of

attention is required of the pilot at the controls (Table 1(3.2)). This data consists of the following observations.

- Experience with ground-based simulation has consistently shown that precision and aggressive hovering tasks are rated Level 2 even with the best Rate Response-Types, and that Level 1 ratings are easily achieved with ACAH. This is discussed in the Supporting Data for Paragraphs 3.3.2.1 (under "Review of Simulation Data") and 3.2.2 (under "Near-Earth; UCE = 2; Ground-Based Simulation"). The inability to accomplish normal hovering with Rate Response-Types on ground simulators has been attributed to a lack of adequate visual cueing with the then-available visual displays\*. Hence, it may be reasoned that ACAH is the proper Response-Type when visual cueing is degraded (i.e.  $UCE > 1$ ). These results have been extended to the flight environment as discussed below.
- In the Reference 187 variable stability rotorcraft flight experiments (discussed in Supporting Data for Paragraph 3.2.2, subsection 2,) ACAH was found to significantly reduce pilot workload when compared to RCAH or Rate Response-Types, of the same Bandwidth, when operating in conditions of poor visual cueing. In addition to being an advantage in poor UCE, ACAH has also been shown to be of value for divided attention tasks as discussed below.
- In the Reference 157 flight tests, the evaluation pilot was required to adjust the control sensitivity potentiometers while in hover. This divided attention task was found to be much easier with ACAH than with Rate or RCAH. This is explained by the fact that the pilot does not have time to perceive and feed back, at some minimum sample rate, the necessary state variables to stabilize the helicopter when attention is required to accomplish non-flying tasks.

The observations noted above have been obtained from a variety of experiments over the past 10 years, and consistently show that it is easier to stabilize a hovering vehicle (helicopter or VTOL) if it can be classified as an Attitude Response-Type. Given this experience, the following analysis serves to illustrate the fundamental principles that cause ACAH to provide reduced pilot workload in degraded visibility ( $UCE \geq 2$ ) and/or divided attention situations.

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\*There is evidence that the significant increase in microtexture available with the newest digital image generators may alleviate this deficiency.



## 2. Supporting Analysis

The generic characteristics of the time and frequency response characteristics of an attitude command augments are shown for the AH-1G in Figure 1. The following characteristics are of interest:

- The low-frequency second-order mode defined by  $\omega$  in Figure 1 is stabilized, eliminating mid-to-low frequency modes, except for a pole-zero pair defined by  $\lambda'$  and  $1/T_{\theta 1}$ .
- The low-frequency pole-zero pair can be effectively eliminated via feedforward equalization,  $(s+a)/s$ , since low-frequency droop is classically reduced or eliminated by the use of a parallel integrator. Compare the solid and dashed time histories in Figure 1c.
- Without feedforward parallel integration ( $a=0$ ), the spread between  $1/T_{\theta 1}$  and  $\lambda'$  decreases as the loop gain, and hence the system Bandwidth, is increased (see dashed time histories in Figure 1c). For low gain, low Bandwidth, cases (e.g., the 1.4 rad/sec case in Figure 1c), the attitude droop is significant. In fact, it would be difficult to interpret the response as "essentially constant" as required for an Attitude Response-Type. However, the acceleration criterion would allow that system to be categorized as an Attitude Command Response-Type.
- Recall that 2 rad/sec is the minimum Bandwidth allowed for  $UCE > 1$  (on the Bandwidth criterion boundaries in Figure 1c(3.3)). Increasing the bandwidth from 1.4 to 2.1 rad/sec (dashed lines in Figure 1c) decreases the droop to the point where the response is "essentially constant."
- Attitude transients are over well before the 6 seconds specified by the criterion. The 1.4 rad/sec Bandwidth system is marginal in that respect, but the potential discrepancy for such low Bandwidth systems is academic since a minimum of 2 rad/sec is required to provide the workload relief in  $UCE = 2$  to 3. The Attitude Response-Type is required only for  $UCE = 2$  or 3, and for conditions of divided attention. Both of these require a 2 rad/sec Bandwidth (Figure 1c(3.3)).

This illustration indicates that attitude augmenters that do not employ parallel integration in the forward loop (i.e.,  $a = 0$ ), will always exhibit some low-frequency droop, which decreases with increasing Bandwidth (by increasing the loop gain,  $K_{\theta}$ ). It will be shown in the following analysis that such droop does not have a direct effect on the ability to stabilize position. This is the reason for the wording in the requirement that attitude must reach an "essentially" constant value. However, large values of droop are associated with low

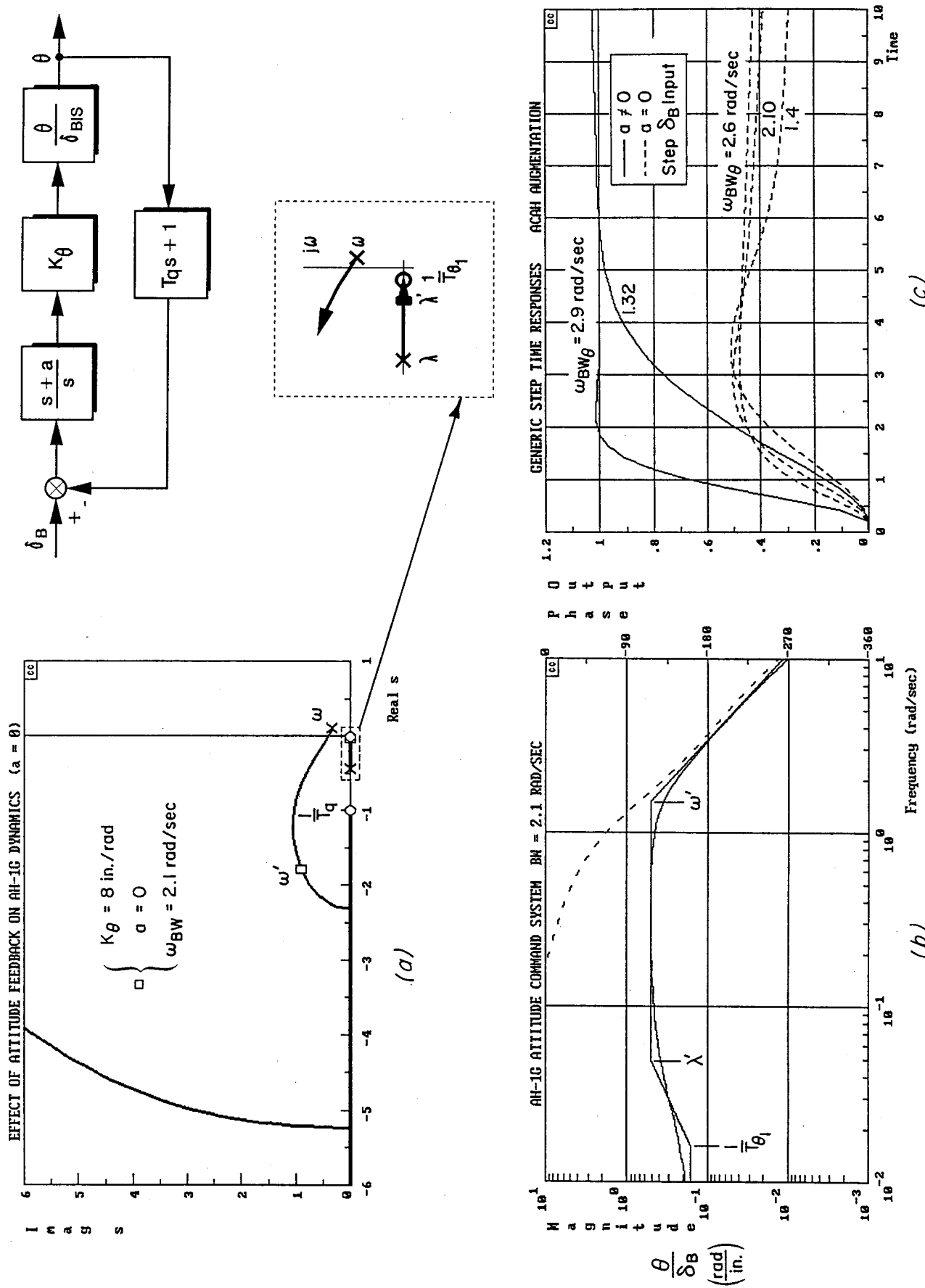


Figure 1(3.2.7). Characteristics of Attitude Command SCAS

Bandwidth, which significantly degrades the value of the Attitude Command Response-Type. This is discussed in detail in the Supporting Data for Paragraph 3.3.2.1.

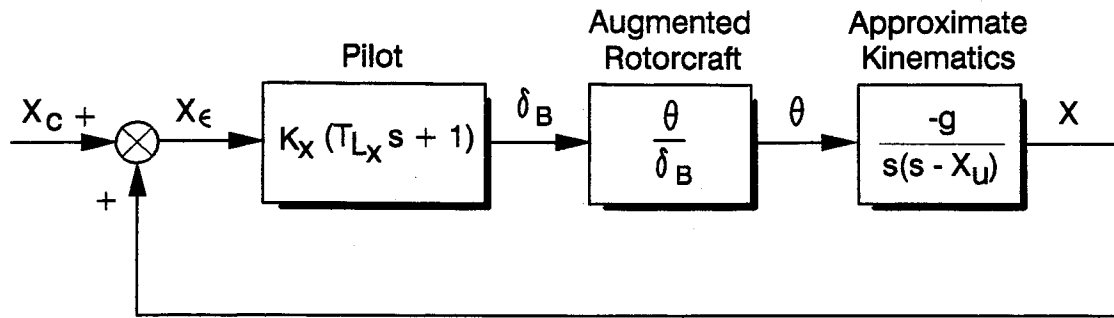
It is important to identify the specific characteristics of attitude command augmentation that provide pilot workload relief, to insure that the Response-Type definition applies in the general case. The following closed loop pilot-vehicle analysis will show that the fundamental characteristic that results in reduced workload, is that the position response, at mid-to-low frequency, should be no more than two integrations from the cockpit controller input. This is best illustrated by a closed-loop pilot-vehicle analysis, where the task is to maintain a position,  $x$ , as shown by the block diagram in Figure 2. These pilot loop closures are based on the premise that one effect of high workload and/or degraded visual cueing is a significant loss in the ability of the pilot to effectively close an attitude loop. This is simulated in the Figure 2 loop structure by attributing any augmentation of the  $\theta/\theta_c$  response to the augmented rotorcraft dynamics alone\* (i.e., no pilot attitude loop closure). It is assumed that the pilot is able to perceive and utilize translational velocity information, which is simulated by the lead term  $(T_{Lx}s + 1)$ .

The locus labeled "position mode" in Figure 2 determines the position damping and frequency characteristics as the pilot closes the position loop with increasing values of gain,  $K_x$ . The generic variation of this locus as a function of four augmentation types is shown in Figure 2. Note that with the Rate and RCAH systems, the position loop is unstable, for any value of position-loop lead,  $T_{Lx}$ , whereas it is stable for the ACAH flight control system if  $T_{Lx}$  is large enough. Of course, this does not mean that the pilot would actually allow the system to go unstable, but that it would be necessary to provide the

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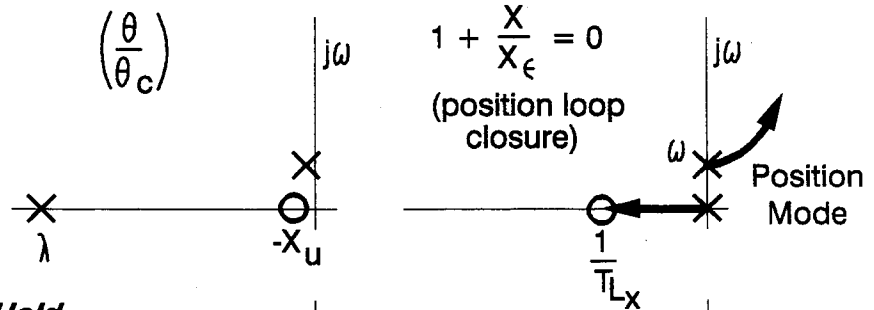
\*It is important to realize that the use of such closed loop pilot-vehicle analysis does not infer an ability to identically reproduce the actions of the human pilot with a simple linear model. Instead, it is used to determine the relative importance of the attitude loop closure in stabilizing position in hover. If, as in this case, the attitude closure is found to be important, it can be reasoned that the loss of attitude cues will have a significant effect. Furthermore, it infers that replacing the lost cues with stabilization is a viable possibility, worthy of pursuing with simulator, or if possible, flight experiments. Such experiments have been conducted, and are reported in detail in the Supporting Data for Paragraph 3.2.2 (subsection 2).

Finally, the loop structure infers that the pilot uses attitude, position, and position-rate as the stabilizing feedbacks. While these feedbacks are highly likely, it could be that other cues are also present. For example, suppose that pilots actually use linear acceleration cues in combination with attitude as stabilizing feedbacks. Since the loss of this combination of cues would have the same effect as the loss of only attitude cues, and since stabilizing attitude with augmentation works, the details of exactly what cues are used, and lost, is not important to the solution.



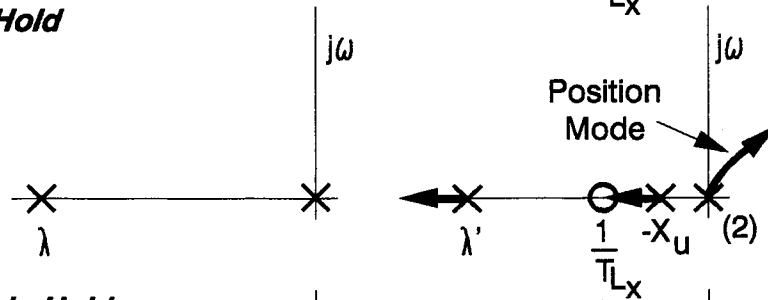
**Rate**

$$\frac{\theta}{\delta_B} = \frac{M_\delta (-X_U)}{(\lambda) [\zeta \omega]}$$



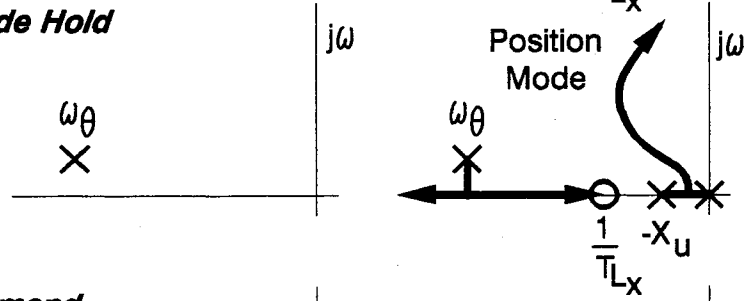
**Rate Command/Attitude Hold**

$$\frac{\theta}{\delta_B} = \frac{M_\delta}{(0)(\lambda)}$$



**Attitude Command/Attitude Hold**

$$\frac{\theta}{\delta_B} = \frac{M_\delta}{[\zeta \omega]}$$



**Linear Acceleration Command**

$$\frac{\theta}{\delta_B} = \frac{M_\delta}{[\zeta \omega]} \times \frac{(-X_U)}{(0)}$$

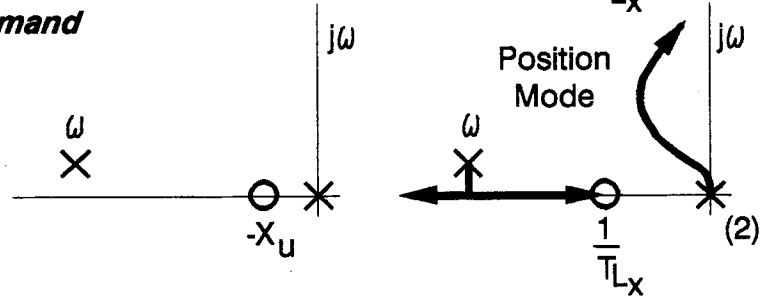


Figure 2(3.2.7). Effect of Augmentation Types on Pilot Position Loop Closure

required equalization (attitude loop closure and additional position lead) to stabilize the hover, at a considerable expense in workload. This has been verified experimentally, in that a significant improvement in pilot ratings has consistently been observed for ACAH, as compared to Rate or RCAH, in conditions of poor visual cueing and/or where division of attention requirements are high, e.g., see Reference 187 or Supporting Data for Paragraph 3.2.2 (subsection 2).

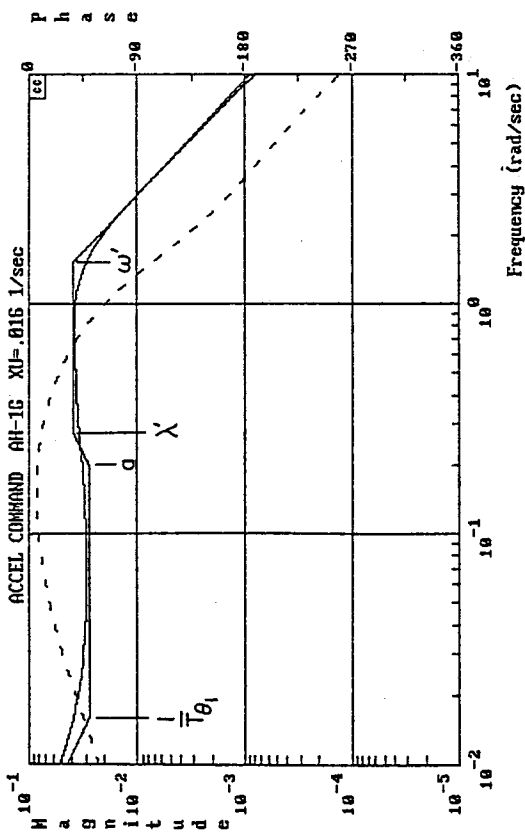
The important distinction between Rate and higher levels of augmentation is rooted in the existence of excess (more than two) mid-to-low frequency poles which must be stabilized by the pilot in the process of closing a position loop, i.e., to hover. In that context, if  $\theta/\delta_B$  has no mid-to-low frequency dynamics (i.e.,  $\theta/\delta_B \approx K$  at mid-to-low frequencies), the pilot is faced with an acceleration system which can be stabilized with a position lead ( $T_{Lx}s+1$ ). That is:

$$\frac{x}{\delta_B} \approx \frac{\theta}{\delta_B} \times \frac{-g}{s(s - X_u)} \approx \frac{\theta}{\delta_B} \times \frac{-g}{s^2} \quad (X_u \approx 0)$$

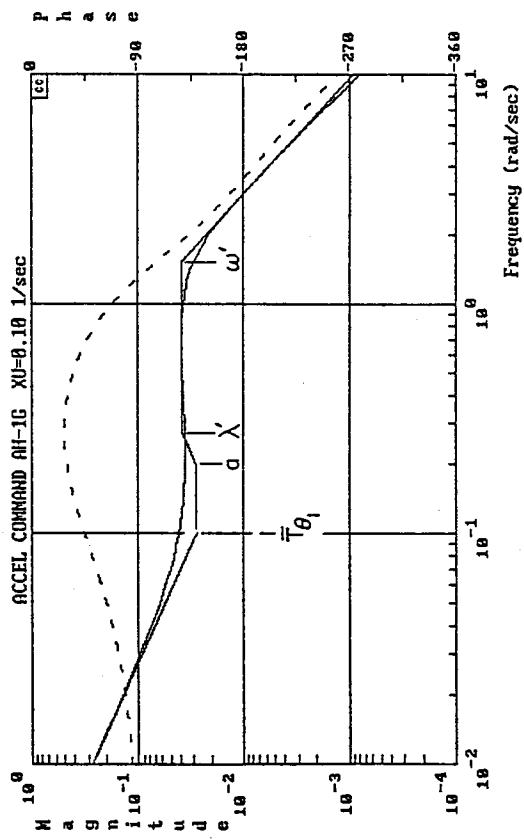
Therefore, if at mid-to-low frequencies,  $\theta/\delta_B \approx K$ , then  $\ddot{x} \approx K\delta_B$ . It follows that any augmentation that causes  $\theta/\delta_B$  to be well approximated by a constant at mid-to-low frequencies would be expected to provide the required pilot workload relief. The criterion accomplishes that objective by requiring attitude to be essentially constant within six seconds of a step cockpit controller input.

Consider now the effect of low-frequency droop, such as shown in Figure 1. Such droop is a result of a lead/lag in  $\theta/\delta_B$  at low frequency, and hence can only improve the ability to stabilize the position loop. Therefore, significant attitude droop is not a factor, except as a warning that the Bandwidth is probably below the required 2 rad/sec (for UCE = 2 or 3 and/or conditions of divided attention).

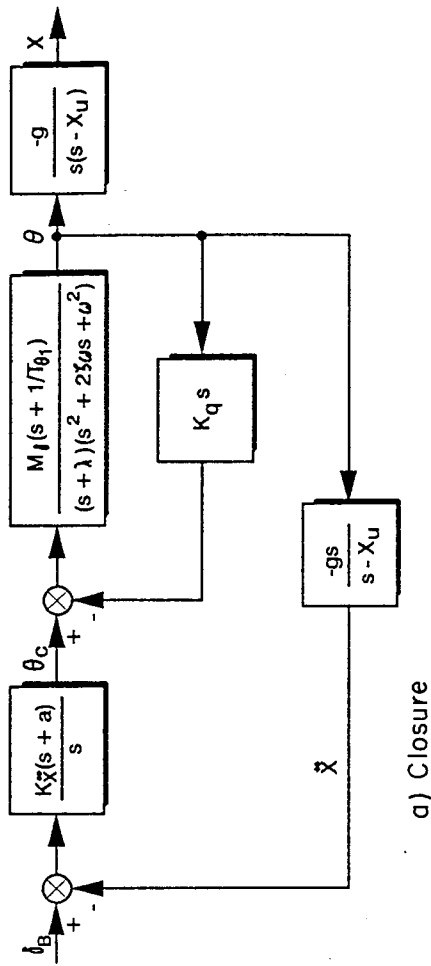
The alternative definition of the Attitude Response-Type is stated in terms of a constant translational acceleration, and is included to insure that a system that satisfies the fundamental requirement noted above (i.e.,  $\ddot{x}/\delta_B \approx \text{constant}$ ), but does not exhibit the specified constant attitude, will be properly classified. An example of such a system would be a linear acceleration command SCAS (with or without velocity hold), as shown in Figure 3. The term  $(s + 1/T_{\theta_1})/s$  occurs in the mid-to-low frequency region of the Bode plots of  $\theta/\delta_B$ . This term results in the slow pitch rate required to maintain the commanded acceleration, by offsetting the deceleration due to  $X_u$  (note that  $1/T_{\theta_1} \approx -X_u$ ). For nominal values of  $X_u$  (say -0.01 to -0.025 1/sec), this effect is small, and the acceleration command system has all the essential characteristics of an attitude command system. However, if  $X_u$  is large (say  $\leq -0.10$  1/sec), the effect is to violate the criterion based on constant attitude at mid-to-low frequencies, even though the fundamental requirement for acceleration command (i.e.,  $\ddot{x}/\delta_B \approx \text{constant}$ ) is ideally met. The function of the alternative requirement



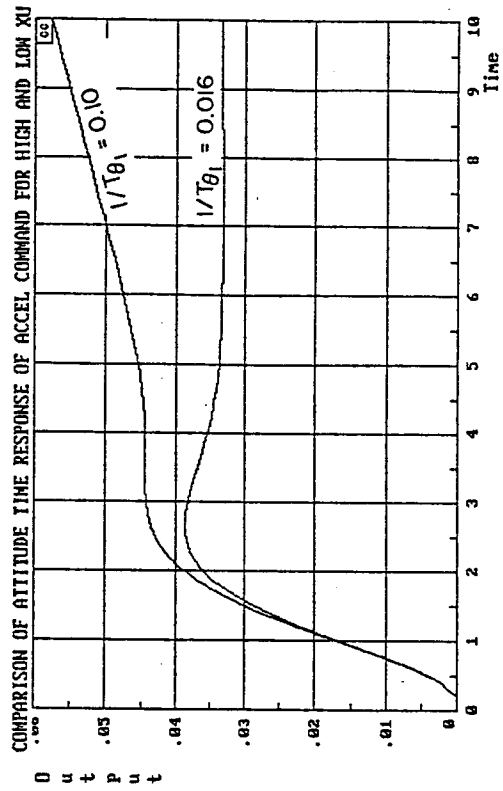
b)  $\theta/\delta_B (X_U = -0.016 \text{ 1/sec})$



c)  $\theta/\delta_B (X_U = -0.10 \text{ 1/sec})$



a) Closure



d) Attitude Time Responses

Figure 3(3.2.7). Generic Characteristics of a Linear Acceleration Command SCAS

is to handle such cases\*. The translational acceleration response to a step longitudinal controller input is shown in Figure 4 for several Response-Types. This figure shows that Rate and ACAH systems result in continuously increasing linear acceleration, and that the translational rate command system (TRC) results in a linear acceleration which approaches zero (as would be expected). The ACAH and linear acceleration command systems (both low and high  $1/T_{\theta 1}$ ) result in an essentially constant translational acceleration.

In the following subsection, the effect of a trim-follow-up on the definition of the Attitude Response-Type is analyzed. Such a trim follow-up is achieved by adding a parallel integrator,  $(s + K_I/K_\delta)/s$ , to the command path of an ideal ACAH augments. If the integrator constant,  $K_I/K_\delta$ , is set equal to  $-X_u$ , a system identical to the above discussed acceleration command system will result. However, if  $K_I/K_\delta > -X_u$ , the translational acceleration will steadily increase following a step controller input (because the pitch rate will exceed the value required to cancel  $-X_u$ ), and the system would therefore not be classified as an Attitude Command Response-Type,\*\* as is discussed in the following subsection.

### 3. The Impact of Trim Follow-up on ACAH

The addition of a trim follow-up to an ACAH augmentation system is analyzed herein to further illustrate the underlying reasons for degradations in handling qualities if the fundamental properties of ACAH are not preserved, and to justify the requirement for a separate trim control for Attitude Response-Types. This effectively requires that it be possible to turn off any trim follow-up in the pitch and roll controller, and that manual trim must be provided. It should be noted that for most conditions, the use of a trim follow-up with an ACAH SAS results in highly desirable handling qualities. However, in conditions of degraded visual cueing, it tends to degrade handling, and should be turned off.

It will be shown that the addition of a trim follow-up tends to eliminate the beneficial features of ACAH in the context of reducing pilot workload for precision hover in conditions of poor visual cueing, and for divided attention operations. It will also be noted that there may be some limited range of trim follow-up which would be acceptable. However, this could not be determined without an experiment including

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\*A logical alternative would be to simply require constant translational acceleration to a constant stick input without regard to attitude (i.e., eliminate the attitude requirement). This was considered, but rejected on the basis that the attitude definition is much more easily measured (in fact it can be identified without any instrumentation at all), and is appropriate for the large majority of cases.

\*\*In such a case, the rotorcraft position is three integrations away from the controller input, thereby violating the first-principle rule of only two integrations between these two variables.

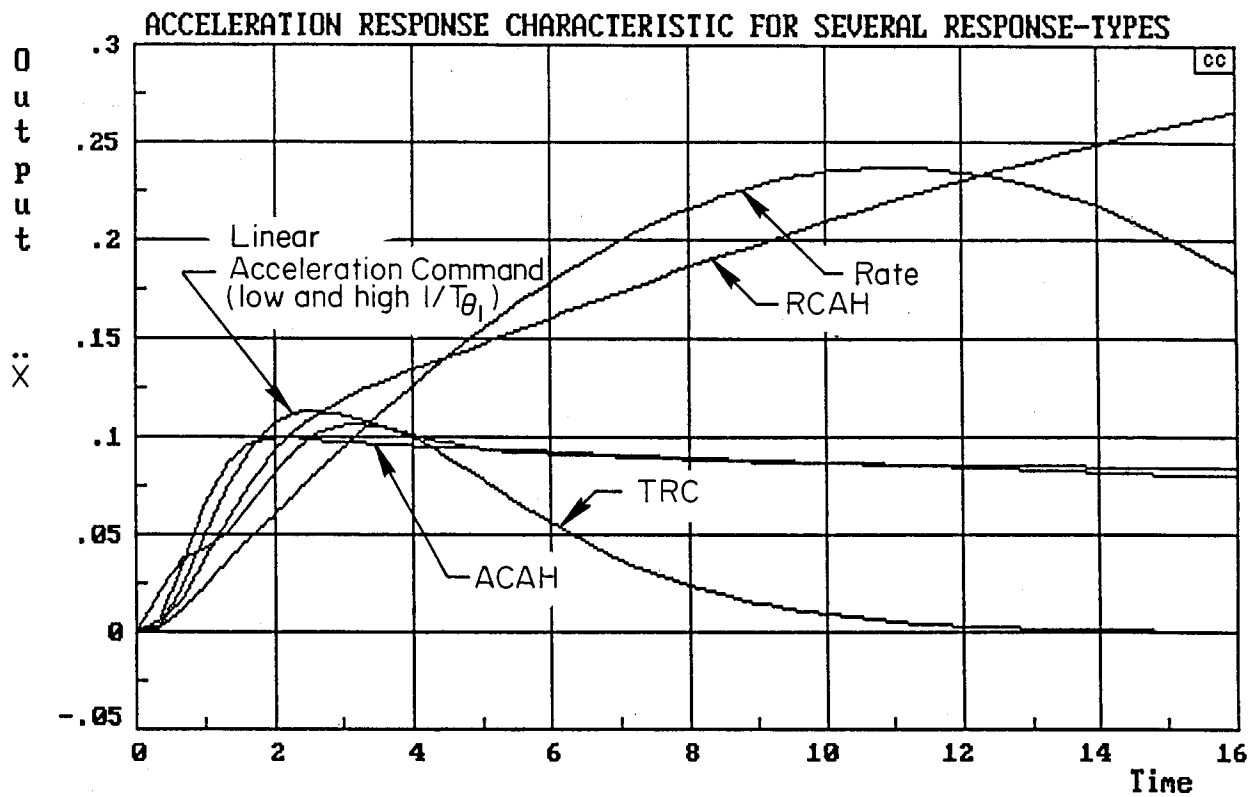


Figure 4(3.2.7). Translational Acceleration Response to Step Longitudinal Controller Input for Several Augmentation Types



flight in UCE=2 and simulated divided attention. In the absence of such data, the following closed-loop pilot-vehicle analysis lends insight into the effects of a trim follow-up on the pilot workload required to stabilize in hover.

The feedback loop structure required to stabilize a conventional rotorcraft in hover was shown in Figure 2, and is repeated in Figure 5, with the addition of a trim follow-up. As in the previous section, degraded visual cueing and/or divided attention is simulated by the lack of attitude stabilization on the part of the pilot\*, and the required attitude loop closure is assumed to be accomplished by the stability augmentation system (i.e., the pilot is provided with a basic Attitude Command SAS). The trim follow-up mechanization is shown as the sum of an integral path ( $K_I/s$ ) and a proportional path ( $K_\delta$ ). The ratio of the output of the trim follow-up to the pilot's command is given as,

$$\theta_c/\delta_p = K_\delta(s+K_I/K_\delta)/s$$

Hence, the trim follow-up shows up as a lag-lead network with a free  $s$  at the origin and a zero at  $K_I/K_\delta$ . The effect of the ratio of  $K_I/K_\delta$  on the Response-Type is summarized below.

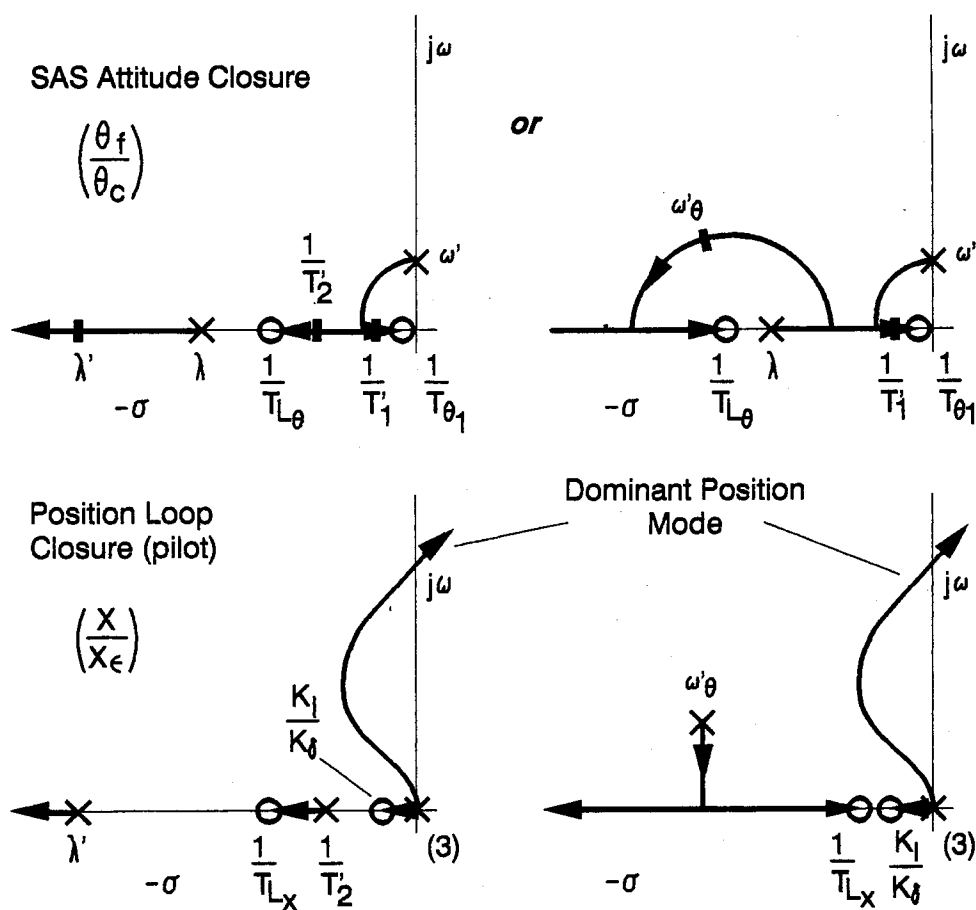
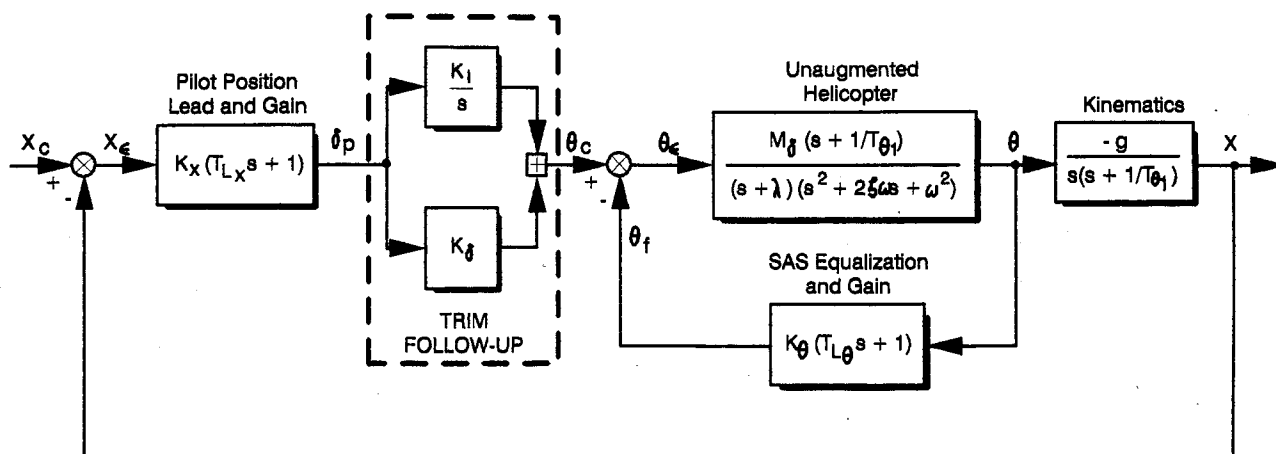
- The Response-Type is ACAH when  $K_I/K_\delta$  is less than  $-X_u$ . (The reason for this is discussed in the previous subsection). Practically speaking, the workload relief associated with an ACAH Response-Type would probably be attained if  $K_I/K_\delta \leq .05$  when  $-X_u \leq .05$ .
- The Response-Type is pure RCAH when  $K_I/K_\delta$  is large (greater than 0.30).
- There is a "grey area" when  $-X_u \leq K_I/K_\delta \leq 0.30$ .

Ideally, an experiment should be conducted to determine where in the grey area the boundary should be drawn. However, some insight can be gained by analyzing the effect of a systematic variation of  $K_I/K_\delta$  on the pilot's position loop closure. The generic attitude and position loop closures are shown in Figure 5. Some observations from these closures are summarized below.

- The basic unaugmented helicopter is assumed to be represented by the conventional hover cubic.

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\*In order to provide adequate stabilization, the pilot must close the attitude loop at 2 to 3 rad/sec, which requires continuous attention to attitude and good attitude cues. Once an attitude SAS is provided to the pilot, it is difficult for him to close a tight attitude loop to improve the basic SAS because of gain margin limiting problems discussed in Supporting Data for Paragraph 3.3.2.1. Therefore, the assumption that the pilot will not close an attitude loop should be a good one.



Notes:

3 poles at origin are due to 1)  $\frac{-g}{s(s + 1/T_{\theta 1})}$  and 2)  $\frac{s + K_I/K_\delta}{s}$

$1/T_1$ , and  $1/T_{\theta 1}$  dipole not shown in position closure for clarity

Figure 5(3.2.7). Pilot-Vehicle Loop Closure Characteristics for Hover

- The attitude stability augmentation system (SAS) can take on two characteristics depending on the relative position of the equalization,  $T_{L\theta}$ , and characteristic root,  $\lambda$ . These are shown by the two root locus plots on the top row of Figure 5. The right plot is representative of the Reference 170 ADOCS experimental helicopter.
- The position loop closure by the pilot has the same fundamental characteristics for each of the above attitude SAS types. This is shown by the bottom two root locus plots in Figure 5.
- The pilot position loop lead  $T_{Lx}$  was set at 4 sec. This was based on a calculation of the minimum value of position loop lead that is required to achieve an ideally stabilized hover, as defined by a position loop damping ratio on the order of 0.50 at a path mode frequency greater than 0.70 rad/sec;  $T_{Lx}=4$  results in  $\zeta_x = 0.50$  at  $\omega_x = 0.70$  rad/sec.

Unfortunately, there is currently no data regarding the effect of required pilot lead generation in the position loop ( $T_{Lx}$ ) on pilot ratings, i.e., similar to the classical attitude control data which shows that requirement for pilot generated lead greater than one second results in Level 2 pilot ratings (see Reference 147). However, a position lead of 4 seconds seems reasonable, as it physically implies that the pilot's control activity is based four times as much on translational rate as on position. Furthermore, there is considerable experience indicating that hovering with an ACAH SAS is easily accomplished. It follows that the pilot lead required to stabilize position in the presence of an ACAH SAS (i.e.,  $T_{Lx} = 4$  sec) must not constitute a high level of workload. Having established the ground-rules for the pilot-vehicle loop closures, the physical interpretations of these closures are discussed in the following paragraphs.

The generic variation of the position mode locus as a function of the trim follow-up ( $K_I/K_\delta$ ) is shown in Figure 6. As  $K_I/K_\delta$  is increased, the maximum damping ratio of the position mode decreases. Assuming a position step command, the closed-loop pilot-vehicle system will respond as shown in Figure 7, as a function of  $K_I/K_\delta^*$ . A significant degradation in performance exists as  $K_I/K_\delta$  is increased from 0 (pure ACAH) to 0.20. This might be compensated for by the pilot, via an increase in the position loop lead,  $T_{Lx}$ , but at some expense in pilot workload, especially if the visual cueing is poor, i.e., the poor translational rate cues associated with UCE=2 would make it difficult (or impossible) to achieve a large value of  $T_{Lx}$ . However, for the sake of illustration, consider an extreme case where the pilot uses 10 times as much translational rate as position feedback (i.e.,  $T_{Lx} = 10$  sec).

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\*The time histories in Figure 7 are based on a pilot closure that would produce a path mode natural frequency of 0.7 rad/sec (boxes plotted on the loci in Figure 6). Such a path mode frequency is representative of a "precision hover."

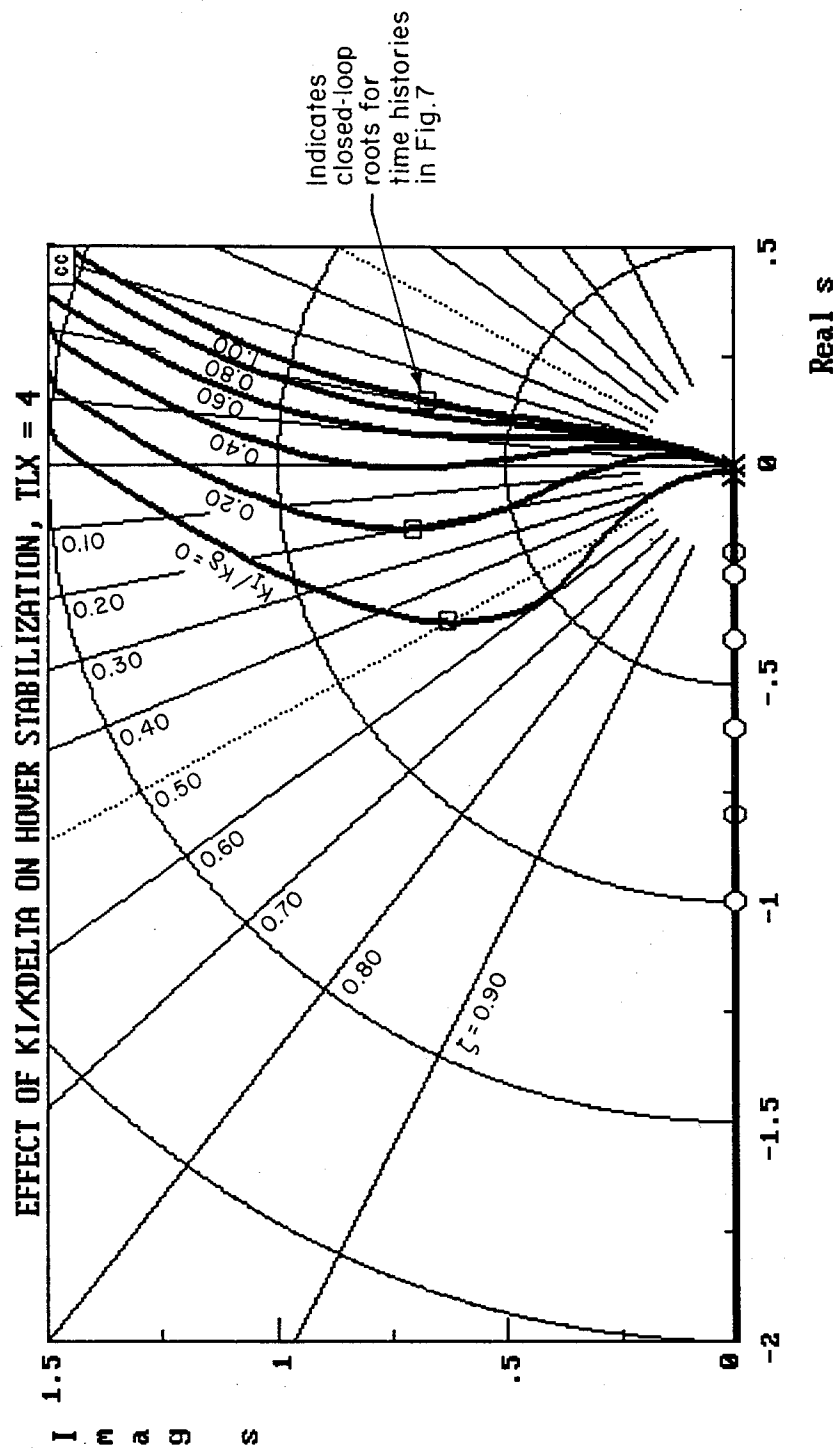


Figure 6(3.2.7). Effect of  $K_I/K_\delta$  on Position Mode Root Locus for Hover  
(Pilot Position Lead  $T_{LX} = 4$  sec)

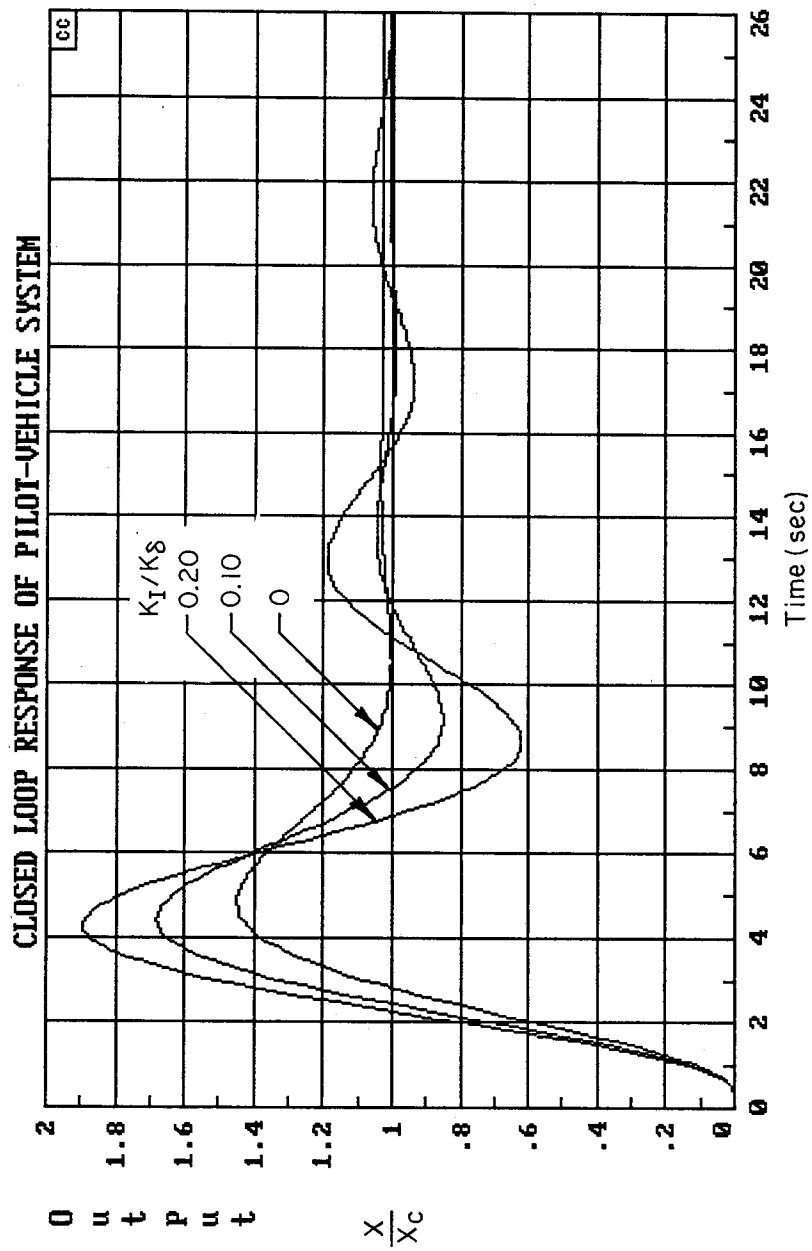


Figure 7(3.2.7). Effect of  $K_I/K_\delta$  on Time Response to Step Position Command

This results in the generic variation in the position locus shown in Figure 8. Even with this significant increase in pilot lead equalization, the baseline position-loop damping ( $\max \zeta_x \approx 0.6$ ) cannot be achieved once  $K_I/K_\delta$  is greater than about 0.15. This indicates that the pilot may not be able to compensate for the trim follow-up as it becomes large enough to trim at a practical rate (e.g., order of 0.2 1/sec from Reference 170).

The position mode damping ratio at a path mode frequency of 0.70 rad/sec is plotted vs.  $K_I/K_\delta$  in Figure 9. Establishing a minimum value of  $K_I/K_\delta$  is difficult because there is no obvious knee in this curve. However, for the minimum pilot lead (lowest workload),  $K_I/K_\delta$  greater than 0.10 could be said to produce unacceptable position damping (i.e.,  $\zeta_x \leq 0.35$ ). Low values of trim follow-up have historically been unacceptable because of the slow trim rate which gives the impression that the controller is always moving, or conversely that the pilot must constantly adjust the control force to hold a desired attitude.\* Therefore, an acceptable region of trim follow-up would be very narrow, bounded by slow trim on one end, and poor stabilization on the other. Considering that  $K_I/K_\delta = 0.15$  seems close to the upper limit (from the above analysis), and that 0.10 is almost certainly too slow, there is a strong possibility that an experiment would reveal that a Level 1 region for UCE=2 and/or divided attention does not exist.

As noted above, the Attitude Command Response-Type is defined when a constant cockpit controller force input produces an essentially constant change in attitude. Based on this definition, the system in Figure 10a would be classified as Rate for  $K_I/K_\delta > 0$ , whereas in Figure 10b the response for  $K_I/K_\delta = 0.10$  would be classified as ACAH because the trim follow-up exactly cancels a low-frequency droop. This is a good example of why the criterion cannot be based on limiting the trim follow-up rate, but instead is more properly derived from fundamental principles as determined from closed-loop pilot-vehicle analysis.

The analysis presented above shows that a trim follow-up can eliminate certain necessary features of the Attitude Command Response-Type. The analysis is not able to predict if an allowable range of  $K_I/K_\delta$  exists for UCE=2 and/or divided attention. Therefore, until experimental data can be obtained to investigate the "grey area" of  $K_I/K_\delta$ , an approach which will guarantee the necessary workload reduction achievable with ACAH is to disallow any additional modes (such as trim follow-ups), if they result in more than two mid-to-low frequency integrations between the controller input (force), and the rotorcraft position.

#### d. Guidance for Application

It is emphasized that for most flight conditions, an ACAH Response-Type with a trim follow-up is a highly desirable configuration. This

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\*The time required to trim a step attitude command (i.e., to return the stick 90% of the way to center) is approximately equal to  $2/K_I/K_\delta$ .

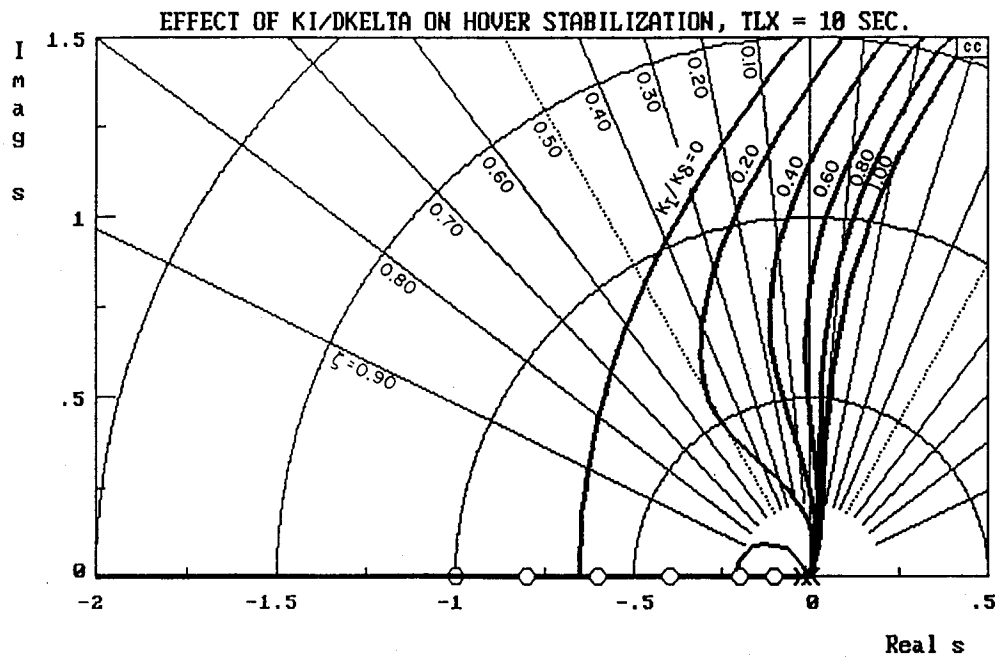


Figure 8(3.2.7). Effect of  $K_I/K_\delta$  on Position Mode Root Locus for Hover (Pilot Position Lead  $T_{LX} = 10$  sec)

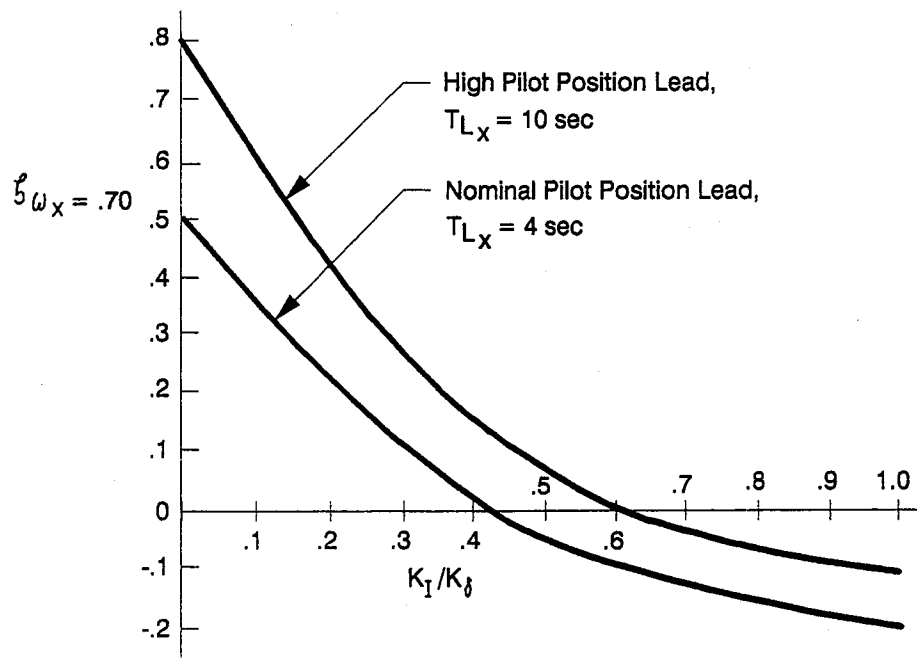
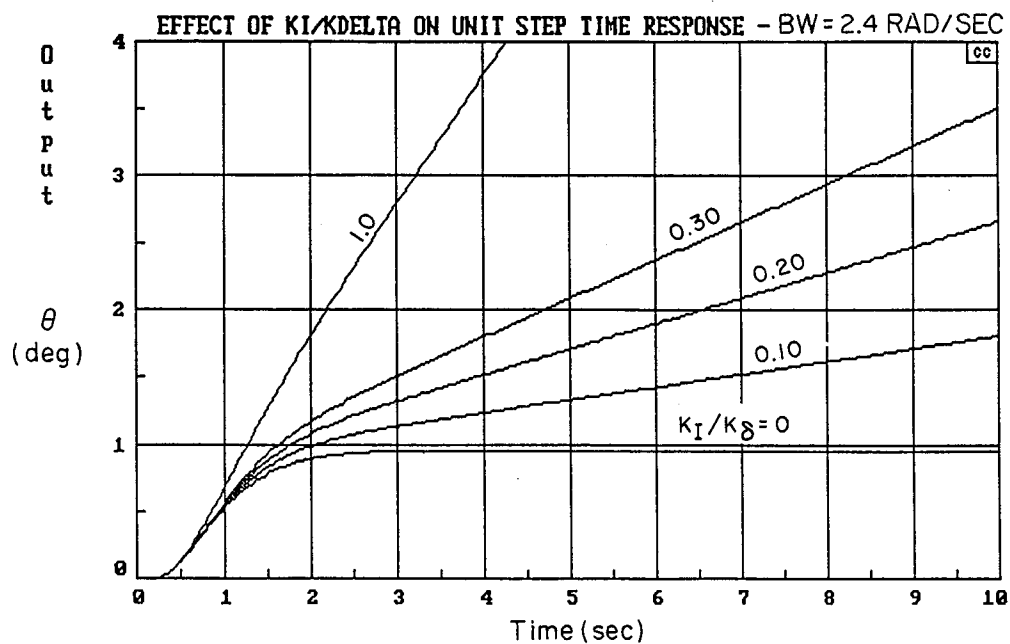
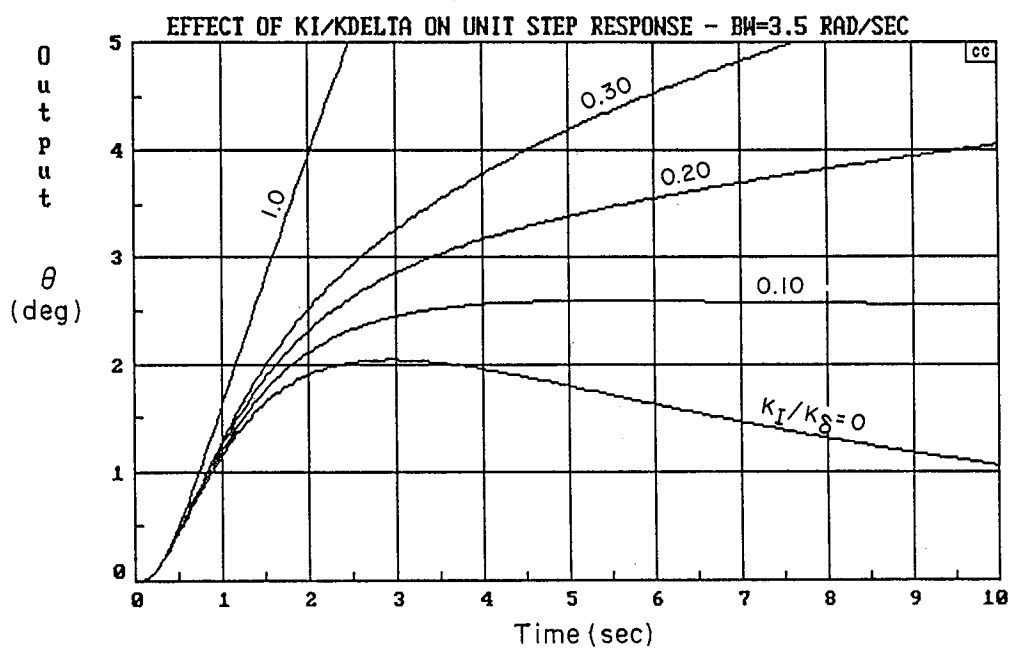


Figure 9(3.2.7). Variation in Position-Loop Damping with  $K_I/K_\delta$



a) ADOCS (Ref 170) SAS



b) ACAH SAS With Low Frequency Droop

Figure 10(3.2.7). Response of Pitch Attitude to Step Controller Input



criterion should be interpreted as a requirement to be able to turn off the trim follow-up in conditions of degraded visual cueing or high divided attention workload.

The use of "ground referenced" translational acceleration indicates that body axis accelerations are not appropriate, a fact that is obvious from the above analysis. However, if the attitudes are small, the body-axis acceleration should be a reasonable approximation to ground referenced acceleration, i.e., acceleration in an earth-fixed coordinate system.

The requirement includes a minimum time of 12 seconds to establish the nature of the response. This was done to insure that the time history of attitude is not simply at an inflection point of a low-frequency oscillation. A value of 12 seconds was selected as simply double the time allowed to establish a constant value. Longer times may be required to establish the true nature of the response (when  $\omega'$  is very small). This is especially important in making a distinction between low-frequency droop and a low-frequency oscillation. The requirement allows for some droop by noting that a variation from a constant acceleration is acceptable, if it is "asymptotically decreasing towards a constant." This clause also allows velocity command systems (including TRC) to pass as an Attitude Response Type. In some cases such a system may not meet all the requirements of a TRC in Paragraph 3.3.12, but does satisfy the fundamental requirement for an Attitude Response-Type (no more than two integrations between control force and rotorcraft position). The acceleration response of a TRC system that just barely meets the requirements of Paragraph 3.3.12 is given in Figure 4, where it is seen to be asymptotically approaching zero. Also, velocity command systems which are not inertially referenced will meet this requirement, but not Paragraph 3.3.12. If this clause is a central issue for compliance, a time significantly longer than 12 seconds may be required to establish if the response is asymptotically approaching a constant, or a low frequency oscillation.

The term "essentially constant" is used in this requirement to avoid setting numerical limits which invariably are "fooled." Experience has shown that it is better to trust engineering judgement in such cases, especially since concurrence with technical personnel from the procuring activity and the government is necessary.

Force inputs are specified (as opposed to position) to insure that the feel systems and trim systems are included in the characterization of the Response-Type. Examples of feel systems currently in use that could affect the determination of the Response-Type are trim follow-up systems (discussed in the previous subsection) and the Flight Path Stabilization System (FPS) used on the Sikorsky UH-60. The FPS is mechanized so that rotorcraft attitude and airspeed are fed directly to the stick, thereby having a dramatic effect on the aircraft and stick responses to a force input. This, in effect, amounts to a parallel mechanization of a stability augmentation system, a technique which is usually reserved for autopilots. A similar system has been mechanized on the McDonnell Douglas Helicopter AH-64. It is reported to be well

accepted by pilots, and therefore there seems to be no reason to regulate against it.

The Advanced Digital Optical Control System (ADOCS) helicopter is an example of a basic Attitude Command Response-Type with a trim follow-up (see Reference 170), and experience with this aircraft is useful to illustrate the intent of the specification. This research aircraft started out as an ACAH SCAS with a very low frequency trim follow-up. As discussed in Reference 170, and shown in Figure 11, this was "optimized" by increasing the rate of the trim follow-up (increasing  $K_I/K_\delta$ ). While the reasons for increasing the speed of the trim follow-up are not discussed in Reference 170, slow trim follow-ups have not been well accepted in the past because pilots need to be able to "set it and forget it." A slowly changing automatic trim requires constant adjustment of control pressure following a change in the reference attitude. Potential solutions are to provide a separate, and sufficiently rapid, trim control on the stick (required for ACAH Response-Types in this specification) or to modify the Response-Type to RCAH which is inherently self trimming. In the context of this specification, the researchers in Reference 170 elected the latter solution, which caused a reclassification of Response-Type from ACAH to Rate with Attitude Hold. A comparison of the time responses of the "original" and "optimized" systems, shown in Figure 12, clearly indicates a transition from attitude to rate. It is important to note that all of the tasks were fully attended, and accomplished in good visual cueing (UCE=1),\* so that an ACAH Response-Type was not necessary, or even desirable.

The data from the NRC UCE experiments (Reference 187) indicate that even pure Attitude Command is marginal in UCE=2, and on that basis, it seems unwise to compromise the stabilization potential of ACAH by allowing trim follow-ups. Finally, it is important to note that a slow trim follow-up is disallowed only for an ACAH Response-Type, and is not prohibited for Rate or RCAH. Trim follow-ups can become considerably more complex than illustrated here. For example, the production UH-60 Blackhawk (not ADOCS) has a flight path trim function which moves the stick in response to changes in attitude and airspeed. This would not be disallowed by this specification because the UH-60 is a Rate Response-Type.

e. Related Previous Requirements

None.

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\*A few flights were made with night vision goggles using daylight training filters. The Reference 187 tests showed that night vision goggles produce UCE=1 in conditions of good illumination (e.g., full moon, no overcast).

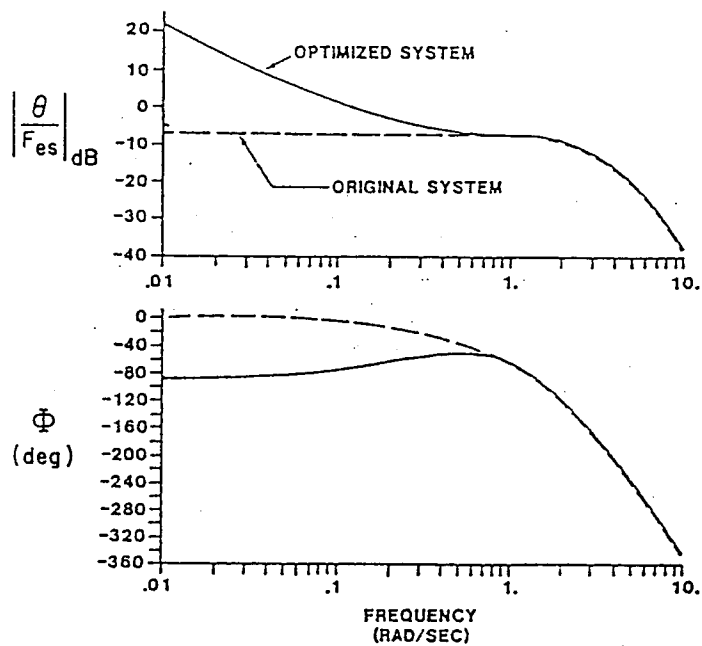


Figure 11(3.2.7). ADOCS Longitudinal Axis Command/Response Characteristics (Taken from Reference 170)

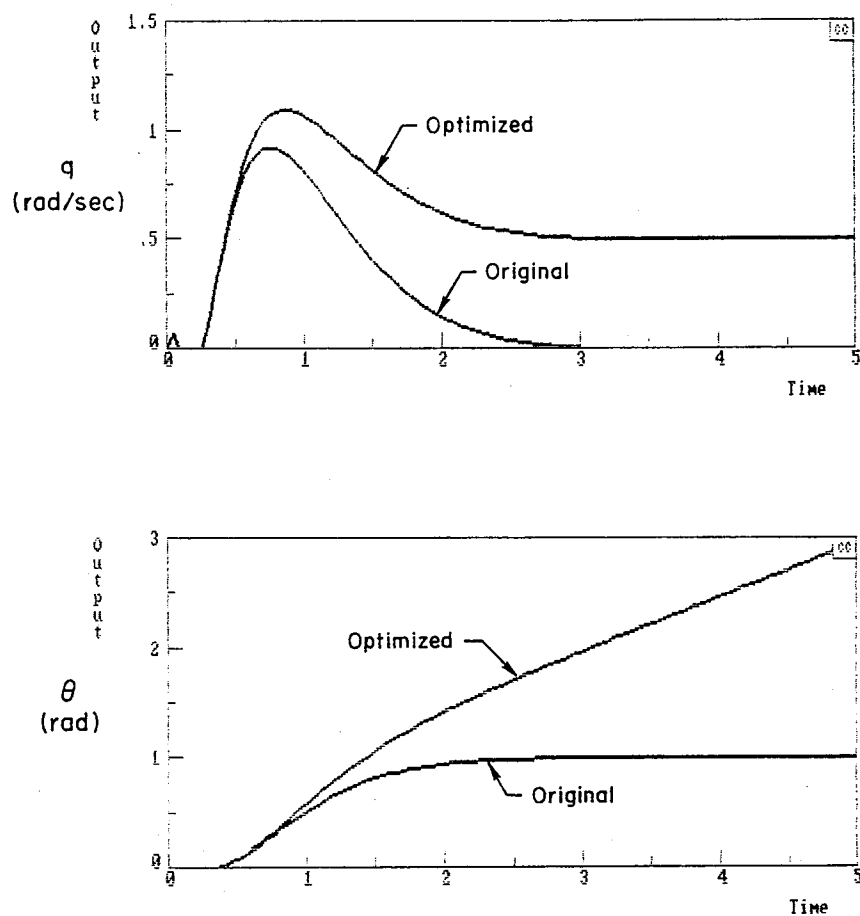


Figure 12(3.2.7). Time Responses to Step Input for "Original" and "Optimized" Pitch Attitude SCAS from Reference 170 (ADOCS)

a. Statement of Requirement

3.2.8 Character of Translational Rate Response-Type. If Translational Rate Command is specified as a required Response-Type in Paragraph 3.2.2, constant pitch and roll controller force and deflection inputs shall produce a proportional steady translational rate, with respect to the earth, in the appropriate direction.

b. Rationale for Requirement

Certain Mission-Task-Elements, in combination with severely degraded Usable Cue Environments (UCE=3), will require Translational Rate Response-Type (TRC) for Level 1 handling qualities (Paragraph 3.2.2). This paragraph defines TRC in the classical sense, i.e., the "pitch" and "roll" controllers become longitudinal and lateral translational rate controllers. The translational rate is defined with respect to the earth to preserve the tactile cue on velocity, i.e., groundspeed is proportional to controller force or deflection. This cue is a primary attribute of TRC in conditions of degraded visual cueing.

c. Supporting Data

A primary advantage of Translational Rate Command is the tactile cue of translational rate when visual cueing is poor (a more detailed discussion is given in Paragraph 3.2.2). The relevant details of TRC Response-Types are discussed in Paragraph 3.3.12.

d. Guidance for Application

See Paragraph 3.3.12.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.2.9 Character of Altitude Rate Response-Types. The rotorcraft is defined as having an Altitude Rate Response-Type if a constant deflection (force if an isometric controller is used) of the vertical axis controller from trim produces a constant steady-state vertical velocity. It must be possible for the pilot to null the cockpit controller force at any achievable vertical rate.

b. Rationale for Requirement

The Altitude Rate Response-Type category is intended to include rotorcraft which are either unaugmented or rate damped in the vertical axis. As noted in the footnote of Table 1(3.2), this Response-Type is required if Rate Command/Height Hold (RCHH) is not specified. It is difficult to imagine a rotorcraft which does not reach a steady vertical velocity following a constant vertical axis controller input. Nonetheless, it seems prudent to include this requirement to preclude a particular controller or governor configuration which might defeat the natural heave damping of the rotor. In addition, this requirement provides a logical location in the specification to require vertical axis trim. For conventional collective mechanizations, such a trim is inherent to the controller. However, for a four axis sidestick with limited vertical motion, vertical rates could be programmed as a function of applied vertical force, which would require a separate trim function. The alternative mechanization, which was employed in the sidestick used in the Reference 157 flight test experiment, is to incorporate a trim follow-up. However, this has the undesirable effect of reducing the effective heave damping.

c. Supporting Data

The detailed data for vertical response characteristics are given in the Supporting Data for Paragraph 3.3.10.

d. Guidance for Application

The ability to trim out the vertical axis controller forces should be checked at the extremes of the flight envelope, including autorotation. It is reasoned that the controller should not resist full down collective during autorotational flight.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.2.9.1 Character of Altitude Rate Command With Altitude Hold. Following an altitude deviation induced by insertion and removal of an input directly into the vertical-axis actuator, the rotorcraft shall return to its original altitude without objectionable delays and with no overshoot. For Hover and Low Speed, the rotorcraft shall automatically hold altitude with respect to a flat surface for land based operations, or a rough sea for sea-based operations, with the altitude controller free. The altitude deviation may not exceed 1 m (3.3 ft) during the performance of the Mission-Task-Elements specified in Paragraph 3.1.1. This may be relaxed to 2 m (6.6 ft) if the bank angle exceeds 30 degrees. Engagement of Altitude Hold shall be obvious to the pilot through clear tactile and visual indication. The pilot shall be able to disengage Altitude Hold, change altitude and reengage Altitude Hold without removing his hands from the flight controls. This requirement shall be met for Level 1, and relaxed according to Paragraph 3.2.12 for Levels 2 and 3.

b. Rationale for Requirement

Altitude hold has been found to be necessary for operations in conditions of poor visual cueing and for divided attention operations. The performance of the altitude holding function is based on an input to the vertical axis actuator for the same reasons discussed in Paragraph 3.2.6 (especially Figure 1(3.2.6)). There it is shown that the response to the actuator is closely related to the response to external disturbances. While it would be desirable to test the response in natural turbulence, it was decided that it would be impossible to separate the input from the response without unduly complicated instrumentation and data reduction techniques.

The altitude-hold mode intended for this requirement is different from the usual altitude hold incorporated in conventional autopilots. It must be automatically selected when the vertical axis controller is in the zero position. This normally involves a detent, but could be zero force for a small displacement four axis sidestick controller, as long as the vertical axis trim requirement (Paragraph 3.2.9) can be satisfied. The specified altitude tolerances were taken directly from the demonstration maneuvers in Section 4.

c. Supporting Data

The visual cueing experiments discussed in Paragraph 3.2.2 resulted in a conclusion that an RCHH Response-Type was necessary in certain conditions. Pilot commentary indicated that the workload associated with manually engaging altitude hold would be excessive, and that even looking down at an annunciator to ensure engagement was unacceptable. The last two sentences of the requirement are based on that experience.

The words "clear tactile and visual indication" are intended to imply that the pilot can determine the status of altitude hold without taking his eyes away from the outside environment.

d. Guidance for Application

There is no known source of data to arrive at a requirement for the time constant of the RCHH response characteristics. However, based on experience with approach couplers (with similar tolerances), a time constant of between 3 and 5 seconds would seem reasonable. That is, it should take about 12 seconds to return to within 5% of the reference altitude, from an initial condition offset (achieved via an input to the actuator).

Depending on the details of the mechanization, collective position may or may not reflect all or part of the vertical axis servo activity. If a collective detent is utilized to engage altitude hold, it is likely that a series servo would be employed (collective does not move unless the pilot moves it). However, if a parallel mechanization (collective moves in response to servo activity) is employed, it must not interfere with the ability of the pilot to find the detent, and should not reflect any mid-to-high frequency feedbacks used to augment heave damping.

e. Related Previous Requirements

None.

a. Statement of Requirements

3.2.10 Character of Yaw Response to Lateral Controller

3.2.10.1 Turn Coordination. For Low Speed flight, during banked turns with any available Heading Hold modes disengaged, the rotorcraft heading response to lateral controller inputs shall remain sufficiently aligned with the direction of flight so that it will not be objectionable to the pilot. Complex coordination of the yaw and roll controls shall not be required.

3.2.10.2 Rate Command with Direction Hold. For Hover, the yaw controller inputs required to maintain constant heading during rolling maneuvers shall not be objectionably large or complex.

b. Rationale for Requirement

The supporting data and rationale for these two requirements are presented in one section of the BIUG since they are very closely related. A distinction between Low Speed and Hover is made here because the piloting technique to control heading is different for these two flight regimes (see Paragraph 2.6 for the formal definition of Low Speed and Hover). For flight very near hover (groundspeed less than 15 knots in this specification), lateral translation is usually accomplished at constant heading, while at low speed, pilots typically prefer to make coordinated turns.

The requirement is qualitative because of a lack of supporting data. A recommended turn-coordination criterion is given below in "Supporting Data" and "Guidance for Application." This criterion is equally applicable for forward flight. Unfortunately, similar guidance is not available for the heading hold mode in terms of the heading response (or lack thereof) to lateral controller inputs. However, there are quantitative requirements which specify the response of heading to inputs to the directional controller actuator, to simulate external disturbances, see Paragraphs 3.2.6 and 3.3.7. Since crosscoupling of roll into yaw is basically an external disturbance, meeting these paragraphs may insure adequate decoupling from roll to yaw.

c. Supporting Data

The NASA Ames CH-47 variable stability flight tests reported in Reference 185 showed that automatic switching from turn coordination (TC) to Rate Command/Direction Hold (RCDH) at an airspeed of 35 knots was unsatisfactory during jinking and slalom maneuvers (HQR 4.5). The pilot complaints centered about a lack of stiffness in yaw below 35 knots, switching in and out of turn coordination, and insufficient awareness when the computer switched between the RCDH and TC modes.



The requirements for TC and RCDH are qualitative because there is no directly applicable data upon which to base a quantitative requirement. The sideslip limits used for forward flight (Paragraph 3.4.6.2) probably do not apply to low speed. However, the aileron/rudder control shaping criterion used in MIL Standard 1797 (Reference 190) would be expected to provide reasonable guidance. The rationale is that the shaping and magnitude of the pedal inputs required to keep the aircraft heading aligned with the velocity vector are primarily a function of the human operator, and should not depend on the aircraft type. This is verified by the pilot rating data which is plotted on the  $\mu$  parameter boundaries in Figure 1 and the normalized pedal boundaries in Figure 2. Guidance for application of this criterion is given in the following subsection.

#### d. Guidance for Application

As noted in Reference 185, continuity in turn-coordination characteristics at all speeds is desirable. This suggests the need for an implementation of yaw-axis augmentation and heading-hold systems that provide continuous control characteristics, avoiding change between RCDH and TC without direct pilot control. For example, RCDH might be activated by releasing pedal switches. On the other hand, it might be possible to blend in the RCDH mode at low groundspeeds in such a way that seems more natural to the pilot than the 35 knot airspeed switch used in Reference 185. Automatic blending would, of course, be more desirable from a pilot workload point-of-view. The requirements in their present form are intended to insure that RCDH is available for hover, and TC is available for low speed. The method for switching between these modes is left to the manufacturer. Based on the Reference 185 results, it would seem that a switch, activated by releasing the pedals to activate RCDH, would be the lowest-risk approach, unless simulator and flight test work are allocated to optimizing the blending between RCDH and TC.

A requirement based on the amount and shaping of the pedal required to coordinate turns is recommended as a quantitative metric to support this qualitative criterion. The recommended criterion is a slightly modified form of the criterion in the fixed wing specification, Reference 190, and is given as follows. The yaw control crossfeed required to maintain zero sideslip should be within the limits shown in Figure 1 for Levels 1 and 2. In addition, for values of the ratio of yawing-to-rolling acceleration due to roll controller inputs ( $|N_{\delta'_{as}}/L_{\delta'_{as}}|$ ) less than 0.07, the normalized pedal must be within the limits of Table 1. The rules for application of this criterion are summarized as follows:

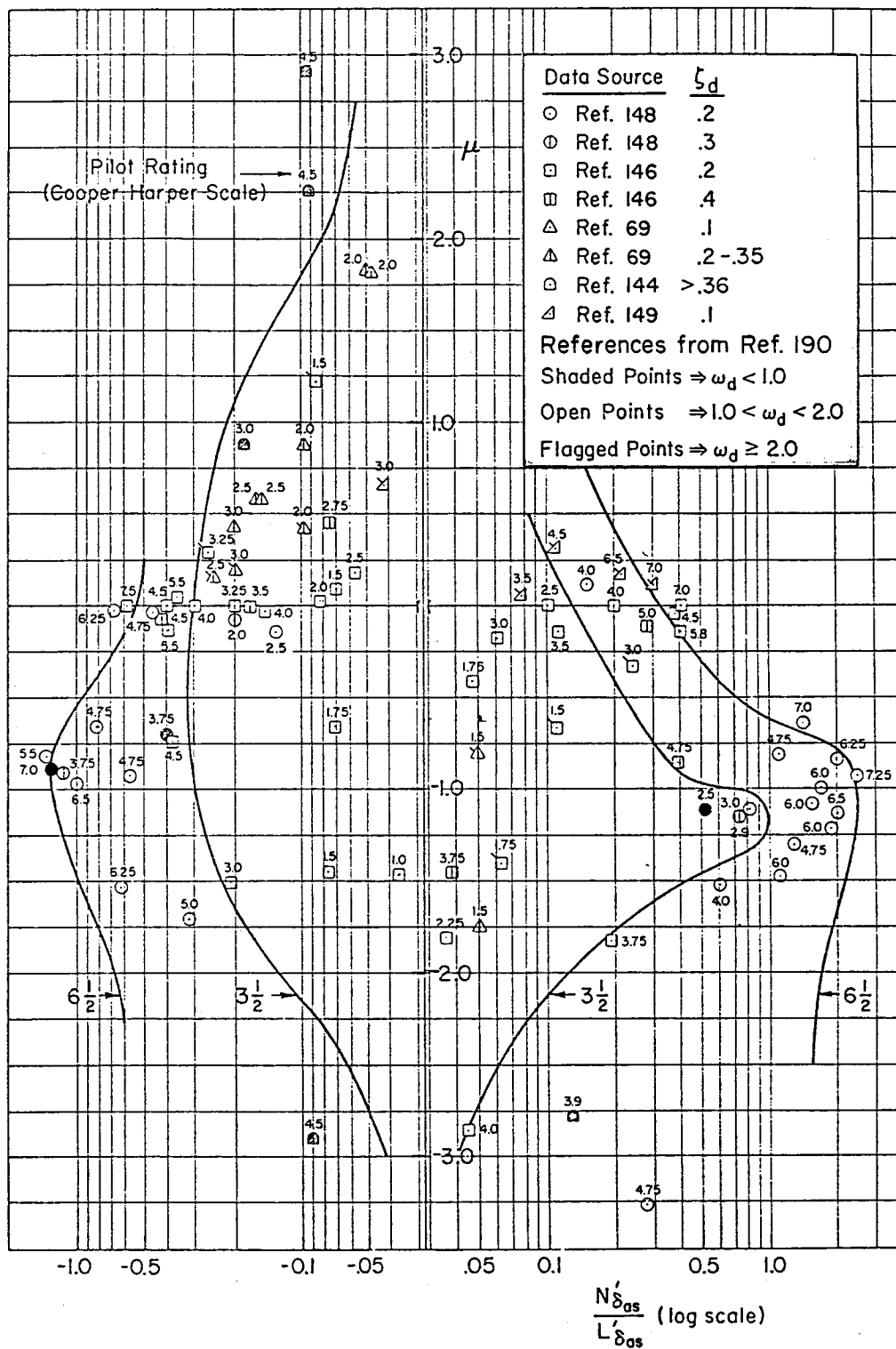


Figure 1(3.2.10). Pilot Rating Correlation on Recommended Boundaries

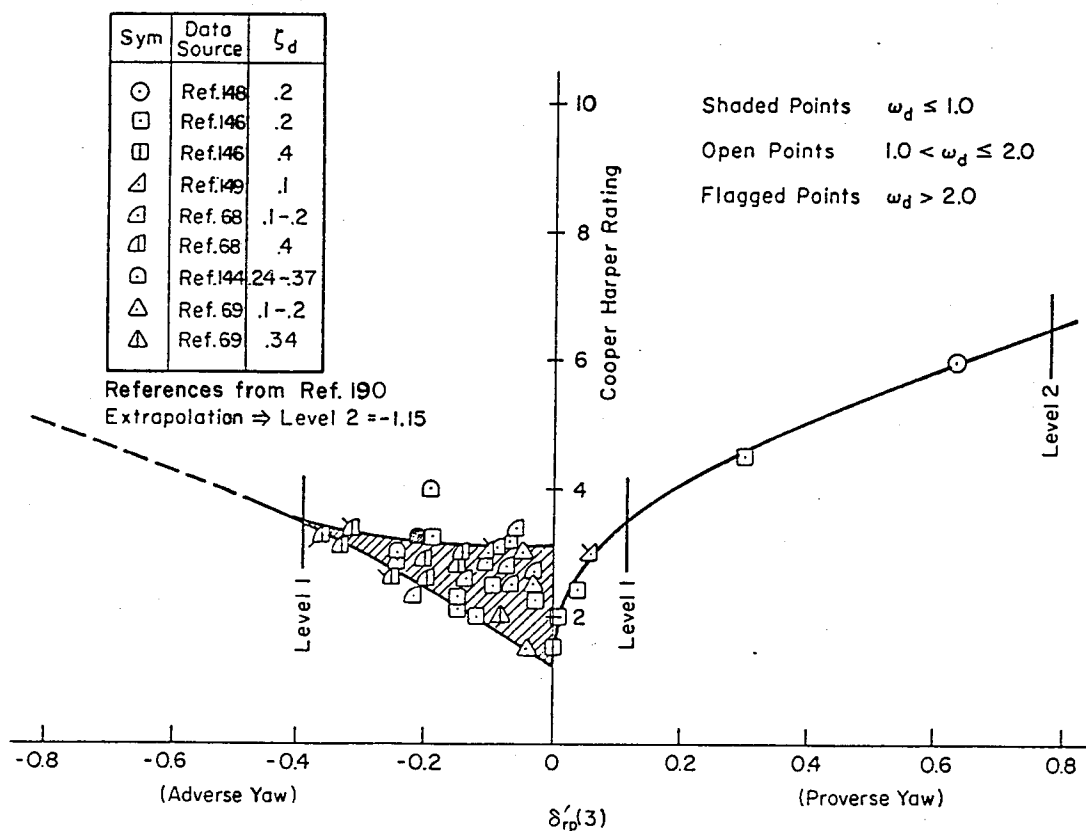


Figure 2(3.2.10). Pilot Rating Correlations When  $|N'_{\delta_{as}}/L'_{\delta_{as}}|$  is Small

TABLE 1(3.2.10). LIMITS ON  $\delta'_{rp}(3)$  FOR  $|N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$

LEVEL	ADVERSE YAW	PROVERSE YAW
1	-0.39	0.11
2	-1.15	0.78

- 1) If  $|N'_{\delta_{as}}/L'_{\delta_{as}}| < 0.03$ , skip to Step 6.
- 2) Formulate  $Y_{CF}$  by taking the ratio of the  $\beta/\delta_{as}$  and the  $\beta/\delta_{rp}$  transfer functions. For augmented rotorcraft the transfer functions must include the effects of augmentation:

$$Y_{CF} = \left( N_{\delta_{as}}^{\beta} / N_{\delta_{rp}}^{\beta} \right)_{aug}$$

- 3) Remove all roots greater than 6 rad/sec in pairs, keeping the order of  $Y_{CF}$  constant. Roots above 6 rad/sec that do not occur in pairs are left unmodified. Set the high-frequency gain of  $Y_{CF}$  equal to unity.
- 4) Calculate  $\delta_{rp}(3)$  from the time response of  $Y_{CF}$  (as modified by Step 3) to a unit step input, i.e.,  $\delta_{rp}(3) = \mathcal{L}^{-1}[(1/s)Y_{CF}(s)]$  evaluated at  $t = 3$  sec.
- 5) Calculate  $\mu$  as:  $\mu = \delta_{rp}(3) - 1$  and plot on Figure 1.
- 6) If  $|N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$ , calculate the normalized rudder required,  $\delta'_{rp}(3)$ , as follows:
  - Calculate the magnitude of the time response of  $Y_{CF}$  (from Step 2) to a unit step input at  $t = 3$  sec.
  - Multiply the result by  $N'_{\delta_{rp}}/L'_{\delta_{as}}$ , i.e.,

$$\delta'_{rp}(3) = Y_{CF}(3) \frac{N'_{\delta_{rp}}}{L'_{\delta_{as}}}$$

Compare  $\delta'_{rp}(3)$  with Table 1.

- 7) If  $0.03 \leq |N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$ , utilize the most conservative result from Steps 5 and 6.
- 8) If the configuration does not meet the requirements, see Table 2 to determine expected piloting problems.

This criterion is more easily applied during the design phase when transfer functions of the rotorcraft are available. However, if  $|N'_{\delta_{as}}/L'_{\delta_{as}}| \leq 0.07$  (this will be the case for most rotorcraft, since  $N'_{\delta_{as}}$  is usually very small), the normalized rudder to be used in Table 1 can be approximated as follows:

$$\delta'_{rp}(3) = \frac{\delta_{rp}(3)/\delta_{rp}(\max)}{\delta_{as}/\delta_{as}(\max)}$$

Using this approximation, the flight test procedure is to conduct a series of coordinated rolls and measure the pilot's lateral and directional control inputs 3 seconds after a step lateral controller input.

#### e. Related Previous Requirement

As noted in Guidance for Application.

TABLE 2(3.2.10). PHYSICAL INTERPRETATION OF  $\mu$ 

VALUE OF RUDDER SHAPING PARAMETER	ROLL-YAW CROSS-COUPLING CHARACTERISTICS
$\mu > 0$	$N'_{\delta^{as}}$ and $N'_p$ are additive, indicating that the cross-coupling effects increase with time after a lateral controller input.
$\mu = 0$	$N'_p = g/U_0$ , indicating that all roll-yaw cross-coupling is due to $N'_{\delta^{as}}$ . The lateral cyclic to pedal crossfeed is therefore a pure gain.
$-1 < \mu < 0$	$N'_{\delta^{as}}$ and $N'_p$ are opposing. Initial cross-coupling induced by $N'_{\delta^{as}}$ is reduced by $N'_p$ as the roll rate builds up. Exact cancellation takes place when $\mu = -1$ , resulting in a zero directional controller requirement for steady rolling.
$\mu \ll -1$	Low-frequency and high-frequency cross-coupling effects are of opposite sign, indicating a need for complex directional controller reversals for coordination. If the directional controller is free, the nose will appear to oscillate during turn entry and exit.

a. Statement of Requirement

3.2.11 Limits on Nonspecified Response-Types. It may be desirable, or even necessary, to incorporate Response-Types that are not explicitly defined in this specification. Examples of such Response-Types are Airspeed Hold, Linear Acceleration Command with Velocity Hold, and hybrid responses such as ACAH for small attitudes which blend to Rate for larger commands or attitudes. It is required that these Response-Types meet their stated objectives (e.g., Airspeed Hold systems must hold airspeed), and the requirements of this specification.

b. Rationale for Requirement

Recognizing that the possibilities for different and innovative control laws are essentially infinite, this requirement has been included to provide a format to include new and novel augmentation schemes. However, if such a system can be shown to fall under the category of one of the Response-Types in this specification, it should not be called out as a "Nonspecified Response-Type" as a means to deviate from the required characteristics.

The only requirement on this category of response is that the objective of the augmentation be defined and achieved. It will be up to the manufacturer to prove that this Response-Type is an adequate replacement for one of the Response-Types called out in the specification, using piloted simulation and flight testing.

Some augmentation schemes involve a tradeoff between outer loop performance and excessive inner loop (attitude) abruptness. This is discussed, in detail, for the Translational Rate Command (TRC) Response-Type in Paragraph 3.3.12. Piloted simulations, conducted in the Reference 4 study, indicated that pilots object to rapid attitude motions that are not directly commanded through the cockpit controller. However, it is impractical to attempt to place qualitative or quantitative limits on responses which are not yet specified. Hence, this shortcoming is pointed out here as an advisory note as opposed to an integral part of the requirement. The crispness of the attitude response is insured by requiring that the requirements of the specification are satisfied.

It is required that the Response-Type meet its stated objective. This has the effect of forcing the manufacturer to state the specific objectives of the SCAS, and to focus attention on whether or not that objective is actually accomplished. The wide variety of augmentation possibilities precludes incorporation of tolerances (e.g., allowable airspeed error for airspeed-hold) in the specification, and it is intended that these will be mutually agreed upon by the manufacturer and the procuring activity.

c. Supporting Data

The supporting data for this requirement is in the form of pilot comments for TRC augmentation contained in Reference 4 (see "Supporting Data" for Paragraph 3.3.12). Those comments indicated that even though excellent performance could be obtained with high gain TRC systems, the required abrupt and uncommanded attitude motions were objectionable. These results have been generalized as applicable to all unspecified Response-Types that may be proposed.

d. Guidance for Application

The desire to achieve excellent outer-loop performance while simultaneously meeting the requirement to avoid abrupt and uncommanded attitude motions can be fundamentally in conflict. It will be up to the evaluation pilots to decide what is an acceptable level of abruptness, since little supporting data is available. The supporting analysis and data supplied for Paragraph 3.3.12 should be used as a guide in the evaluation of the tradeoff between attitude abruptness and outer-loop performance.

e. Related Previous Requirement

None.

a. Statement of Requirement

3.2.12 Requirements for Inputs to Control Actuator. The requirements to check for adequate disturbance rejection via inputs directly into the control actuator (Paragraphs 3.2.6, 3.2.9.1, 3.3.2.3, 3.3.7, and 3.4.10) are waived for Levels 2 and 3. This is to allow the use of control input shaping to achieve the necessary command-response relationship for backup flight control systems.

b. Rationale for Requirement

In the backup data and analysis supporting the requirement for the character of Attitude Hold Response-Types, a distinction was made between the response to commands and to disturbances, see in particular Figure 1(3.2.6). As illustrated in Figure 1(3.2.6), the response to external disturbances, and to test inputs to the control surface actuator (swash plate in conventional rotorcraft), are essentially identical. On this basis, four requirements have been included which require measurements of rotorcraft response to inputs to the actuator. Two of these requirements are to test the mid and low frequency characteristics of hold functions, i.e.,

- attitude and heading hold functions (Paragraph 3.2.6).
- altitude hold function (Paragraph 3.2.9.1).

The remaining two requirements are included to insure that the bandwidth of the regulation against external disturbances is adequate. These are,

- the short-term pitch, roll, and yaw responses to disturbance inputs for Hover and Low Speed (Paragraphs 3.3.2.3 and 3.3.7).
- the short-term pitch, roll, and yaw responses to disturbance inputs for Forward Flight (Paragraph 3.4.10).

This paragraph (3.2.12) is included as a relaxation which eliminates the necessity to insure that the above requirements for disturbance regulation are met for Levels 2 and 3. The rationale is that disturbance rejection is not necessary to achieve flying qualities consistent with Levels 2 and 3 (i.e., HQRs of 4 and worse). In practice, such a relaxation allows the use of feedforward and input shaping ( $G_a$  and  $G_i$  in Figure 1(3.2.6)), without feedback ( $G_f = 0$ ), to achieve the necessary response to commands for Levels 2 and 3.



c. Supporting Data

The Reference 45 XV-15 experiments, conducted by the Army and NASA, indicated that removal of the feedbacks of the command augmentation flight control system allowed Level 2 flying qualities, even in turbulence. Unfortunately, these pilot ratings are not reported in Reference 45, and we must resort to informal discussions with the researchers as support for this result.

The development of the flight control system discussed in Reference 45 stresses the value of input shaping (called forward loop shaping in Reference 45). In fact, the philosophy of that control system design was to provide most of the required bandwidth with input shaping ( $G_1$ ), which allows the feedback gains to be low, and provides Level 1 command/response bandwidth in the event of a failure of one or more elements of the feedback loop (e.g., rate gyros). The fact that the ratings degraded to Level 2 in turbulence is evidence that the disturbance rejection function of the augmentation is significant. This fact forms the basis for requiring that, for Level 1, the bandwidth requirements be met for disturbance inputs (inputs to actuator), as well as the response to commands.

d. Guidance for Application

This requirement should be taken into account during the development of the prototype aircraft so that a provision is made to allow disturbance inputs directly into the control actuator.

For non-fly-by-wire/light rotorcraft, input and feedforward shaping is not possible so that the bandwidth of the response to commands and disturbances will be identical. This is also true for highly augmented aircraft without input shaping ( $G_1 = 1$ ). The specification accounts for this by allowing demonstration by analysis, i.e., it is easy to show that the bandwidths to commands and disturbances are equal if  $G_1 = 1$ .

e. Related Previous Requirement

None.

a. Statement of Requirement

3.2.13 Transition Between Airborne and Ground Operations. There shall be no tendency for uncommanded, divergent motions of any primary flight control surface when the rotorcraft is in contact with any potential landing platform. This requirement is aimed specifically at integrators in the flight control system that must be turned off when rotorcraft motion is constrained by contact with a fixed object.

b. Rationale for Requirement

The use of integrators in the flight control system can cause large uncommanded aircraft motions immediately after liftoff. A requirement to limit such motions would necessarily involve the details of the allowable excursions and/or the cockpit control activity required to limit the undesirable motions. Such a criterion would require detailed data which is not available, so it was felt to be more effective to limit the cause of the problem. Physically, uncommanded attitude excursions immediately following liftoff result from uncommanded control surface motions, which occur while the rotorcraft is in contact with the ground or deck. Therefore, the specification is written to disallow such control surface motions, i.e., the pilot is assured that the control position for trim does not change while the rotorcraft is constrained.

Another source of uncommanded motions immediately following liftoff, and in low hover, is the potentially asymmetric plume from dual rotors mounted side-by-side (e.g., see discussion of the XV-15 in Reference 45). This requirement is not intended to deal with such effects, although they are serious enough to cause the pilot ratings to be Level 2. As discussed in Reference 45, the asymmetric plume ("fountain effect") appears to the pilot as interaxis coupling which is addressed elsewhere in this specification.

c. Supporting Data

The need for this requirement is intuitively obvious, and supporting data is not required for justification, although the following experience does lend some insight to the issue. The variable stability Bell 205A experiments reported in Reference 157 utilized an integrator in the input path shaping to achieve an RCAF Response-Type. This integrator tended to drift on the ground resulting in serious control problems immediately following liftoff, frequently requiring the safety pilot to take over. A control position indicator was added to the cockpit to allow the evaluation pilot to center the actuators before liftoff. This was acceptable in a research environment (i.e., a safety pilot with separate controls), but was unanimously considered to be unacceptable for operational use by all participating pilots.

d. Guidance for Application

The use of the word "divergent" in the requirement is to make a distinction between control surface motions due to stabilizing feedbacks (such as might occur during a slope landing with one skid on the ground), and a monotonically increasing actuator (such as due to a drifting integrator). In testing for compliance with this requirement, a questionable result (is the surface diverging?) can be resolved by noting if control problems are encountered immediately following lift-off, or if the trim control positions change while in a constrained hover.

e. Related Previous Requirement

None.

### 3.3 HOVER AND LOW SPEED

This section of the specification is intended to insure good flying qualities for operations conducted in the near-earth environment. The term "near earth" is meant to include "operations sufficiently close to the ground or fixed objects on the ground, or near water and in the vicinity of ships or oil derricks, etc., that flying is primarily accomplished with reference to outside objects" (Paragraph 2.3).

Separate attitude response criteria are provided for small, moderate, and large-amplitude attitude changes. The attitude response criteria are classified in terms of amplitude, based on the rationale that the required precision of control tends to be inversely proportional to the amplitude of the attitude motions. This tendency is a natural result of the fact that the normal accelerations and angular rates become excessive, and control servos begin to saturate, for large-amplitude high-frequency motion. On that basis, bandwidth is specified for small amplitude tracking only. At moderate amplitudes the peak angular rate that can be obtained in changing from one steady attitude to another is specified. Furthermore, the required peak angular rate is reduced as the magnitude of the attitude change is increased. It can be shown that this is equivalent to allowing a decreasing value of bandwidth for increasing amplitude.

For large attitude changes, the angular rate that must be achievable is specified. As an example, for a large-amplitude rolling maneuver, the ability to crisply start and stop the maneuver depends on bandwidth (damping), whereas the ability to achieve an acceptably large roll rate depends on this criterion (control power). The criteria are intended to make a distinction between control power and damping (Bandwidth).

The criteria in this section are discussed in detail under their respective paragraph headings.

a. Statement of Requirement

3.3.1 Equilibrium Characteristics. The equilibrium pitch and roll attitudes required to achieve a no-wind hover, and to achieve equilibrium flight in a 35 knot relative wind from any direction, shall not result in pilot discomfort, disorientation, or restrictions to the field-of-view that would interfere with the accomplishment of the Mission-Task-Elements of Paragraph 3.1.1. Nose-up trim attitudes which potentially result in tail boom clearance problems are discouraged.

b. Rationale for Requirement

Experience has shown that pilots complain of a tendency for being disoriented in steady flight at large aircraft attitudes, and object to a need for large changes in attitude to variations in airspeed (see, for example, Reference 105). Because these attitudes and attitude changes are a function of the basic aircraft geometry, it is not practical to modify the equilibrium attitudes with conventional feedbacks (direct force controllers would be required). Therefore, a quantitative specification is not included. However, there is data available to indicate the attitudes, and attitude changes, that are objectionable. These data suggest the limits defined in Table 1. In addition, it would be desirable to limit the magnitude of the gradient of equilibrium pitch (roll) attitude with the relative wind velocity along the longitudinal (lateral) axis to less than 0.30 degrees per knot.

The last sentence of the requirement is based on comments from AH-64 pilots indicating that the large nose-up attitude of that aircraft in low speed causes problems with tail boom clearance. For that reason, the AH-1, which has a relatively level equilibrium pitch attitude for hover and low speed, is preferred for NOE Mission-Task-Elements.

TABLE 1(3.3.1). RECOMMENDED MAXIMUM AND MINIMUM ALLOWABLE EQUILIBRIUM ATTITUDES FOR LEVEL 1

	$\theta$ (deg)	$\phi$ (deg)
No-wind Hover	+10* -2	±4.5
35 kt Relative Wind	+10* -4.5	±6.0

The remainder of this section presents supporting data and rationale for the above recommendations.

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\*The positive pitch attitude limit may be less critical if over-the-nose visibility is not compromised.

Assuming linear aerodynamics, the derivative  $X_u$  (and, by symmetry,  $Y_v$ ) defines the equilibrium speed/attitude relationships as follows:

$$g\Delta\theta - X_u\Delta u = 0$$

$$X_u \doteq g \frac{\Delta\theta}{\Delta u} \doteq \frac{1}{3} \frac{\Delta\theta}{\Delta u} \sim \frac{\text{deg}}{\text{kt}}$$

Hence, the recommended attitude/airspeed limit of -0.3 deg/kt can be thought of as a limit on the "effective"  $X_u$  and  $Y_v$ , although it does not depend on the linearity of these derivatives, since the attitudes and attitude-airspeed gradients are specified directly.

As noted above, setting limits on  $X_u$  and  $Y_v$  presents a dilemma in that they are determined by the basic configuration of the rotorcraft, and cannot be practically altered via augmentation, i.e., direct force controllers would be required. The problem is complicated by the fact that typical values can result in objectionable attitudes in relative winds of 35 knots.\* A compromise solution has been adopted which insures that the no-wind hover attitudes will result in Level 1 flying qualities, with some reasonable degradation allowed for a 35 knot relative wind. The intention is to retain the best possible flying qualities without imposing unrealistic restrictions on the basic rotorcraft configuration. Because of this compromise, and the impracticality of changing  $X_u$  and  $Y_v$  via feedbacks, the requirement is qualitative and the quantitative limits of Table 1 are given only as a recommendation.

### c. Supporting Data

The ground-based simulation study reported in Reference 142 was conducted specifically to investigate the effects of  $X_u$  and  $Y_v$  on flight near hover. The pilot rating data indicated that  $X_u$  or  $Y_v = -0.20$  1/sec resulted in HQRs of 5 to 8.5. The pilot commentary revealed that the dominant factor in the evaluations was an excessively steep nose-down attitude in hover for a 10 kt steady wind, and a requirement for excessive attitude changes to translate in a square pattern while holding a constant heading. With  $X_u$  and  $Y_v$  set to -.05 1/sec, the HQRs improved to the 2-3 range. Attitudes of 6 deg and 1.5 deg were required for  $X_u$  ( $Y_v$ ) values of -0.05 and -0.2 1/sec, respectively, to maintain a steady hover in the 10 kt steady wind used in the Reference 142 simulation. The no-wind hover attitude was zero, and linear aerodynamics were assumed. These data are discussed in some detail in Reference 12, where it is shown that the HQRs for the large  $X_u$  ( $Y_v$ ) cases improve dramatically in a no-wind or gust environment. For example, in two cases with large  $X_u$ , the average HQRs improved from 8.5 to 3.5, and from

---

\*The Operational Flight Envelopes are usually based on a relative wind of at least 35 knots from any direction. There have been comments from Navy specification reviewers that 45 knots would be more desirable for shipboard operations. It would therefore be expected that the compromises made in this paragraph would be more critical for Navy helicopters.

7 to 3, with the removal of all atmospheric disturbances.\* This clearly illustrates that the problem is related to the gust sensitivity, and not the maneuvering characteristics of the high  $X_u$  configurations. This factor must be taken into account when making piloted evaluations in Section 4, where significant winds are not required. A configuration which does not meet the recommended limits in this paragraph could be quite acceptable for the Section 4 maneuvers. Such configurations must be tested in conditions of high winds and turbulence to identify the deficiency upon which the recommended limits are based.

Insight into the bank angle limits can be obtained from an in-flight simulation performed using the Canadian National Research Council (NRC) variable stability Bell 205A (Reference 194). A primary objective of that experiment was to determine the maximum allowable steady lateral acceleration that pilots were willing to tolerate in a wing-low crosswind approach with a deceleration to 20 kts. Approaches were made in varying simulated wind conditions, and the pilots were asked to rate the lateral accelerations as "not noticeable," "noticeable," "borderline objectionable," and "objectionable." The results showed that for  $a_y > 0.06g$ , most runs (greater than 50%) resulted in the lateral acceleration being rated as borderline objectionable or objectionable. A steady lateral acceleration of  $0.06g$  is equivalent to a bank angle of 3.7 degrees. These results suggest a lower acceptable steady bank angle than the Reference 142 results discussed above where a 6 deg trim attitude was excessive.

All U.S. single-main-rotor helicopters hover "left wing low" to counter the thrust of the tail rotor, and as seen in Figure 1, current conventional configurations vary between -1 and -3 degrees. Based on the NRC flight data and the Reference 142 data, it was decided that 4.5 degrees represents a reasonable compromise as a recommended Level 1 limit on the steady bank angle in a no-wind hover. To put this limit in perspective, the Bell 222 hovers at a bank angle of -4 degrees, and pilots generally indicate that this is on the high side, but that the deficiency does not warrant a change.

A maximum of 6.0 degrees is recommended in Table 1 for a 35 knot sidewind based on the NRC data because 90% of the runs at this level of lateral acceleration were rated as objectionable. For a helicopter that hovers at the recommended limit of -4.5 degrees of bank in zero wind, this would imply that the effective linear  $Y_v$  would be greater than -0.10 1/sec.\*\* Fortunately,  $Y_v$  is usually nonlinear and approaches

---

\*Only a few cases were tested in both the 10-knot wind and no-wind environment in the Reference 142 simulation. Of these, two cases were available for comparison that had otherwise Level 1 flying qualities.

\*\*This estimate is based on the linear relationship

$$Y_v \doteq \frac{1}{3} \frac{\Delta\phi}{\Delta v} \doteq \frac{1}{3} \frac{(-6.0 - 4.5)}{35} \doteq -0.10 \text{ 1/sec}$$

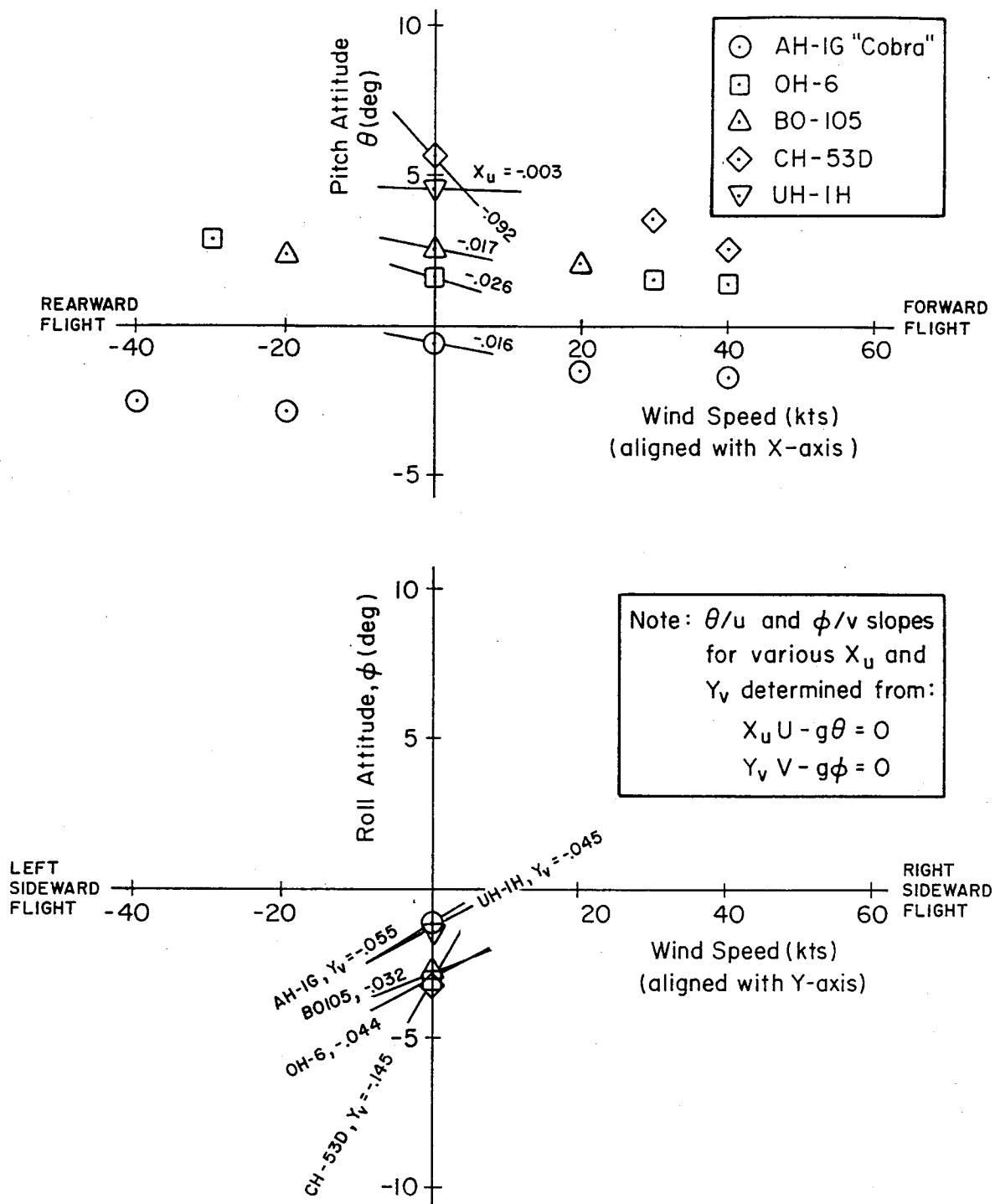


Figure 1(3.3.1). Hover Attitudes for Existing Rotorcraft



zero at 20 knots of side velocity\* (for example, Figure 2, taken from Reference 189). This requirement could have an impact on the basic rotorcraft configuration, since the equilibrium hover bank angle is set by the height of the tail rotor, main rotor tilt, and hinge offset.

Pilot commentary from Reference 142 regarding equilibrium pitch attitude in hover indicated that 6 degrees nose-down pitch attitude was excessive (HQR=5-8.5), and that 1.5 degrees was Level 1. The CH-53 and UH-1 no-wind hover attitudes are 6 and 4.5 degrees nose-up, respectively (see Figure 1), and are apparently not objectionable. On this basis, it is surmised that pilots are more tolerant of nose-up than nose-down equilibrium pitch attitudes, i.e., it is better to "lean back" than to "hang on the seat belts." Positive pitch attitudes are commonly designed into the helicopter for hover to allow an essentially level deck angle in cruise. Hence, it is unlikely that a large negative attitude will be required for a no-wind hover. A recommended limiting value of -2 degrees for the no-wind hover was selected for use in Table 1 based on the Reference 142 HQRs of 2 to 3 for -1.5 degrees, and typical helicopter data of Figure 1. A limit value of -4.5 degrees is recommended in Table 1 for a 35 knot headwind to allow for some attitude-airspeed gradient (negative  $X_u$ ) and still maintain a hover attitude above the objectionable -6 degrees. The above arguments regarding the highly nonlinear nature of  $Y_v$  also apply to the  $X_u$  derivative.

An example of the effect of nose-down equilibrium hover on pilot opinion is found in the U.S. Army Aviation Engineering Flight Activity (AEFA) flight test report on the Bell 214ST helicopter reported in Reference 188. The HQRs varied from 3 in no wind to 5 in a 25 knot wind with specific commentary related to the undesirable nose-down attitude of between 4 and 5 degrees, depending on density altitude. The equilibrium bank angle varied from approximately -1 deg with zero crosswind to -3 deg in a 25 knot left crosswind. The pilots did not find this objectionable, which supports the Table 1 recommended bank angle limits.

A nose-up limit of 10 degrees is included in Table 1 for equilibrium hover independent of the relative wind. This boundary is based on over-the-nose visibility since no other data is available to indicate a nose-up limit. The large deck angles (order of 15 degrees) used by conventional jet transports during climb-out are evidence that pilots are not sensitive to large positive pitch attitudes. A limit value of 10 degrees seems reasonable to maintain if an adequate field-of-view is not a factor. Flight tests conducted by AEFA on the YAH-64 (Reference 132) indicated a 9 degree nose-up attitude in 45 knot rearward flight, and the pilots did not find that objectionable (HQR=3).

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\*On the basis of an effective 20 knot side velocity,  $Y_v$  would be required to be equal to or greater than -.033 1/sec which is met by most helicopters in Figure 1.

In addition to recommending limits on absolute attitudes, a limit on the gradient of equilibrium attitude with airspeed is suggested to insure that excessive attitude changes are not required to maneuver, and that the gust sensitivity is not excessive. Based on the results of References 12 and 142, an effective  $X_U$  and  $Y_V$  of  $-.05$  1/sec produces HQRs of 2 to 3, and  $-.20$  1/sec produces HQR of 5 to 8.5. A value of  $-.10$  1/sec was selected as a reasonable estimate of a Level 1 limit. Since these derivatives can be highly nonlinear for rotorcraft, the change in equilibrium attitude with airspeed is suggested, in lieu of  $X_U$  and  $Y_V$ . Based on the linear relationships, the equivalent gradient is  $-0.3$  deg/kt.

During the specification review process, suggestions were made that the gradients of attitude to relative wind-speed gradients should have an upper limit of zero to maintain a stable (negative) attitude-airspeed relationship. However, a review of the AEFA flight test reports for 5 helicopters revealed that this gradient was positive for the YAH-1S, 214ST, and OH-58, and negative for the UH-60 and the YAH-64, i.e., positive for the teetering rotors, and negative for the articulated rotors (References 76, 132 151, 152, and 188). In all cases, the stick position gradient with airspeed was stable. In none of the teetering rotor cases did the test reports comment on the positive gradient of attitude with airspeed, which would indicate that such a gradient is not objectionable. It must be assumed that this result is contingent on stable, or at least neutral, cockpit control position and force gradients with airspeed. Based on this data, only the magnitude of the attitude-relative wind-speed gradient is recommended, with an additional recommendation that the control position and force gradients be neutral or stable. Note that the control position and force gradients are always neutral for a Rate Command/Attitude Hold Response-Type. Position and force gradients are specified to allow for isometric controllers.

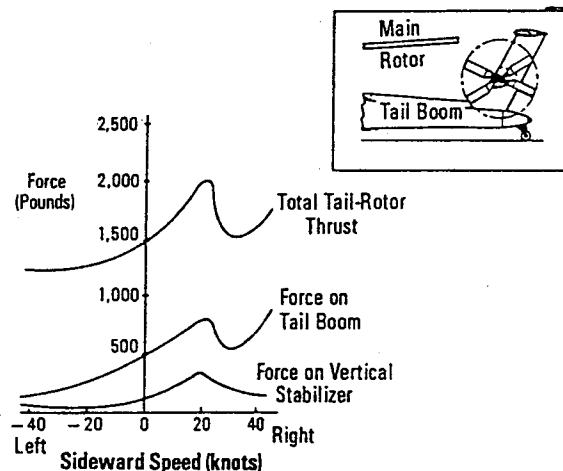


Figure 2(3.3.1). Measured Forces on YAH-64 in Sideward Flight (from Reference 189)

d. Guidance for Application

Comparisons with the equilibrium attitudes suggested by Table 1 may be achieved via 35 knot translations in the fore-aft and left-right directions about a no-wind hover, typical of flight testing currently conducted by the AEFA, e.g., References 76, 132, 151, 152, and 158.

The longitudinal attitude-airspeed gradient should be checked for relative winds of up to 35 knots, or the limits of the Service Flight Envelopes. This may also be accomplished via translations about hover on a no-wind day.

e. Related Previous Requirements

Requirements similar to this one are given in the V/STOL specification, MIL-F-83300, under paragraphs 3.2.1.1 ("changing trim") and 3.2.1.2 ("fixed trim"). These requirements are substantially less stringent than those recommended herein, allowing 10 degrees of bank, a 20 degree pitch change with no absolute limit, and a slope of the equilibrium attitude-speed gradient of 0.6 degrees per knot. However, no supporting data other than "discussions with pilots" were offered to substantiate those more lenient boundaries. The current recommendation is, for the most part, based on data generated after MIL-F-83300 was published. It is notable, however, that the Reference 142 simulation was conducted in direct support of MIL-F-83300, but that the interpretation of the data was less conservative than in the present specification. Specifically, the cases for  $X_u$  ( $Y_v$ ) of -0.2 were considered as "troublesome" but were allowed for Level 1 in MIL-F-83300. In the present discussion such cases are considered to be objectionable based on the HQRs which ranged from 5 to 8.5.

### 3.3.2 Small-Amplitude Pitch (Roll) Attitude Changes

Paragraph 3.3.2 contains the criteria which are intended to insure good precision control involving small amplitudes. This can consist of tight compensatory closed loop tracking, or operations where the pilot's attention must be shared between flying and non-flying tasks. The criteria are divided into short-term (3.3.2.1) and mid-term (3.3.2.2) responses. The short term response refers to the rotorcraft characteristics in fully attended, high-gain, closed-loop tracking tasks. The mid-term response criteria are intended to insure good flying qualities when less aggressive control is required. This is accomplished by specifying the lower frequency modes. For fully attended operations, a low frequency instability is not of significant concern, since such modes are easily stabilized by the pilot. If the pilot must tend to non-control duties, such instabilities become unacceptable, hence the requirements are cast in terms of the divided attention needs of the proposed missions.

a. Statement of Requirement

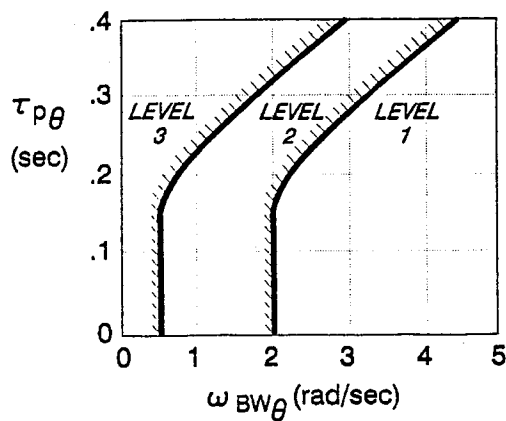
3.3.2.1 Short-Term Response to Control Inputs (Bandwidth). The pitch (roll) response to longitudinal (lateral) cockpit control force or position inputs shall meet the limits specified in Figure 1(3.3). The bandwidth ( $\omega_{BW}$ ) and phase delay ( $\tau_p$ ) parameters are obtained from frequency responses as defined in Figure 2(3.3).

It is desirable to meet this criterion for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish.

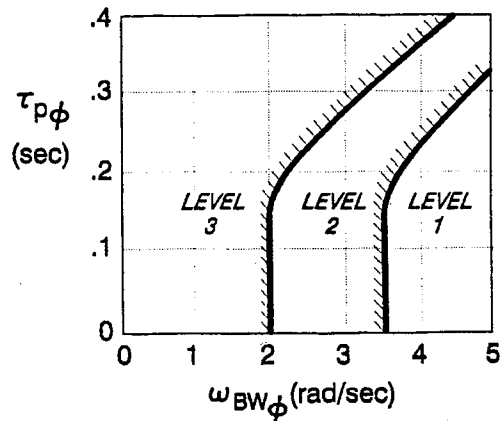
b. Rationale for Requirement

Following is a summary of the rationale for each part of Figure 1(3.3), including the data source and its location in this report:

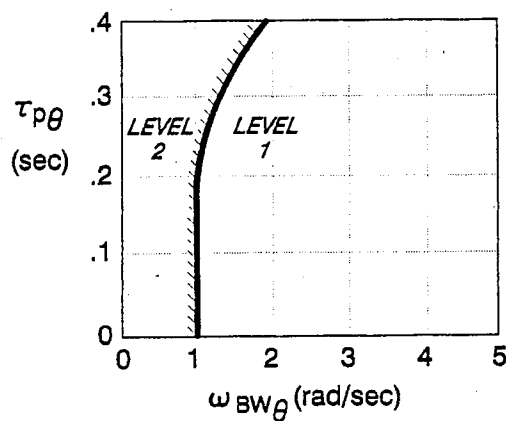
- (a) Target Acquisition and Tracking (pitch): the requirements for this MTE are based on two pitch tracking experiments: an in-flight experiment conducted by the British Royal Aeronautical Establishment (Reference 178), and a fixed-base simulation conducted by Systems Technology, Inc. The results of both experiments and the development of the Figure 1a(3.3) boundaries are given in the Supporting Data for Paragraph 3.4.1.1.
- (b) Target Acquisition and Tracking (roll): the limits in Figure 1b(3.3) are not based directly on roll data, but were developed on the basis of the following three observations: 1) a moving-base simulation study of yaw requirements for air combat (Reference 183), documented in Supporting Data for Paragraph 3.4.7.1; supports a yaw Bandwidth of 3.5 rad/sec for Level 1 and 2 rad/sec for Level 2 (no phase delay variations were made). 2) The roll Bandwidth limits should be at least as high as the pitch limits (2 rad/sec for Level 1, Figure 1a(3.3)); in addition, the lower inertia in roll compared to pitch makes higher Bandwidths possible, so a Bandwidth of 3.5 rad/sec for Level 1 is not unreasonable. 3) The required increase in Bandwidth frequency as phase delay increases should be at least as stringent as for pitch; the shape of the boundaries in Figure 1b(3.3) is identical to that of the limits in Figure 1a(3.3).
- (c) All Other MTEs -- UCE = 1 and Fully Attended Operation (pitch): this limit is based on flight data from the National Research Council of Canada (documented in Reference 157, and in Supporting Data for this Paragraph). Bandwidth, phase delay, task,



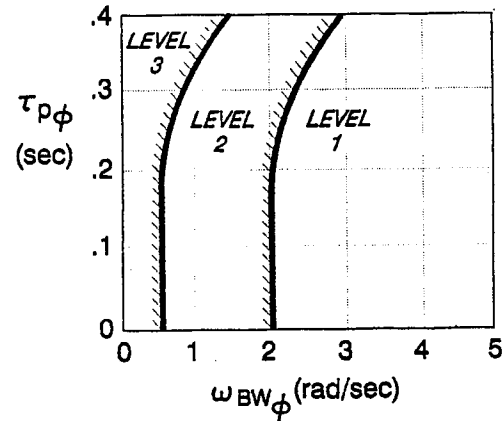
a) Target Acquisition and Tracking (pitch)



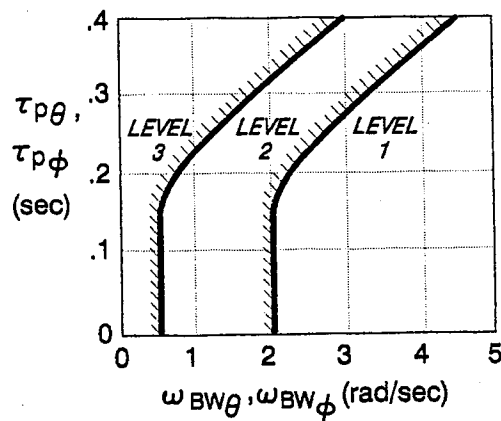
b) Target Acquisition and Tracking (roll)



c) All Other MTEs - UCE = 1 and Fully Attended Operations (pitch)



d) All Other MTEs - UCE = 1 and Fully Attended Operations (roll)



e) All Other MTEs - UCE > 1 and/or Divided Attention Operations (pitch and roll)

Figure 1(3.3). Requirements for Small-Amplitude Pitch (Roll) Attitude Changes -- Hover and Low Speed

### Phase Delay:

$$\tau_p = \frac{\Delta\Phi_{2\omega_{180}}}{57.3(2\omega_{180})}$$

Note: if phase is nonlinear between  $\omega_{180}$  and  $2\omega_{180}$ ,  $\tau_p$  shall be determined from a linear least squares fit to phase curve between  $\omega_{180}$  and  $2\omega_{180}$

### CAUTION:

For ACAH, if  $\omega_{BW_{gain}} < \omega_{BW_{phase}}$ , or if  $\omega_{BW_{gain}}$  is indeterminate, the rotorcraft may be PIO prone for super-precision tasks or aggressive pilot technique.

### Rate Response-Types:

$\omega_{BW}$  is lesser of  $\omega_{BW_{gain}}$  and  $\omega_{BW_{phase}}$

### Attitude Command/Attitude Hold Response-Types (ACAH):

$\omega_{BW} \equiv \omega_{BW_{phase}}$

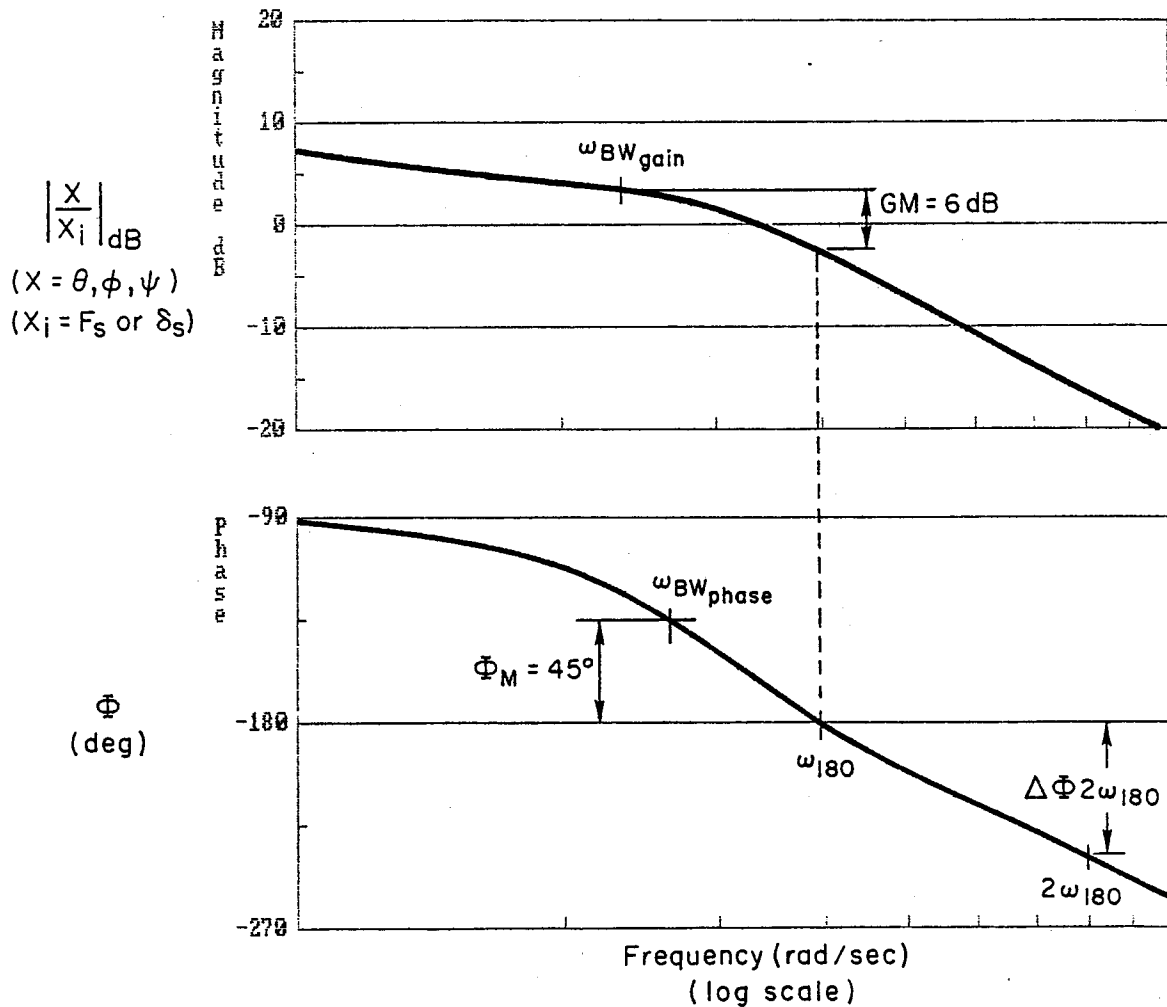


Figure 2(3.3). Definitions of Bandwidth and Phase Delay

and cockpit controller arrangement (conventional and sidestick) were varied in this experiment; the specific limits of Figure 1c(3.3) are based on results of a dash/quickstop (primarily longitudinal) maneuver.

- (d) All Other MTEs -- UCE = 1 and Fully Attended Operation (roll): as for Figure 1c(3.3), discussed above, the limits for Figure 1d(3.3) are based on the NRC experiments of Reference 157, discussed in Supporting Data for this Paragraph. In this case, the boundaries are based on results of a lateral sidestep (primarily lateral) maneuver. Additional support for the Bandwidth limits (though not for the shapes of the limits, since phase delay was not varied) comes from several flight experiments conducted at forward flight speeds, described in Supporting Data for Paragraph 3.4.5.1.
- (e) All Other MTEs -- UCE > 1 and/or Divided Attention Operations (pitch and roll): these requirements are identical in shape to the pitch air combat limits in Figure 1a(3.3). Numerous ground and flight simulations, reported as Supporting Data for this Paragraph, as well as for Paragraphs 3.4.1.1 (pitch) and 3.4.5.1 (roll), suggest Bandwidth limits of 2 rad/sec for Level 1 and 0.5 rad/sec for Level 2. The UCE variation experiments conducted by the NRC also demonstrate a need for Bandwidths of at least 2 rad/sec (see Supporting Data for Paragraph 3.2.2.1). The shapes of the limits have been adopted from Figure 1a(3.3) on the basis that more relaxed limits, such as those in Figure 1d(3.3), can allow relatively large amounts of time delay (on the order of 1/2 second), and this is considered to be excessive when operating in conditions of degraded UCE or divided pilot attention.

A frequency domain criterion has been selected for the small amplitude response characteristics, and time domain criteria are utilized for the larger amplitude motions (e.g., 3.3.3 and 3.3.4) that tend to involve pursuit and precognitive pilot behavior. The justification for using the frequency based bandwidth criterion in this paragraph is given below, followed by a detailed definition of the criterion parameters. References 1, 2, and 3 contain extensive discussions on, and applications of, bandwidth for conventional airplanes, STOLs, and V/STOLs, respectively.

#### 1. Frequency vs. Time Domain Criteria

Early drafts of the specification used limits based on time delay and rise time (e.g., Reference 52). As experience was gained with both the time-domain and frequency-domain criteria, it became clear that the specification of handling qualities for precision tracking with aircraft



attitude is best accomplished with frequency based criteria. The bandwidth criterion emphasizes features directly related to the piloted loop closure. Time domain criteria have been found to be more appropriate to lower frequency phenomena such as pursuit tracking, flight path control, etc. Most time domain criteria for attitude control are based on step or boxcar inputs. Such inputs emphasize the mid-and-low frequency characteristics, at the expense of the response in the region of piloted crossover, which tends to be suppressed to the origin.

A moving-base piloted simulation experiment was conducted on the NASA Ames Vertical Motion Simulator (Ref. 4) specifically to compare rise-time criteria (time to 50% of the first peak in this case) vs. the Bandwidth criterion. The tasks were to 1) hover over a point on the deck of a ship in Sea State 3, and 2) land on that point. Four configurations were formulated which had nearly identical bandwidth, but exhibited wide variations in rise time due to changes in the damping ratio. The relationship between rise time, damping ratio and bandwidth is given in Table 1 for several Response-Types. ACAH was used in the Reference 4 experiment because of known problems with simulator validity for Rate Response-Types. The step input time responses and corresponding pilot ratings for the tested configurations are given in Figure 1. The pilot ratings are essentially invariant in spite of a wide variation in rise time. The pilots noted differences in the open-loop responses during initial familiarization, but these were apparently unimportant in terms of the task. These results suggest that Bandwidth is a better metric than rise time for the prediction of handling qualities for small-amplitude precision tracking tasks. In addition to these results, the time domain criteria had other shortcomings:

- The Level 1 values of rise time involved very small values (order of 0.05 sec for Rate Response-Types). See Reference 52.
- Slight variations in the shape of the "step" input caused significant changes in the rise time.
- Rise time data obtained from flight tests were not repeatable, due to the input shaping problem noted above, atmospheric disturbances, and problems with establishing ideal initial conditions.

Frequency domain criteria tend to be unreliable at the low end of the spectrum because there is typically insufficient power at lower frequencies. The large amplitudes associated with low frequency inputs present practical problems in terms of maintaining trim, maneuvering room, and large deviations from the reference flight condition. Therefore, it is better to utilize time domain criteria for lower frequency tasks. This specification utilizes a mix of time and frequency domain criteria based on the above considerations.

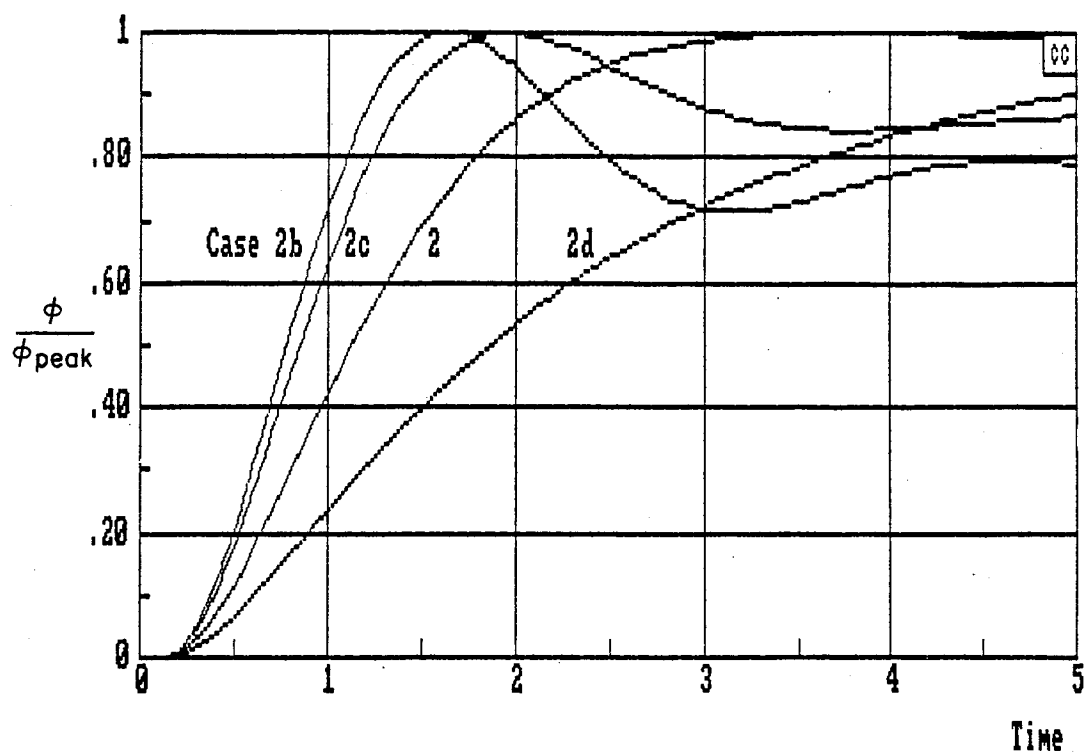
## 2. Definition of Bandwidth

Bandwidth is a term that has classically been used to describe the ability of an electrical network, or a servomechanism, to follow a range

TABLE 1(3.3.2.1) GENERAL RELATIONSHIPS FOR BANDWIDTH

- $\frac{\omega_{BW}}{\omega_n} = K\zeta + \sqrt{K^2\zeta^2 + 1} \quad (\zeta > 0)$
- Approximate Time Delay by  $e^{-\tau s} \doteq \frac{-1(s - 2/\tau)}{s + 2/\tau}$
- $K = \frac{(\omega_{BW}\tau)^2 + 4\tau\omega_{BW}/K' - 4}{(\omega_{BW}\tau)^2/K' - 4\tau\omega_{BW} - 4/K'} = K' \text{ if } \tau = 0$

RESPONSE TYPE	FORM	K'
Attitude	$\frac{\theta}{\delta} = \frac{A_\theta}{[\zeta, \omega_n]}$	1
Rate	$\frac{\theta}{\delta} = \frac{A_\theta (1/\tau_q)}{(0) [\zeta, \omega_n]}$	$\frac{\omega_{BW} - 1/T_q}{\omega_{BW} + 1/T_q}$
Unaugmented (Classical Hovering Cubic)	$\frac{\theta}{\delta} = \frac{A_\theta (1/T_\theta)}{(1/T_{sp}) [\zeta, \omega_n]}$	$\frac{\frac{2}{\omega_{BW}} - (1/T_\theta - 1/T_{sp}) \omega_{BW} + 1/T_\theta T_{sp}}{\frac{2}{\omega_{BW}} + (1/T_\theta - 1/T_{sp}) \omega_{BW} + 1/T_\theta T_{sp}}$



COOPER-HARPER PILOT RATINGS

<u>CASE</u>	<u><math>\zeta, \omega_n</math></u>	<u><math>\omega_{BW\phi}</math></u>	<u><math>\tau_{p\phi}</math></u>	<u><math>\tau_{R50}</math></u>	<u>G</u>	<u>P</u>	<u>S</u>	<u>AVG.</u>
2	0.82, 1.54	2.3	0.09	1.12	4, 5	5	3, 4, 5	4.3
2b	0.37, 2.24	2.6	0.10	0.77	3-3.5	5	5	4.4
2c	0.51, 2.01	2.5	0.10	0.85		3.5, 4	5	4.2
2d	1.36, 1.19	2.3	0.11	1.86	4	4	5	4.3

Figure 1(3.3.2.1). ACAH Cases with Similar Bandwidths But Varying Damping Ratio and Natural Frequency (Reference 4)

of input frequencies. In that context, it is defined as the frequency where the output magnitude is 3 dB less than the input (ratio of 0.707). A good system will have a high bandwidth, and a poor one will have a low bandwidth relative to the maximum input frequency that it is designed to follow. In most cases, the upper bandwidth limit is set by system stability considerations. The development of the Crossover Model in the early 1960's was based on the concept that the human pilot can be treated as an element of a closed loop system for compensatory tracking tasks, and experimental measurements have verified that concept (e.g., see Reference 147). The Bandwidth criterion is an application of the crossover model concept. It is based on the premise that the maximum crossover frequency that a pure gain pilot can achieve, without threatening stability, is a valid figure-of-merit of the controlled element (i.e., similar to a servomechanism). Physically, low values of bandwidth indicate a need for pilot lead equalization to achieve the required mission performance. Excessive requirements for lead equalization have been shown to result in degraded HQRs (Reference 147).

Bandwidth, as defined in the specification, is referenced to the aircraft with all augmentation loops closed (if there are any), but not with the pilot in the loop. The frequency-response data required to measure the bandwidth parameters, defined in Figure 2(3.3), must include all of the elements of a typical analog or digital flight control system as discussed in Reference 171 (e.g., anti-aliasing filters, bending mode filters, stick filters, zero order hold effects, actuators, equalization, and computational delays).

Bandwidth is measured from a frequency response (Bode) plot of angular attitude response to cockpit controller force. As Figure 2(3.3) shows, two bandwidth frequencies are measured: the frequency for 6 dB of gain margin ( $\omega_{BW\text{gain}}$ ), and the frequency for 45 deg of phase margin ( $\omega_{BW\text{phase}}$ ). This describes the margin above the (augmented) vehicle's response in which the pilot can double his gain or add a time delay or phase lag without causing an instability. To apply this definition, one first determines the frequency for neutral stability ( $\omega_{180}$ ) from the phase portion of the Bode plot (Figure 2(3.3)). The next step is to note the frequency at which the phase margin is 45 deg. This is the bandwidth frequency as defined by phase,  $\omega_{BW\text{phase}}$ . Finally, note the amplitude corresponding to  $\omega_{180}$  and add 6 dB. Find the frequency at which this value occurs on the magnitude curve; this is  $\omega_{BW\text{gain}}$ . For Rate Response-Types, the bandwidth criterion frequency,  $\omega_{BW}$ , is the lesser of these two frequencies, while for Attitude Response-Types  $\omega_{BW} = \omega_{BW\text{phase}}$  as discussed in detail in subsection 4, Rationale for Gain Margin Definition of Bandwidth.

### 3. Definition of Phase Delay

Efforts to develop Bandwidth as a generalized criterion for highly augmented aircraft showed that pilots were also sensitive to the shape of the phase curve at frequencies beyond the Bandwidth frequency. This is characterized by the phase delay parameter in Figure 2(3.3).

Figure 2 illustrates that the phase curve can drop off rapidly or slowly, and that the pilot ratings are quite sensitive to this; note that the bandwidths of the two configurations in Figure 2 are essentially identical. Phase delay is a measure of the steepness with which the phase drops off after  $-180^\circ$  and indicates the behavior of the aircraft as the pilot increases his crossover frequency, i.e., "tightens up" beyond the bandwidth frequency. Large values of phase delay mean that there is a small margin (range of frequencies) between normal tracking at 45 degrees of phase margin and instability. The inevitable pilot commentary for an aircraft with large phase delay is that it is PIO prone.

Phase delay is defined so that it represents all of the contribution to phase less than  $-180^\circ$ , and is based on the observation that the phase curve tends to be linear (when plotted on a linear scale of  $\phi$  vs.  $\omega$ ). On the usual linear vs. log scale used for Bode plots, this translates to:  $\Delta\phi = \arg(e^{-\tau_s})$ . The difference between the phase at some frequency  $\omega_1$  and  $-180^\circ$  is given as:

$$\Delta\phi_1 = \phi_1 - (-180) = -57.3\tau_p\omega_1$$

and if  $\omega_1 = 2\omega_{180}$ ,

$$\tau_p = - \frac{\phi_{2\omega_{180}} + 180}{2 \times 57.3\omega_{180}}$$

Therefore, phase delay is defined as a two-point approximation to the phase curve (at  $\omega_{180}$ , and at  $2\omega_{180}$ ). As discussed in Appendix A, it is

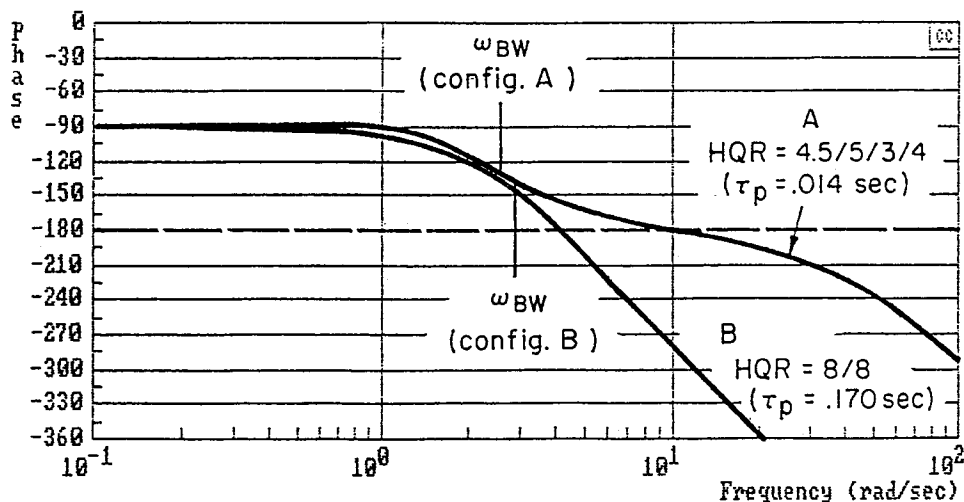


Figure 2(3.3.2.1). Illustration of Effect of Shape of Phase Curve Above  $\omega_{BW}$  on Pilot Ratings (Data for Fixed-Wing Aircraft)

sometimes possible that the phase curve between  $\omega_{180}$  and  $2\omega_{180}$  will contain significant nonlinearities, especially if aircraft vibrational or rotor modes occur in this range. If such a condition exists, a simple least-squares linear fit must be applied to the phase data, plotted on a linear grid of  $\phi$  vs.  $\omega$ , and  $\tau_p$  measured from this linear approximation.

#### 4. Rationale for Gain Margin Definition of Bandwidth

As noted above, the bandwidth criterion is based on the Crossover Model (Reference 147), and is a measure of how tightly a "pure gain pilot" can close the attitude loop without threatening stability. Correlations of pilot commentary with bandwidth as defined by 45 degrees of phase margin, or "phase bandwidth," invariably show that the aircraft becomes increasingly sluggish as the bandwidth decreases. Because of this, "phase bandwidth" is easily related to the more familiar "modes," e.g., short period frequency, roll subsidence, etc. The physical interpretation of bandwidth defined by gain margin is more subtle, and is discussed below.

The definition of bandwidth based on 6 dB of gain margin, or "gain bandwidth," is included because a low value of gain margin tends to result in a configuration which is PIO prone. Low gain margin is a good predictor of PIO prone configurations because small changes in the pilot gain result in a rapid reduction in phase margin. This is illustrated by the example in Figure 3. The combination of a flat region of the amplitude plot and a rapid rolloff in phase (due to a moderate to large value of phase delay) causes a small increase in pilot gain to result in a large decrease in phase margin. Gain-margin-limited systems are defined when  $\omega_{BW_{gain}} < \omega_{BW_{phase}}$ , so that the actual available gain margin for a pilot operating at  $\omega = \omega_{BW_{phase}}$  (point A in Figure 3) is always less than 6 dB. As long as the pilot does not "tighten up," the problem is transparent, and in fact, the response could be described as crisp and desirable if the phase bandwidth is large. Such an aircraft could be quite acceptable as long as the required tasks do not involve aggressive closed loop control, or extreme precision. The phenomenon is insidious because it depends on pilot technique. A smooth, non-aggressive pilot might never encounter the problem, whereas a more aggressive pilot could encounter a severe PIO.

It may be expected that a manufacturer that fails the bandwidth requirement due to a gain margin problem would file for a deviation on the basis that pilot comments indicate that the response is crisp, and very adequate for all tasks accomplished during the evaluations. Such configurations should be tested in the most critical environment (e.g., a gusty wind) and the most precise closed-loop mission task envisioned for the helicopter, with several pilots, before ignoring the quantitative or recommended gain margin limits in this paragraph; i.e., a gain margin limit should not be ignored for an ACAH Response-Type even though it is not part of the requirement in Figure 1(3.3).

The definition of bandwidth for ACAH Response-Types does not include a gain margin limit. This was done because correlations of the flight test results with the variable stability Bell 205A (discussed in

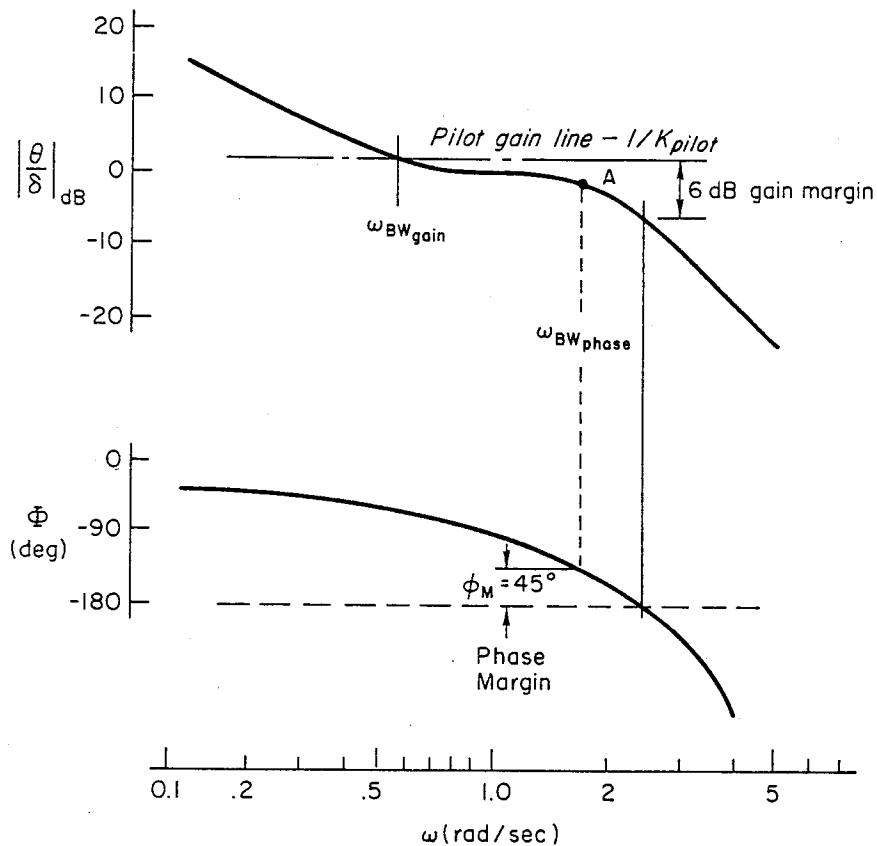


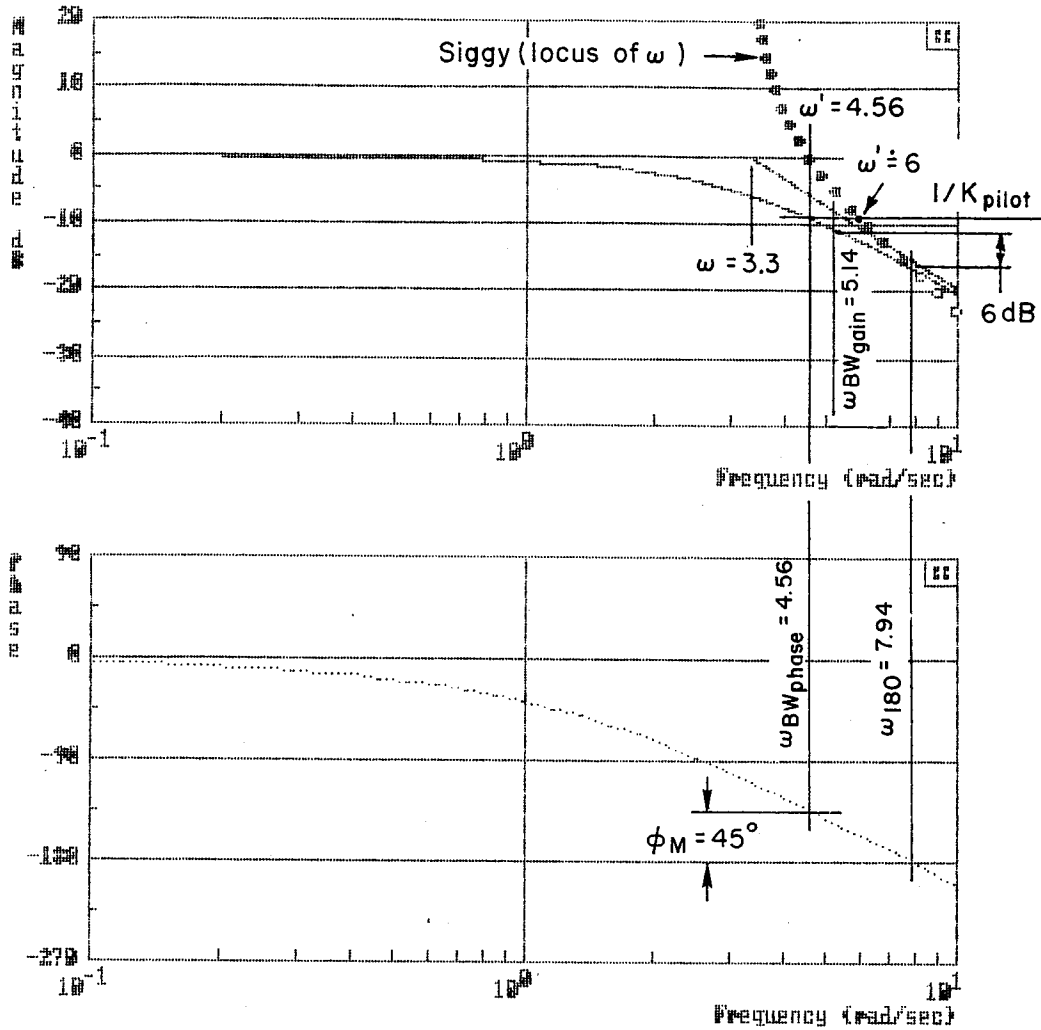
Figure 3(3.3.2.1). Example of a Gain Margin Limited System

Supporting Data) predicted that very low bandwidths defined by gain margin were rated as acceptable for ACAH (but not for RCAH). The nature of ACAH is such that the pilot does not have to close the attitude loop for stabilization purposes (as is necessary for Rate, RCAH, and unaugmented helicopters) so that gain margin problems are less apparent. A first-principles physical explanation of this data is rooted in the theory of closed-loop pilot-vehicle analysis, which is given in the backup material for Paragraph 3.2.5.

To take advantage of an ACAH SCAS, it is necessary for the pilot to revise his technique from "normal" helicopter flying, either by not closing the attitude loop, or by closing it with very low gain. Problems can arise if the pilot is asked to perform a precision task with ACAH that requires a bandwidth higher than can be attained with the basic attitude SCAS. In such a case, the pilot must improve on the SCAS attitude loop. If the attitude loop is gain-margin-limited, there will be a PIO tendency as discussed below.

Ideally, the Bode plot for ACAH has the generic characteristics shown in Figure 4a. As shown by the siggy plot in Figure 4a (see Reference 35 for a discussion of siggy plots), the pilot gain line (defined by  $1/K_{pilot}$ ) must fall below the value of the low frequency

$$\frac{\theta}{\delta_{es}} = \frac{K e^{-0.1s}}{(s^2 + 2(.97)(3.3)s + 3.3^2)}$$

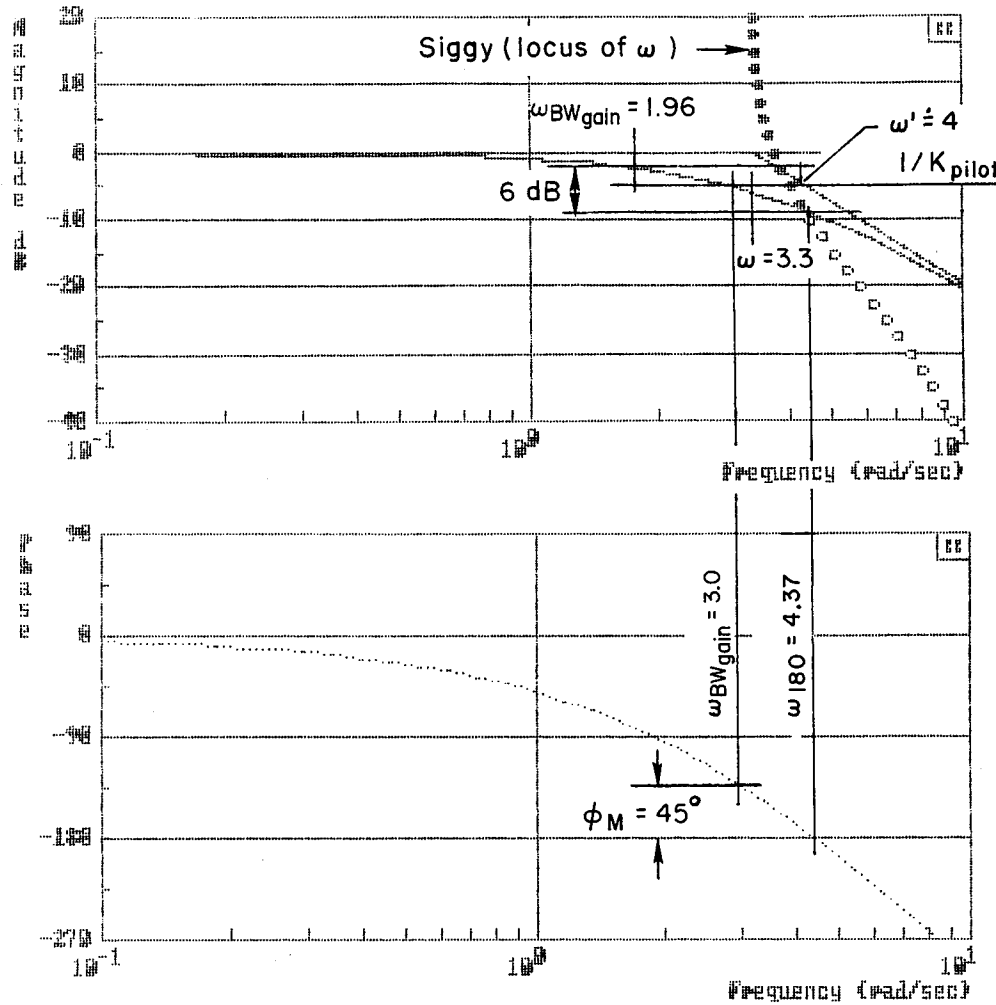


a) Case I- Phase Margin Limited System  
 ( $\omega_{BW_{phase}} < \omega_{BW_{gain}}$ )

Figure 4(3.3.2.1). Pilot Loop Closure Characteristics for ACAH Response-Types



$$\frac{\theta}{\delta_{es}} = \frac{K e^{-0.3s}}{(s^2 + 2(.97)(3.3)s + 3.3^2)}$$



b) Case 2 – Gain Margin Limited System  
 $(\omega_{BW_{gain}} < \omega_{BW_{phase}})$

Figure 4(3.3.2.1). (Concluded)

asymptote before the characteristic roots of the augmented rotorcraft begin to move (i.e., before his closure has any noticeable effect).

In this example, the siggy plot shows that the characteristic second order mode is increased from an open loop value of 3.3 to approximately 6 rad/sec (labeled  $\omega'$ ) if the pilot closes the loop at 45 degrees of phase margin, i.e., at the bandwidth frequency. Contrast this with a gain-margin limited system, as shown in case 2 (Figure 4b). If the pilot closes the loop at the same 45 degrees of phase margin as in case 1, the characteristic second order mode is only slightly affected (second order characteristic root moved from 3.3 to 4 rad/sec). If the pilot attempts to further increase his gain to quicken the response (i.e., to further modify the characteristic second order mode), he is in danger of encountering a closed-loop instability because of low gain margin (less than 6 dB). As noted above, for gain-margin-limited systems, the gain margin is always less than 6 dB at the phase bandwidth frequency. The root causes of the poor loop closure characteristics in case 2 are a combination of a relatively flat amplitude response combined with a rapid roll-off in phase (large value of the phase delay parameter  $\tau_p$ ). Such characteristics are classically defined when the gain bandwidth is less than the phase bandwidth, i.e., the system is gain-margin-limited.

The phase bandwidth is Level 1 in both of the examples in Figure 4. If the required task precision is such that the desired performance can be achieved without an attitude closure, cases 1 and 2 would both be satisfactory, and probably indistinguishable to most pilots. An overly aggressive pilot would probably encounter some PIO tendencies with case 2, but this does not warrant a change in the requirement if the defined mission tasks can be accomplished, and the PIO can be easily stopped without significant excursions in position.

The above discussion illustrates that ACAH Response-Types, which are highly effective for reducing workload, can produce a PIO if pushed beyond the inherent capabilities of the SCAS. For example, during the NRC of Canada flight test experiments (discussed in Supporting Data) the horizontal touchdown tolerance was on the order of 2 ft, and the ACAH Response-Types were rated very well. However, at the end of each session it was necessary to land the helicopter on a trailer so that it could be rolled into the hangar. Here the touchdown precision tolerance was on the order of a few inches. A short experiment was conducted to determine if this could be accomplished with an ACAH SCAS which was borderline gain-margin-limited, but well within Level 1 (HQR=2) in terms of the tasks implemented in the experiment. These tests resulted in pilot comments of "unable to tighten up on the gain because it is slightly PIO prone," and "you can't tease it" (HQR=4). The pilot indicated that the PIO tendency was slight, which is probably because the nature of ACAH is to provide improved stability as soon as the pilot backs off.

We are faced with a dilemma: on one hand gain-margin-limited ACAH Response-Types lead to PIO for super-precision tasks, and on the other, disallowing such configurations robs the pilot of workload relief for many other, less aggressive, Mission-Task-Elements. The approach taken

herein has been to eliminate gain margin from the definition of bandwidth for ACAH Response-Types, but to recommend avoidance of ACAH systems where the gain bandwidth is less than the phase bandwidth, especially if super-precision tasks are required. Additional motivation for not including gain bandwidth as a formal requirement for ACAH was that the PIO due to gain margin limiting has not been found to be violent for ACAH Response-Types. It should be emphasized that this is not expected to be the case for Rate or RCAH Response-Types, where the pilot attitude closure is necessary to maintain the stable hover, and consequently, it is not possible to completely "back out" of the loop. Therefore, gain bandwidth is retained for these Response-Types.

Gain-margin-limited systems result from a large phase delay, combined with flat amplitude characteristics. Large phase delays usually result from inherent rotor system time delay (65 to 130 ms), combined with computer throughput delays, actuator lags, filters, etc. The flat amplitude characteristic is, of course, inherent to ACAH, and can occur in RCAH Response-Types due to the nature of the feedforward equalization.

#### 5. Effect of Task and Visual Environment on the Required Bandwidth

The Bandwidth criterion boundaries in Figure 1(3.3) depend on the task (MTE), the visual conditions (UCE), and the required divided attention. Note that a higher Bandwidth is required for "target acquisition and tracking" than other MTEs. Increased Bandwidth is also required in a degraded usable cue environment ( $UCE > 1$ ), and for divided attention operations. These requirements indicate that a crisper, more predictable response is necessary if the helicopter is to be used as an aiming platform, or if the visual cues are degraded. The need for increased bandwidth for target acquisition and tracking is supported by the data in Section c, Supporting Data.

The use of 2 rad/sec as the minimum Bandwidth for operations in conditions of degraded visual cueing or divided attention (see Figure 1e(3.3)) is based on the results of variable stability flight tests with night vision goggles. It was noted during the initial trials in an inflight visual cueing experiment (described in Paragraph 3.2.2.1) that the pilot workload relief associated with attitude augmentation could not be realized for values of Bandwidth less than about 2 rad/sec (and 2.5 to 3 rad/sec was preferred). This result has also been noted during ground based simulation where the visual cueing is a well known problem for hovering tasks. These observations are supported by closed-loop pilot-vehicle analyses for the hover flight condition. Using the closed-loop pilot-vehicle model structure from Figure 2(3.2.7), the root loci resulting from varying the piloted position loop closure gain,  $K_x$ , are shown in Figure 5 as a function of the Bandwidth of the attitude system ( $\theta/\theta_c$ ).<sup>\*</sup> The maximum achievable closed-loop damping ratio of

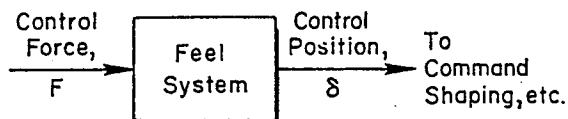
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<sup>\*</sup>More details concerning the assumptions supporting this closed loop analysis are given in the discussion for Paragraph 3.2.7. The value of  $T_{L_x}$  used in this analysis was 4 sec.

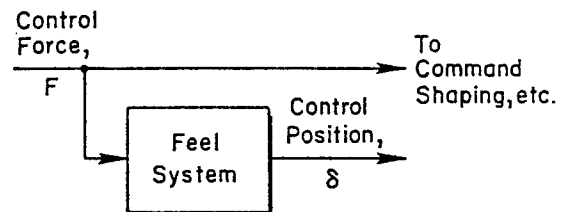
the position mode is essentially zero for  $\omega_{BW} = 1$  rad/sec, and increases to 0.30 for  $\omega_{BW} = 2.0$  rad/sec. The unstable position locus for the low bandwidth attitude system indicates that additional equalization is required to accomplish a stable hover. This has been shown to result in excessive pilot workload in conditions of degraded visual cueing (UCE > 1), or where divided attention requirements are high.

#### 6. Force vs. Position as the Cockpit Controller Input

The role of the feel system in the calculation of  $\omega_{BW}$  and  $\tau_p$  is currently not well understood. Hence, the user is asked to check the criterion parameters for force and position inputs. This actually only applies when the flight control system inputs are based on control position -- i.e., when position sensors are used. In this case, the force feel system dynamics are in series with the other vehicle dynamics, and directly affect the pilot's ability to control the aircraft. The generic block diagram in the following sketch (part(a)) is a position-sensing system.



*a) Position Sensing: Feel System is in Series*



*b) Force Sensing: Feel System is in Parallel*

If, instead, control force sensing is used, the feel system dynamics are in parallel, not series (part (b) of the sketch), so they clearly should not be included in the calculation of  $\omega_{BW}$  and  $\tau_p$ . Experience has shown that force sensing results in overly abrupt responses which must be filtered, thereby adding to the total lag. Position sensing is almost always used except for aircraft with small-displacement controllers.

As noted above, it is currently not known whether the feel system should be included as part of the aircraft in the calculation of  $\omega_{BW}$  and  $\tau_p$ , or treated separately, even for position sensing flight control systems. The argument centers around what effect, if any, the feel system dynamics might have on the pilot's ability to close the pilot/vehicle loop. At this writing, this is a controversial subject that has not been resolved. The approach taken in the specification has been to allow the contractor to meet the bandwidth and phase delay requirements for position or force inputs. The response to force inputs will usually have a lower bandwidth and higher phase delay because of the lags in the feel system. If this results in an excursion into the

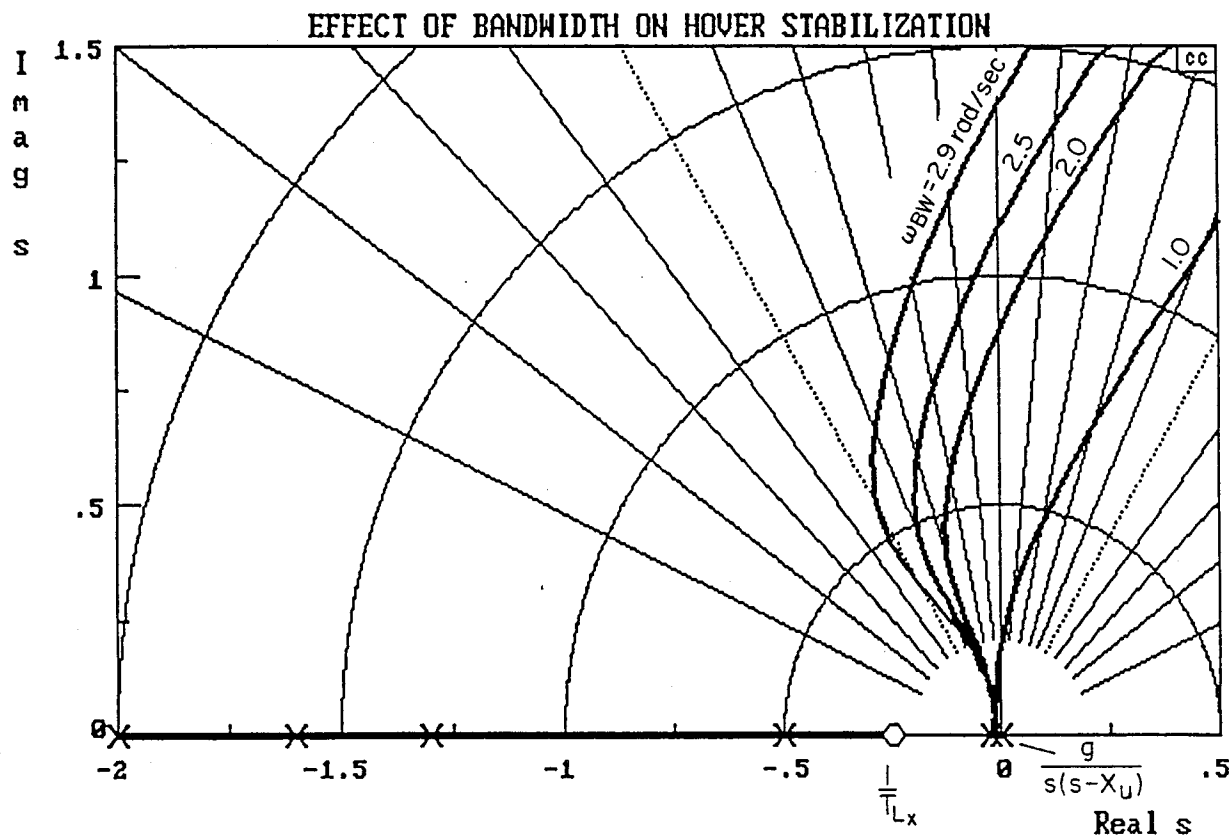
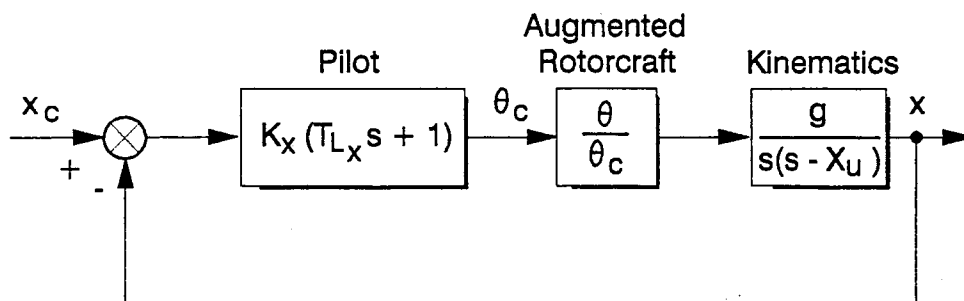


Figure 5(3.3.2.1). Loci of Pilot-Vehicle Position Mode for Hover Stabilization as a Function of Pitch Attitude Bandwidth

Level 2 region, the feel system may be excessively sluggish, and flight testing is required to investigate this possibility. Such flight testing should focus on the most precise pointing tasks envisioned for the rotorcraft. Fixed-wing experience has shown that such testing is usually not accomplished in practice. For example, a well-accepted highly augmented fighter aircraft, known to have large phase delay, was nonetheless touted as having excellent up-and-away flying qualities. However, recent testing for agility showed that it was easier to perform precision pointing with much older aircraft (HQR 3 to 4) than with the modern highly augmented one (HQR 7 to 8). The exceptional thrust-to-weight of the aircraft, and the fact that most air-to-air engagements do not require precision pointing, apparently caused the high phase delay to be of little importance. Recent air-to-air tactics have caused the community to focus on improved agility, which exposed the problem. The lesson to be learned is that the flying qualities tasks must be well defined, precise, and aggressive. And a discriminating pilot attitude is essential. This is especially important in cases where quantitative criteria cannot be written (such as the last paragraph in the present requirement).

This paragraph should be updated when sufficient data is available.

#### 7. Control Sensitivity

None of the criteria for attitude and height control provide explicit limits for control sensitivity. However, it is required (Paragraph 3.6.3) that it be separately optimized during simulations and initial flight testing. The lack of experimental data upon which to base control sensitivity is probably a result of two factors: 1) it is assumed that the control gearing can be easily changed, especially with fly-by-wire aircraft, and therefore it is a low-priority parameter, and 2) a very large data base would be required to formulate a quantitative control sensitivity specification, especially considering that sidestick, centerstick, fixed, and moving controllers should all be considered.

The lack of detailed requirements for control sensitivity represents a weakness of the specification. Even the most experienced and perceptive test pilots can be, and have been, fooled by varying control sensitivity. Excessively high control sensitivity looks like low damping, is therefore PIO prone, and will receive comments to that effect (few, if any, pilots will identify the problem as excessively high control sensitivity). Similarly, excessively low control sensitivity will receive comments related to an overly sluggish response.

The control sensitivity should logically be specified over the band of frequencies where the pilot is most sensitive to the aircraft response. For closed-loop compensatory tracking tasks this would, by definition, be the region of piloted crossover (approximately 0.8 to 4 rad/sec for attitude control, and 0.3 to 2 rad/sec for flight path control, depending on the task). Until such data becomes available, it is important that the user of this specification be aware of the impact

of a non-optimized control sensitivity. This one factor has been, and continues to be, a major source of confusion and misinterpretation of handling qualities criteria such as Bandwidth.

Control sensitivity is further discussed under Related Previous Requirements.

### c. Supporting Data

The bandwidth versus phase delay limits of Figure 1(3.3) are separated into three categories: 1) requirements for target acquisition and tracking (the low-speed equivalent of air combat), in ideal visual conditions; 2) MTEs performed in ideal visual conditions with full attention to controlling the aircraft; and 3) MTEs performed in conditions of reduced visual cues or with divided-attention tasks. The supporting data reviewed here are almost entirely concerned with full-attention flight in an ideal UCE. These data are reflected directly in the limits of Figures 1c(3.3) and 1d(3.3) of the specification.

The rotorcraft data base for both Rate and Attitude Response-Types performing hovering and low-speed tasks is quite limited, especially for flight test data. The flight data are comprised primarily of the results of experiments conducted by the National Research Council (NRC) of Canada in collaboration with the U.S. Army Aeroflightdynamics Directorate, explicitly for developing this specification. Before reviewing the flight data, results from ground-based simulations will be discussed.

## REVIEW OF SIMULATION DATA

### 1. V/STOL Shipboard Landing Experiment (Reference 4)

A piloted simulation was performed on the NASA Ames VMS (Reference 4) to gather a thorough set of pilot ratings and comments for several Response-Types to support revision work on MIL-F-83300. Figures 6 and 7 summarize all the pilot ratings obtained for the shipboard landing task using Rate Command/Attitude Hold (RCAH) and Attitude Command/Attitude Hold (ACAH) Response-Types, respectively.

There is some scatter in the pilot ratings for RCAH Response-Types (Figure 6) that, at first glance, does not appear to be excessive. However, this is because most of the evaluations were performed by Pilot S, who was apparently able to distinguish between low-bandwidth and high-bandwidth systems. Neither Pilot G nor Pilot P was able to make use of increased bandwidth for improved handling qualities on the simulator.

There are several theories to explain the problems with evaluating Rate (or RCAH) Response-Types on the ground-based simulator:

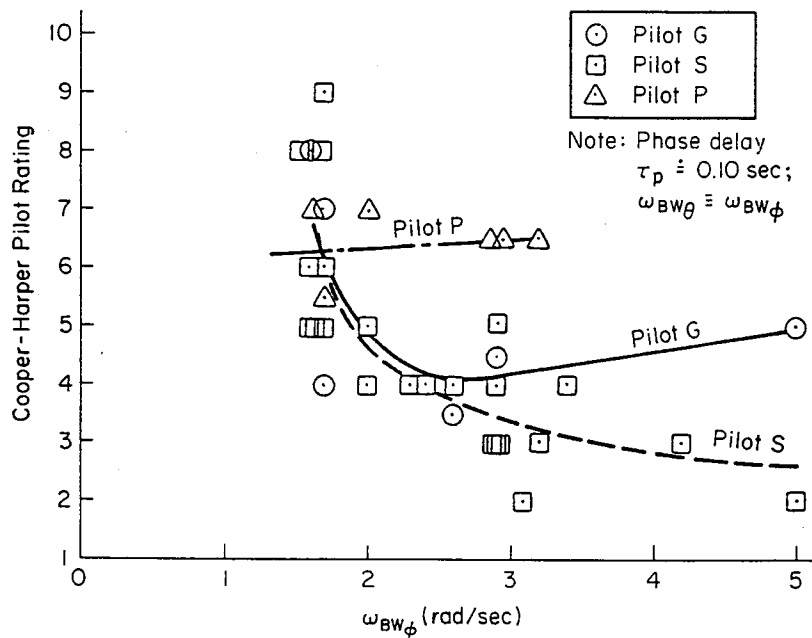


Figure 6(3.3.2.1). Pilot Rating vs. Bandwidth for RCAH Cases.  
Vertical Landing Task (Reference 4)

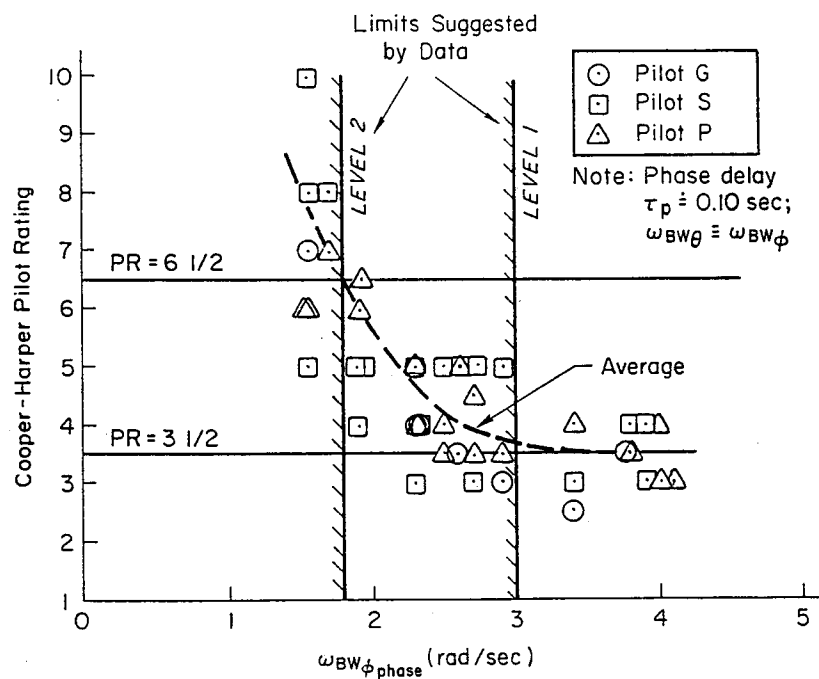


Figure 7(3.3.2.1). Pilot Rating vs. Bandwidth for ACAH Cases.  
Vertical Landing Task (Reference 4)



- Excessive time delays due to combined computer, CGI, and motion system. These delays for the VMS were on the order of 200 msec (Reference 49).
- Lack of fine-grain details (microtexture) on the CGI display. This would have a major influence on the pilot's ability to perceive small drift rates and thus would degrade performance. In the landing task used for the Reference 4 simulation, this was a problem expressed by all the pilots. The flight tests of Reference 156 strongly support the need for microtexture for precise hovering.
- Motion washout dynamics that introduce phase differences between motion and visual perception.

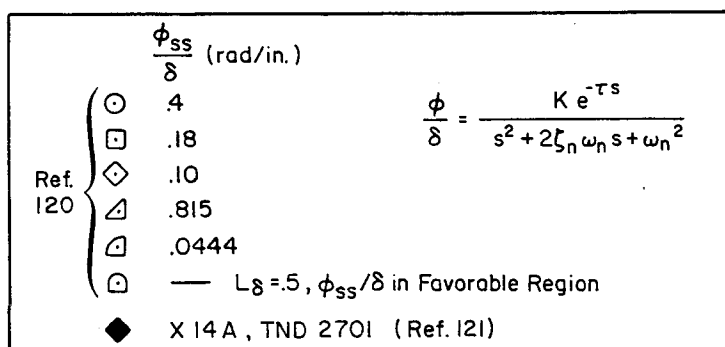
An unfortunate result of all this is that only flight test data can be reliably used to define bandwidth boundaries for Rate and RCAF Response-Types, and the availability of such data is extremely limited.

The ratings for Attitude Response-Types in Figure 7 are quite consistent between pilots, especially when contrasted with the Rate Response-Type data of Figure 6: inter-pilot ratings are within a two-point spread until bandwidth becomes very low. The average rating trend (dashed line on Figure 7) suggests that the Level 1 and 2 limits on roll bandwidth should be approximately 3 and 1.8 rad/sec, respectively.

## 2. VTOL Precision Hover Experiment (Reference 120)

The experiment of Reference 120 was conducted on the NASA Ames S01 six-degree-of-freedom motion simulator. This simulator has limited motion capability (45 deg in pitch, roll, and yaw, and 9 ft translational limits in all directions), and the tasks were performed using a "real-world" scene -- a hangar ramp, available by opening hangar doors in front of the simulator -- so there are none of the concerns about visual delays, texture, etc., inherent to other simulations. Two pilots participated in the program; the tasks involved precision hover and limited maneuvering. No winds, gusts, etc., were included.

Pilot ratings from the Reference 120 simulation for Attitude Response-Types are plotted as a function of natural frequency ( $\omega_n$ ) vs. total damping ( $2\zeta\omega_n$ ) in Figure 8. Also shown are a limited number of flight test ratings from Reference 121. The data of Figure 8 indicate that the proposed roll bandwidth limit of 2 rad/sec may be slightly higher than is suggested by this data (note the lines of constant  $\omega_{BW}$ ). The flight test data discussed below support the higher limit, however.



Sensitivity Separately Optimized

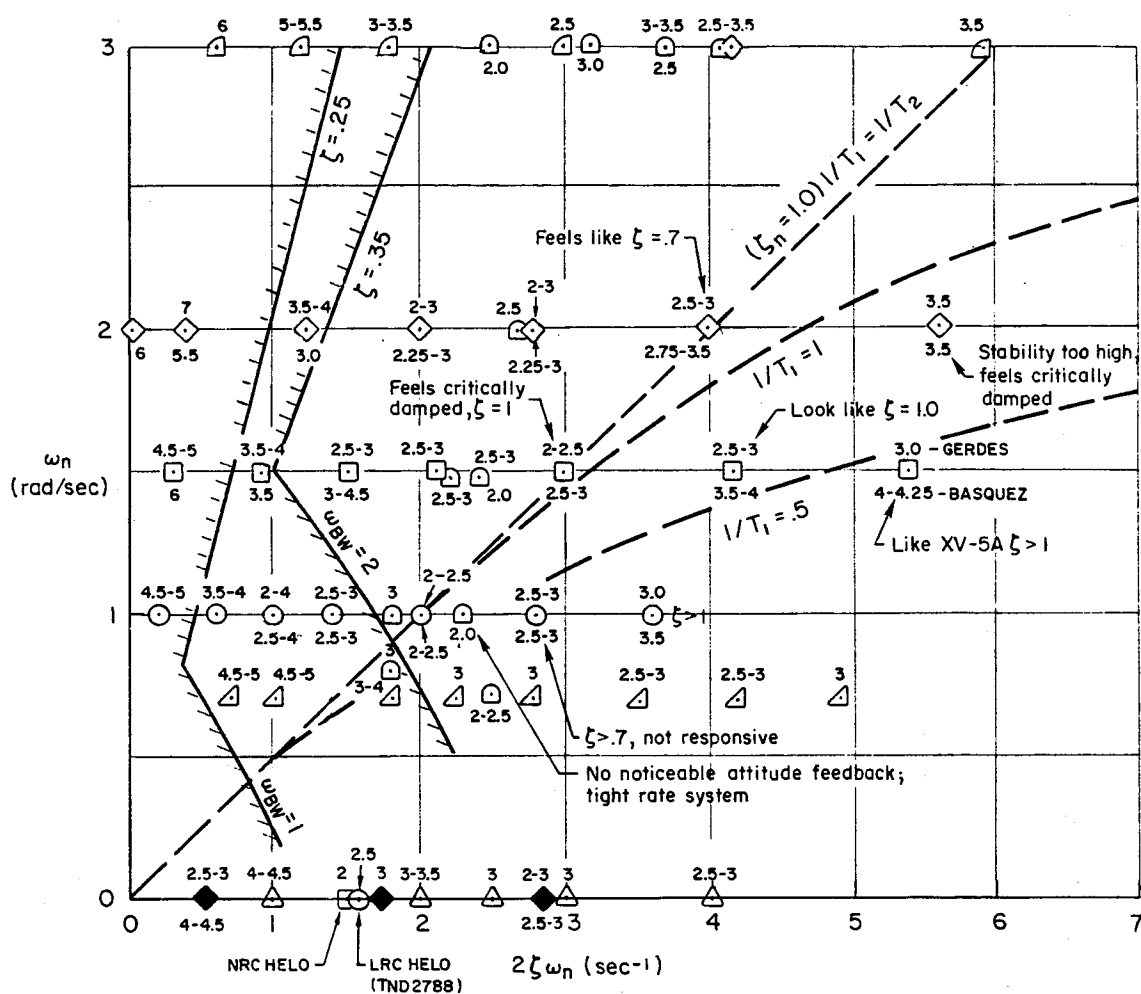


Figure 8(3.3.2.1). Pilot Ratings from S01 Simulation of Reference 120. Cooper Ratings (Figure Reproduced from Reference 12)

## REVIEW OF FLIGHT DATA

### 1. Variable Stability Bell 205 Experiments (Reference 157)

Using the National Aeronautical Establishment (NAE) variable-stability airborne simulator (a modified Bell 205A-1 helicopter), researchers at the Canadian National Research Council (NRC) have conducted several experiments that increase the data base considerably. The Reference 157 experiment was directed at defining the pitch and roll response requirements for low-speed maneuvering. One of the goals of the flight tests was to reproduce a selected portion of the evaluations from the Reference 4 ground-based simulation.

#### Experiment

The handling qualities tasks are illustrated in Figure 9. The course shown in Figure 9 was flown a minimum of three times for each configuration. As a result, the pilot performed 9 precision hover tasks, 9 landings, and 3 sidesteps and dash/quickstops, for a flying time of between 20 and 30 minutes for each evaluation. Cooper-Harper pilot ratings and comments were obtained for four tasks: hover, landing, sidestep and quickstop. Six evaluation pilots participated in the study.

Fourteen pitch and roll configurations were evaluated, covering a wide range of Response-Type and response bandwidth. The primary Response-Types were Attitude Command/Attitude Hold (ACAH), Rate Command/Attitude Hold (RCAH), and pure Rate (without any hold capability). All of the systems employed full-time pitch and roll rate feedbacks.

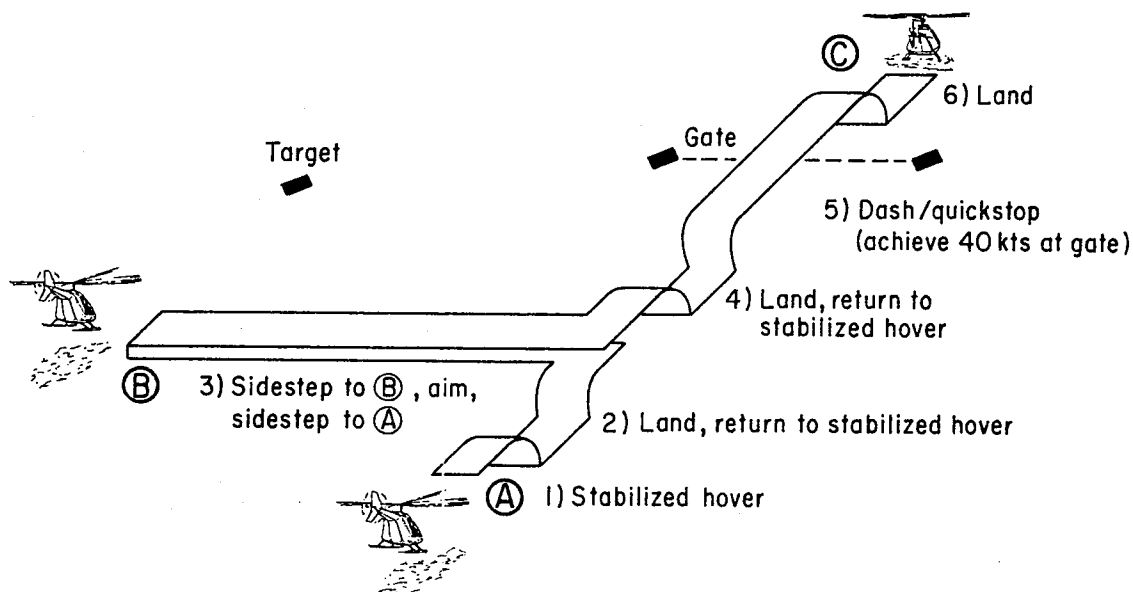


Figure 9(3.3.2.1). Task Scenario for NRC (Canada) Flight Tests  
(from Reference 157)

Attitude feedback was added to achieve the Attitude Command/Attitude Hold (ACAH) Response-Types. An integral-plus-proportional element was added to the input path to obtain a RCAH Response-Type from the ACAH system. Table 2 summarizes the characteristics of the configurations, identified from flight derived "frequency sweeps" (see Appendix A).

Pitch and roll control sensitivity were separately optimized by each pilot prior to making a formal evaluation.

The yaw control system was a Rate Command/Heading Hold Response-Type, and heave control was the basic Bell 205A. The aircraft was judged to be Level 1 in the directional and vertical axes, albeit slightly worse than Level 1 in the vertical axis.

Two cockpit controller arrangements were used for the evaluations: conventional controls, with a centerstick cyclic, left hand collective, and pedals; and an integrated four-axis limited displacement sidestick. A complete evaluation was conducted with both controller arrangements.

### Results for Conventional Controls

Figure 10 summarizes the pilot ratings from the conventional-controller evaluations as a function of pitch and roll bandwidth. Each symbol represents between three and six evaluations of each case. It is important to note that Figure 10 does not account for variations in phase delay (Table 2).

The data of Figure 10 do not show any significant differences in pilot ratings between the three Response-Types evaluated and it appears that the pilots were, for the most part, indifferent to the specifics of the short- or long-term response. Other studies have shown that ACAH is preferred over RCAH for operations in low visibility (e.g., Reference 156), and that RCAH is preferred over ACAH for air combat (e.g., Reference 155), but the current results indicate that for low-speed, maneuvering tasks accomplished in good visibility, such differences do not exist.

Comparison of all of the average-pilot-rating curves in Figure 10 indicates that the precision hover task was the easiest overall (i.e., the average ratings are better), while the landing task was the most difficult. The fact that landing was more difficult than precision hover is consistent with the Reference 4 simulation results; it is interesting, however, that the quickstop and sidestep tasks were also somewhat easier than the landing task. In general, there are only slight differences in the average-rating curves for all four of the tasks.

As Figure 10 shows, bandwidths considerably less than 3 rad/sec (the Level 1 limit suggested by the simulation results of Figure 7) were acceptable in flight. In addition, there is a difference in the variation of pilot rating with bandwidth for pitch versus roll; Figure 10 suggests that the pilots were more sensitive to roll bandwidth. Based

TABLE 2(3.3.2.1). DYNAMICS OF CASES EVALUATED  
(FROM REFERENCE 157)

CASE	RESPONSE TYPE	PITCH CHARACTERISTICS			ROLL CHARACTERISTICS		
		BANDWIDTH		$q_{pk}/q_{ss}$ [ $\theta_{pk}/\theta_{ss}$ ]	BANDWIDTH		$p_{pk}/p_{ss}$ [ $\phi_{pk}/\phi_{ss}$ ]
		$\omega_{BW\theta}$ (rad/sec)	$\tau_{p\theta}$ (sec)		$\omega_{BW\phi}$ (rad/sec)	$\tau_{p\phi}$ (sec)	
0	ACAH	1.75	0.25	[1.1]	1.6	0.29	[1.0]
1	ACAH	2.7	0.25	[1.0]	2.8	0.21	[1.0]
2	ACAH	2.7	0.21	[1.0]	3.1	0.16	[1.1]
3	ACAH	2.4	0.28	[1.0]	2.8	0.28	[1.1]
4	RCAH	0.75	0.29	1.0	1.2	0.31	1.2
5	RCAH	1.9	0.23	1.0	1.5	0.16	1.0
6	RCAH	2.0	0.20	1.0	2.0	0.23	1.0
7	RCAH	2.8	0.18	1.8	2.3	0.15	2.0
8	RCAH	0.85	0.0	1.7	1.0	0.23	1.8
9	RCAH	1.6	0.35	1.1	1.85	0.17	1.0
10	Rate	1.5	0.22	1.2	3.0	0.12	1.0
11	Rate	1.8	0.09	1.1	3.4	0.12	1.0
12	Rate	2.2	0.17	1.1	1.9	0.14	1.0
13	Rate	1.9	0.21	1.0	3.0	0.11	1.3

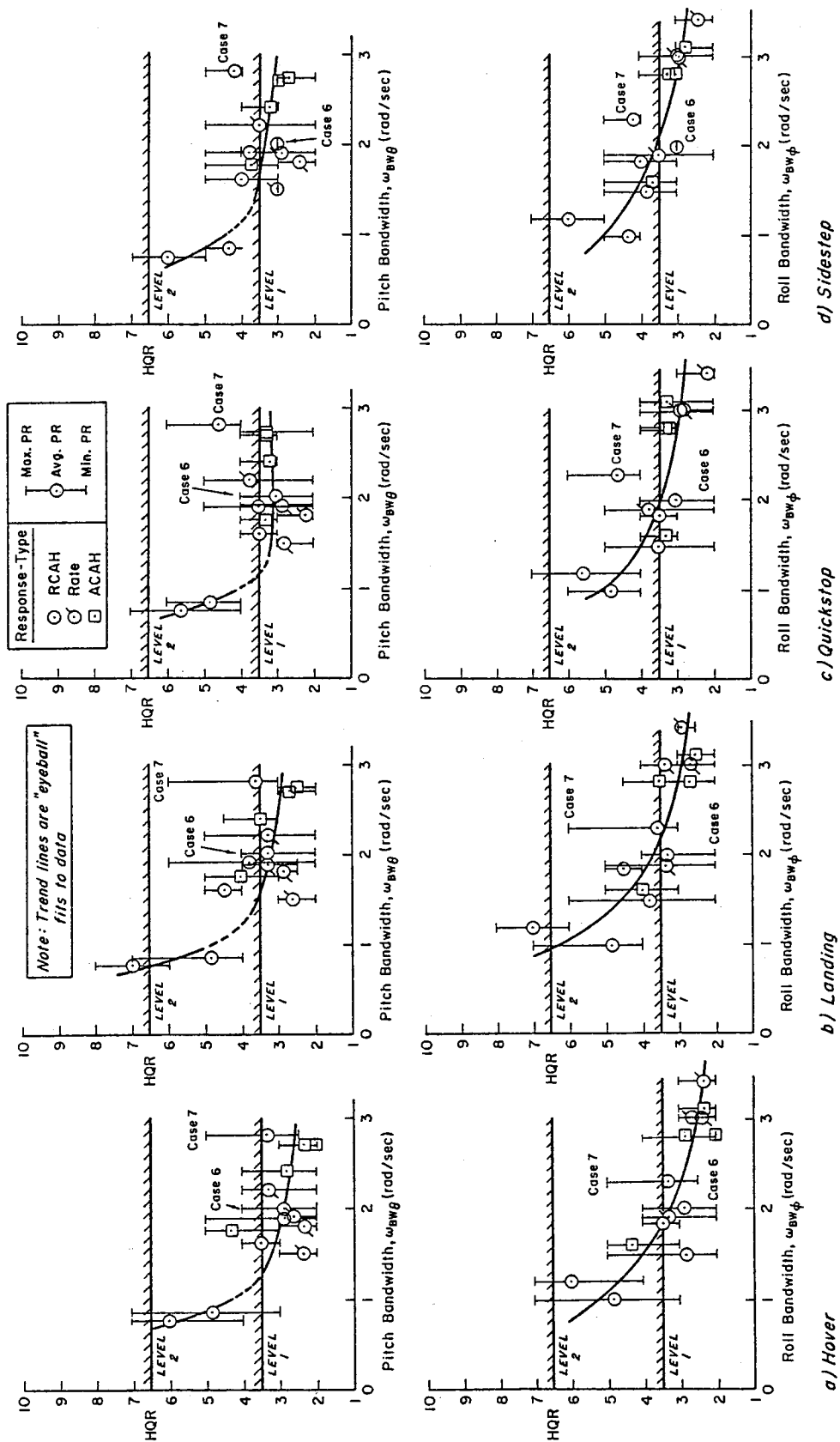


Figure 10(3.3.2.1). Pilot Rating vs. Pitch and Roll Bandwidth (Conventional Controller)

on Figure 10, a pitch bandwidth as low as 1 rad/sec is Level 1, while the Level 1 limit in roll is about 2 rad/sec.

The data of Figure 10 do not show any significant differences in pilot ratings between the three Response-Types evaluated and it appears that the pilots were, for the most part, indifferent to the specifics of the short- or long-term response. Other studies have shown that ACAH is preferred over RCAH for operations in low visibility (e.g., Reference 156), and that RCAH is preferred over ACAH for air combat (e.g., Reference 155), but the current results indicate that for low-speed, maneuvering tasks accomplished in good visibility, such differences do not exist.

Comparison of all of the average-pilot-rating curves in Figure 10 indicates that the precision hover task was the easiest overall (i.e., the average ratings are better), while the landing task was the most difficult. The fact that landing was more difficult than precision hover is consistent with the Reference 4 simulation results; it is interesting, however, that the quickstop and sidestep tasks were also somewhat easier than the landing task. In general, there are only slight differences in the average-rating curves for all four of the tasks.

As Figure 10 shows, bandwidths considerably less than 3 rad/sec (the Level 1 limit suggested by the simulation results of Figure 7) were acceptable in flight. In addition, there is a difference in the variation of pilot rating with bandwidth for pitch versus roll; Figure 10 suggests that the pilots were more sensitive to roll bandwidth. Based on Figure 10, a pitch bandwidth as low as 1 rad/sec is Level 1, while the Level 1 limit in roll is about 2 rad/sec.

Two of the cases in the NRC flight experiment (Cases 7 and 8) were included specifically to investigate whether high levels of pitch and roll rate overshoot resulted in handling qualities problems. The pilot ratings for Case 7 are particularly insightful. Figure 11 shows the individual pilot ratings from the five pilots for Case 7 as a function of task. This case was acceptable to four of the five pilots for the hover and landing tasks. For the quickstop and sidestep, all five pilots considered it to be Level 2. The ratings from Pilot 3 were consistently higher, and his comments reflect an objection to the abruptness of the response, even at low control sensitivities. Other pilots also noted the abrupt response, and considered it more an annoyance than a handling qualities deficiency. On this basis, it was decided not to place a limit on angular rate overshoot, and to note here that excessive overshoot (order of 1.8) is objectionable. The "drop-back" characteristics of this case may also have been a negative factor (see Paragraph 3.2.5 for a discussion of drop-back).

#### Results for Integrated Four-Axis Sidestick

The following discussion reviews sidestick results from the standpoint of bandwidth requirements. A comparison of conventional vs. sidestick control results is given in the discussion for Paragraph 3.6.2.

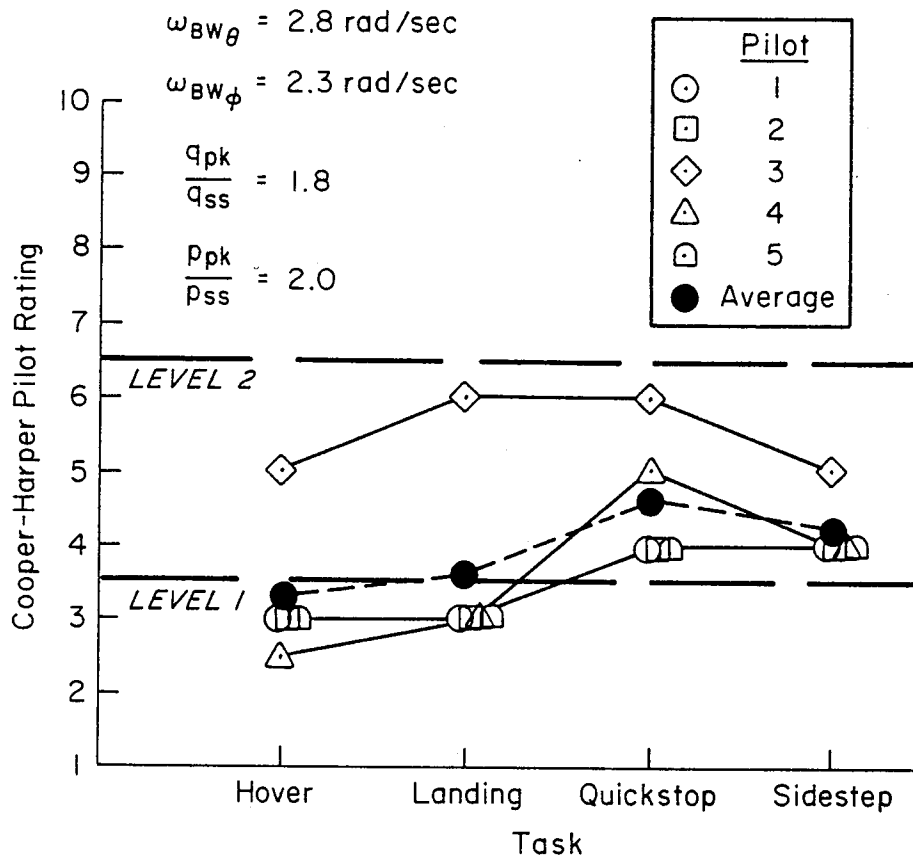


Figure 11(3.3.2.1). Pilot Ratings for High Overshoot Configuration (Case 7) -- Conventional Controller

The four-axis small-displacement sidestick controller used in the Reference 157 experiments utilized twist for yaw control and vertical force commands for heave control. The response dynamics in the heave axis were inherently deficient for the four-axis sidestick.\* Because of the inadequacies in the vertical axis, and because the intent of the experiment was to study pitch and roll response requirements, the pilots were instructed to ignore any minor shortcomings in height control in their ratings. Informal discussions indicated that the height control for landing was an HQR of 4 to 4.5 with this controller.

Results for the precision hover and landing tasks, with the four-axis sidestick, were very similar to those for the conventional controls and are not shown here. The most significant differences were found for the dash/quickstop and lateral sidestep tasks. A summary of pilot ratings as a function of pitch and roll bandwidths for these tasks is shown in Figure 12. There is a significant difference in pilot ratings for the three Response-Types. The Rate configurations were

\*Since large vertical displacements are not possible with a sidestick, it is necessary to integrate its output to avoid excessive abruptness in the response. This additional integration results in an acceleration response ( $K/s^2$  in the frequency domain) which is undesirable, and would be Level 2 by the requirement in Paragraph 3.3.10.1.



preferred, while the ACAH configurations were all rated poorly, i.e., all received Level 2 average ratings. These tasks involve maneuvering, and the poor ratings are attributed to the inherent lack of agility of ACAH Response-Types, combined with sidestick geometry problems. In the conventional- control evaluations, Figures 10c and 10d, only one ACAH configuration was considered to be Level 2.

For a multi-axis task like the dash/quickstop, inputs are required into every axis essentially simultaneously. With ACAH this can result in very uncomfortable and difficult control forces and hand/arm positions which may explain why ACAH was a degradation in Response-Type for the four-axis sidestick.

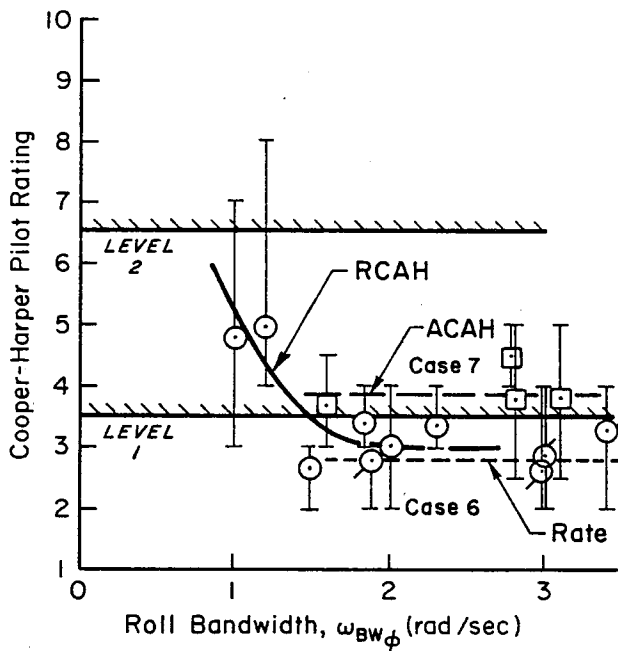
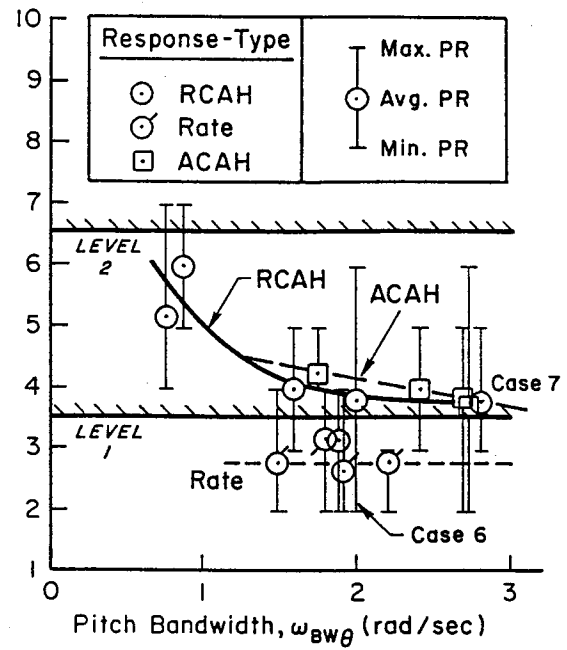
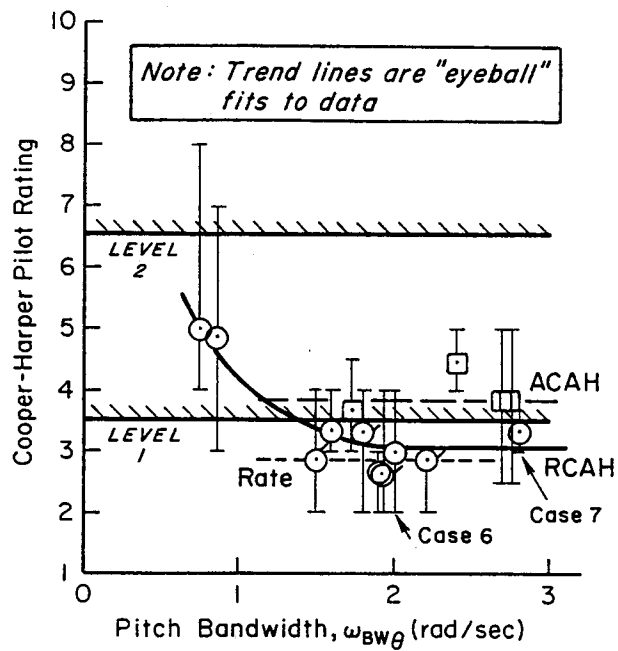
Pitch rate overshoot was not found to be as troublesome for the sidestick evaluations as it was for the centerstick. Case 7, with overshoot ratios of 1.8 in pitch and 2.0 in roll, received pilot ratings that were very close to those for Case 6, which had no overshoot.

#### Development of Boundaries for Fully Attended Operations, UCE = 1

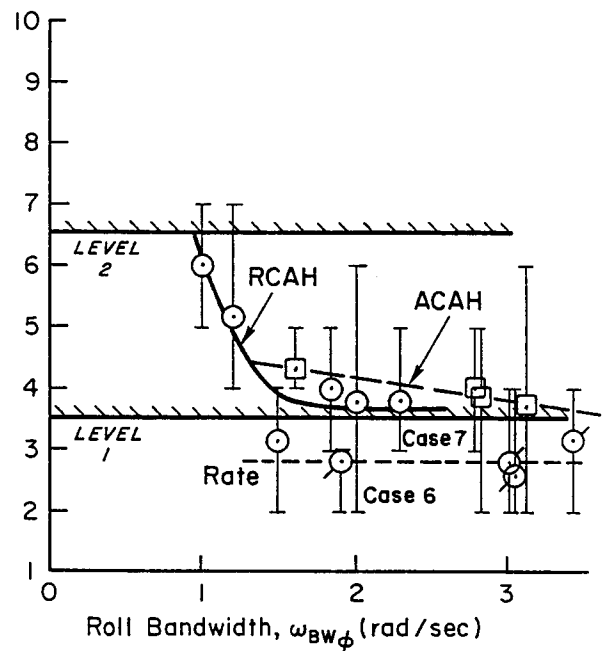
The limits on Bandwidth and phase delay for fully attended operations in good visual cue conditions (UCE = 1), Figures 1c(3.3) for pitch and 1d(3.3) for roll, were developed entirely from the NRC flight test data. Since the HQRs were given for overall tasks, and the Bandwidth parameters were varied in pitch and roll simultaneously (Table 1), it is difficult to determine whether a particular HQR is due primarily to pitch dynamics, roll dynamics, or a combination. Therefore, some assumptions were required in analyzing these data: 1) the quickstop was essentially a longitudinal task, and therefore the HQRs for this task were dominated by pitch Bandwidth and phase delay; 2) the sidestep was a lateral task, and the HQRs for this task were dominated by roll Bandwidth and phase delay; 3) the hover and landing tasks were intrinsically multi-axis, and therefore the HQRs for these tasks were influenced by both pitch and roll dynamics; and 4) for all tasks the HQRs were not adversely influenced by heave or yaw dynamics (especially for the runs with conventional cockpit controllers).

Based on these assumptions, it is reasonable to use the HQRs from the quickstop (longitudinal) task to set limits on pitch Bandwidth and phase delay, and the HQRs from the sidestep (lateral) task for roll Bandwidth and phase delay, with the ratings from the most stringent multi-axis task (landing) applied only as a final check of the boundaries.

Figure 13 shows the data for the quickstop task. The actual HQRs from the five pilots are noted beside each symbol. The Level 1 boundary of Figure 1c(3.3) is taken from this figure. Only four cases received Level 2 average HQRs; two of these cases lie in the Level 1 region, though one is Case 7 (HQRs of 4/4/6/5/4, average 4.6), which may have been Level 2 due to high pitch rate overshoot; the other is Case 12 (HQRs of 2/4/5/4, average 3.75), which is the only pure Rate Response-Type that also has low roll Bandwidth (Table 2).



a) Quickstop



b) Sidestep

Figure 12(3.3.2.1) Pilot Ratings for 4-Axis Sidestick for Dash/Quickstop and Sidestep

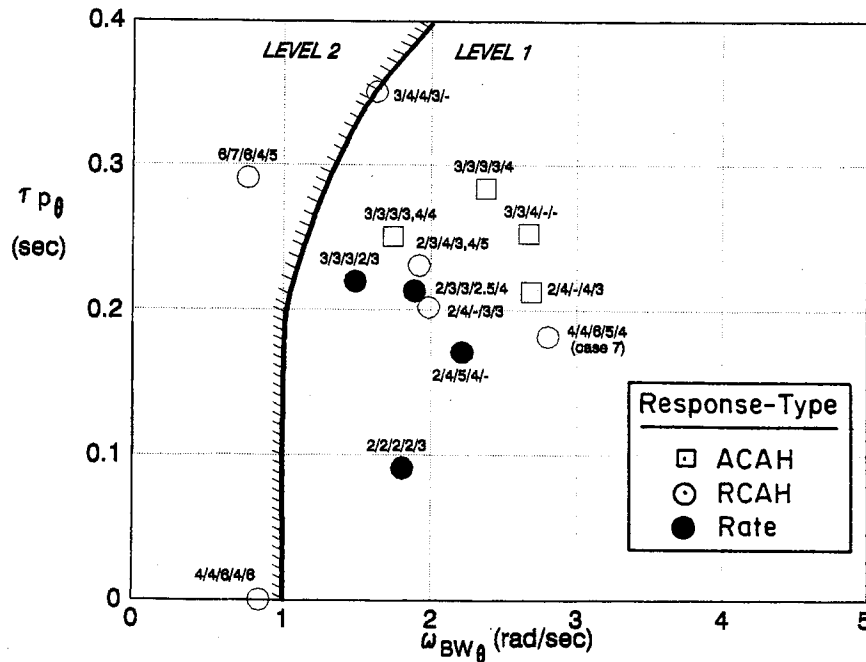


Figure 13(3.3.2.1). HQRs for Quickstop with Conventional Controller from Flight Tests of Reference 157 (Basis for Pitch Requirements of Figure 1b(3.3))

Data for the sidestep task are plotted in Figure 14. These data are the basis for the roll Bandwidth requirements of Figure 1d(3.3). Of the seven cases in the Level 1 region, only one (the high-overshoot Case 7) received Level 2 HQRs. One of the seven configurations in the Level 2 region (Case 6, with HQRs of 3/3/3/3) received a Level 1 average HQR. There are no Level 3 data; the Level 3 boundary is based in part on the low-Bandwidth, high-delay case that received two HQRs of 7, and on ratings for the SCAS-off XV-15, discussed below.

As a final check of the Figures 1c(3.3) and 1d(3.3) limits, Figure 15 compares the average HQRs for both conventional (Figures 15a and 15b) and sidestick (Figures 15c and 15d) evaluations. Since the landing was an inherently multi-axis task, requiring full-time control of both pitch and roll, we expect the overall HQRs to be somewhat worse (higher numerically) for this task, as they are. For example, one configuration that received an average HQR of 7.0 (Level 3) is in the Level 2 regions for both pitch and roll (Figures 15a and 15b). This illustrates the effect of the combined-axis task, since this case received average HQRs of 5.6 for the quickstop (Figure 13) and 6.0 for the sidestep (Figure 14).

A brief experiment was performed during the flight tests of Reference 157 to determine the effects of added pure time delays on pilot opinion. The results of this mini-experiment (not reported in Reference

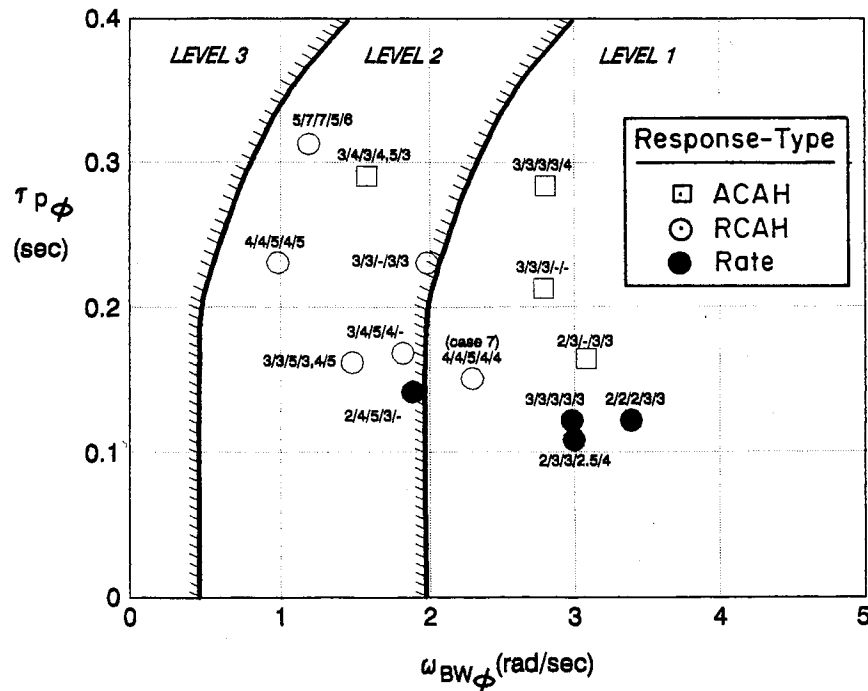


Figure 14(3.3.2.1). HQRs for Sidestep with Conventional Controller from Flight Tests of Reference 157 (Basis for Roll Requirements of Figure 1d(3.3))

157 but shown here in Figure 16), provide further support for the Figure 1(3.3) boundaries. One pilot (Pilot 3) flew the precision hover and landing tasks with the integrated four-axis sidestick with varying levels of time delay added to an RCAH and an ACAH system. The pilot was unaware of the value of time delay, and the configurations were presented in a random order. The ratings support the limits as drawn (Figure 16), again bearing in mind that landing was a multi-axis task by nature.

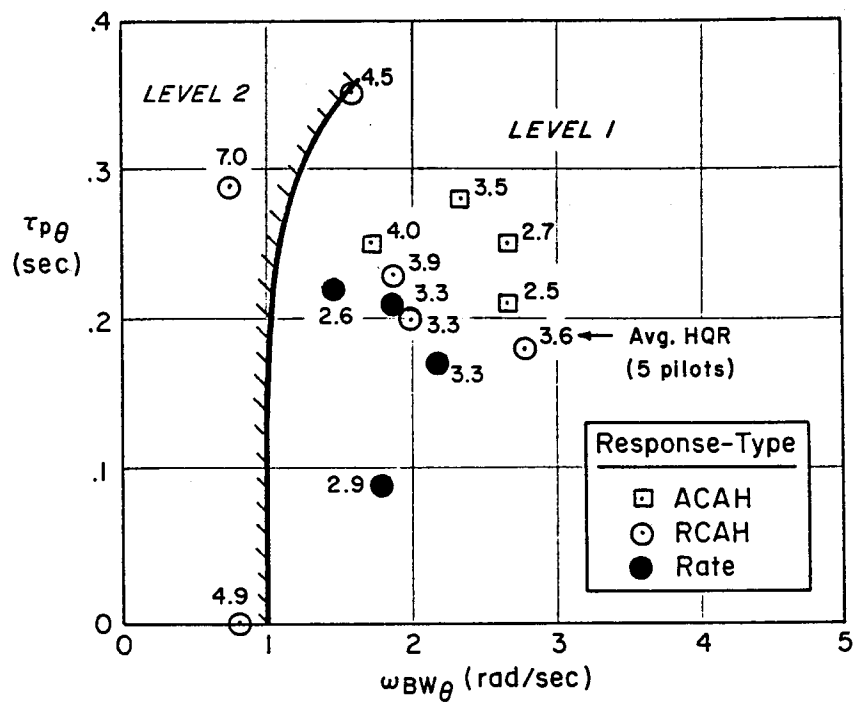
## 2. UH-60 Experiments (Reference 5)

The bandwidth characteristics of several helicopters, including the UH-60, are shown in Figure 15. The UH-60 bandwidth values were determined from flight-derived frequency responses and the ratings are from five pilots for the sidestep task (data from Reference 5). The helicopter received a Level 1 average rating (2.6). It is interesting to note that the roll bandwidth is much higher than the pitch bandwidth.

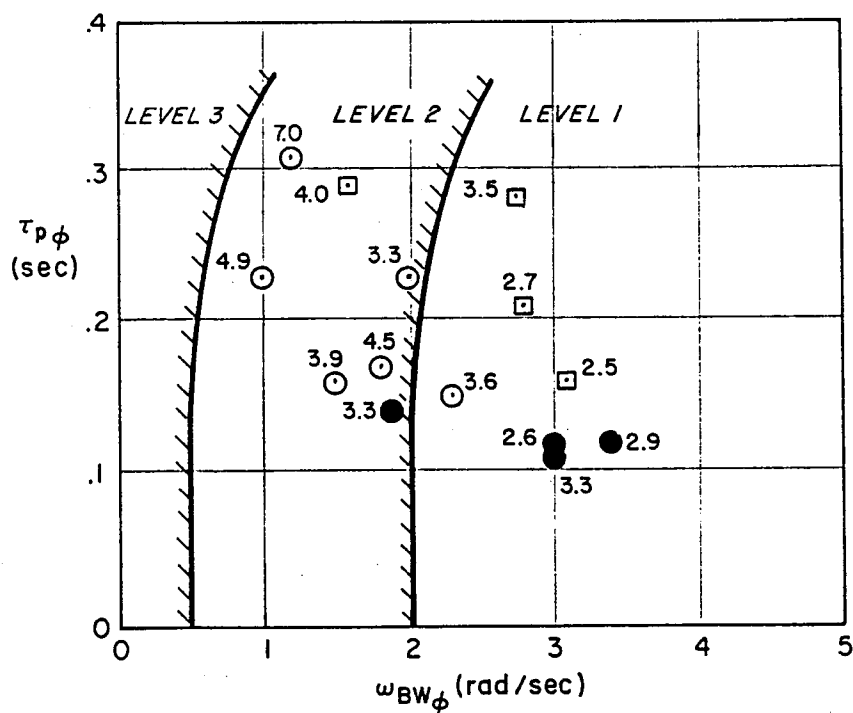
## 3. XV-15 Experiments (Reference 45)

The results of the XV-15 experiments are described in some detail in the discussion for Paragraph 3.2.2.

The XV-15 was flown through hovering tasks with a Rate Command/Attitude Hold SCAS, and with all SCAS off (a short-term Rate, but long-term Acceleration, Response-Type). One pilot flew and rated the tasks and configurations. The relevant ratings are shown in Figure 17.

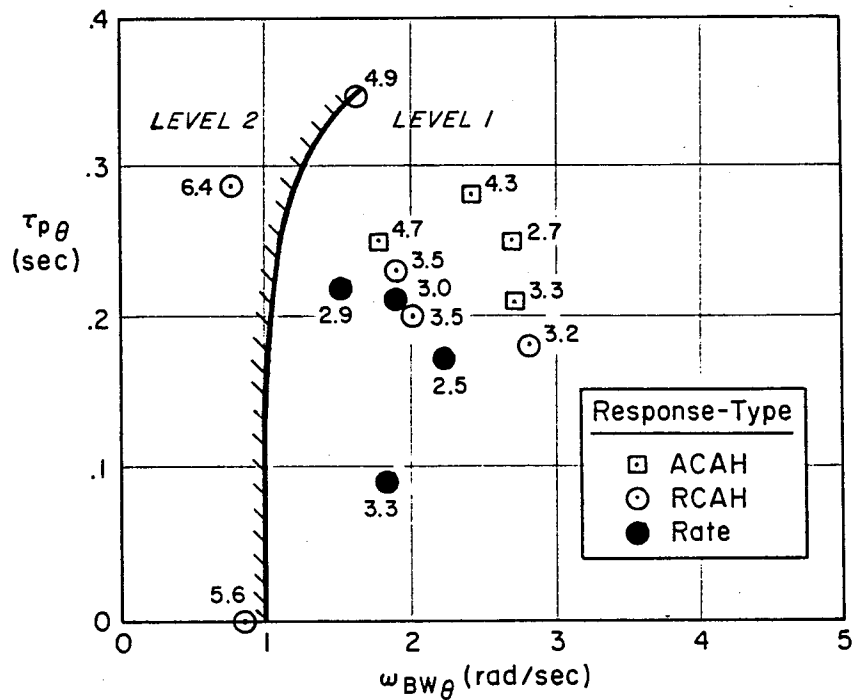


a) Pitch Characteristics (conventional controls)

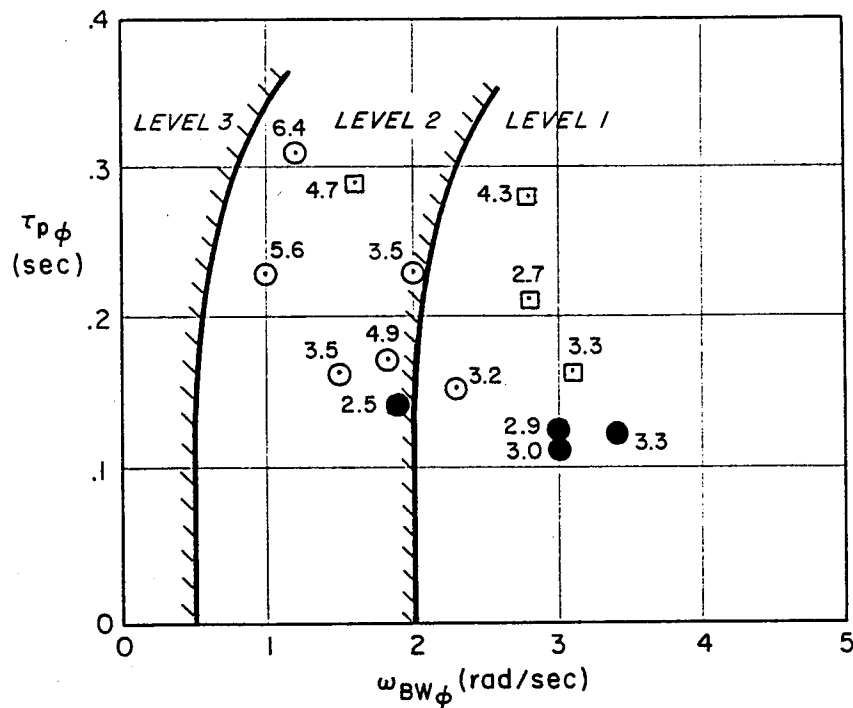


b) Roll Characteristics (conventional controls)

Figure 15(3.3.2.1). Average Pilot Ratings for Landing Task from Flight Experiment (Reference 157) Compared with Requirements

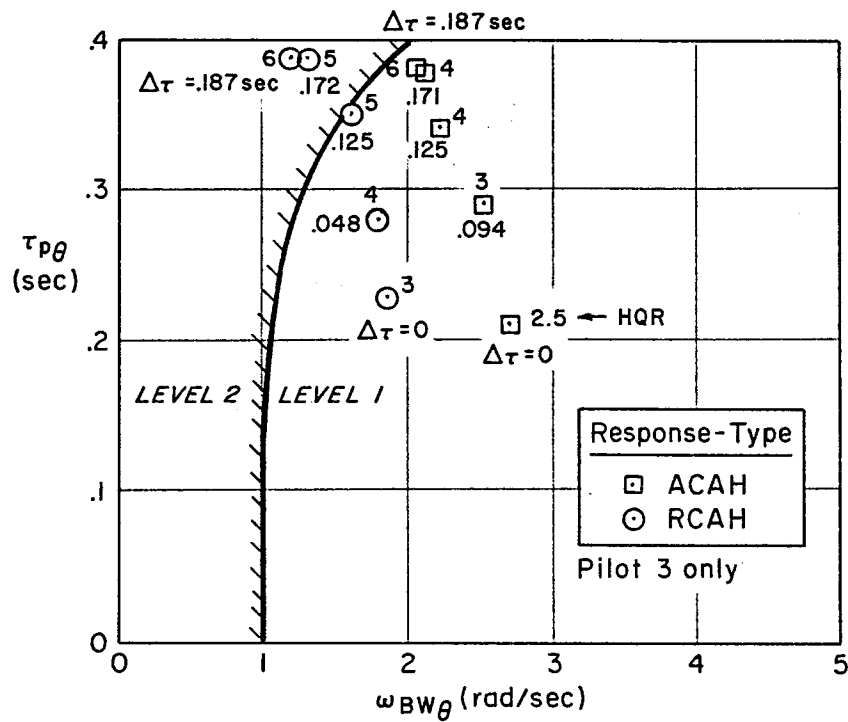


c) Pitch Characteristics (sidestick)

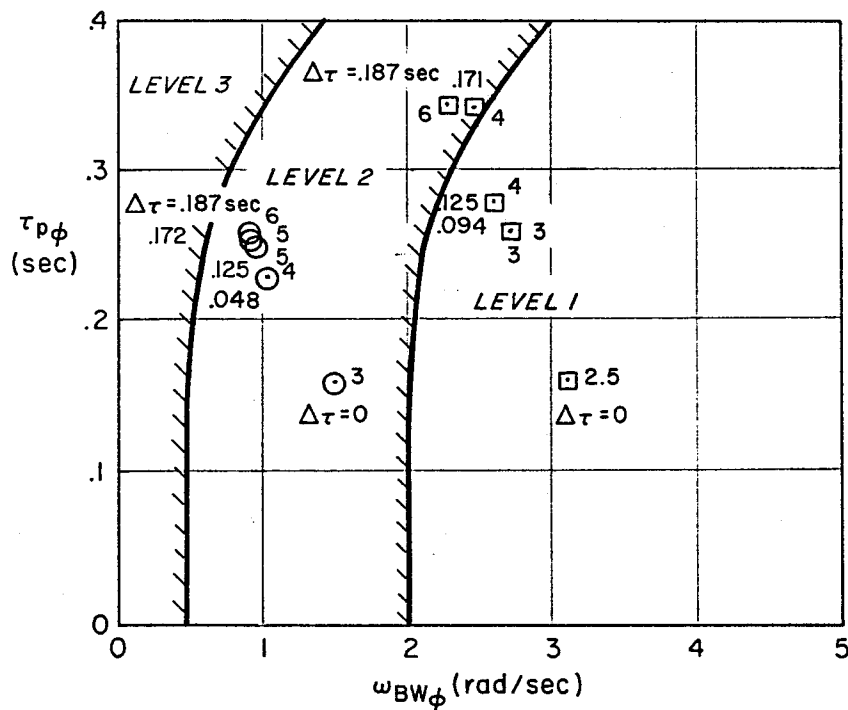


d) Roll Characteristics (sidestick)

Figure 15(3.3.2.1). (Concluded)

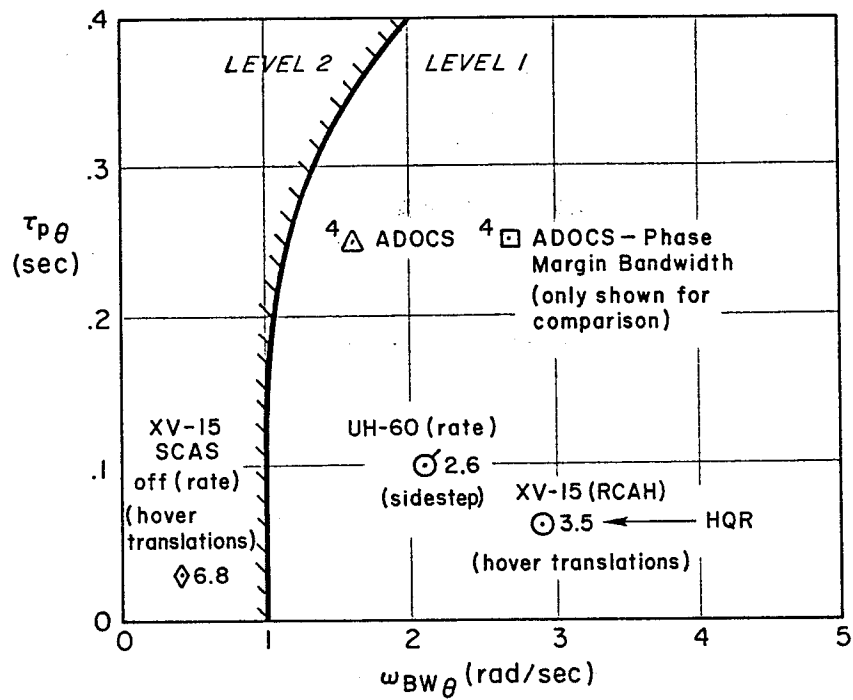


a) Pitch Requirements

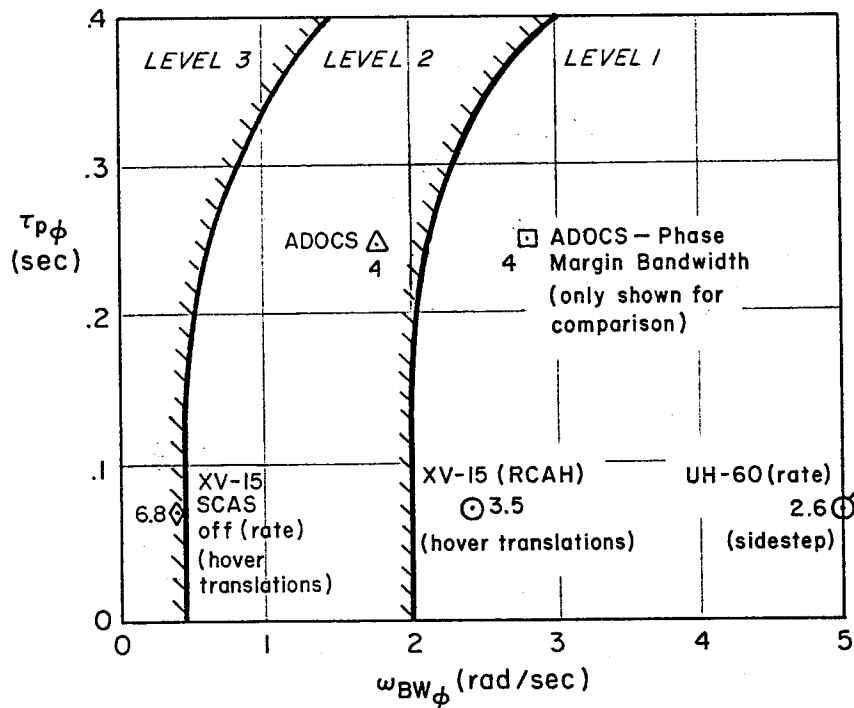


b) Roll Requirements

Figure 16(3.3.2.1). Pilot Ratings from Time Delay Investigation Using NAE Variable-Stability Helicopter. Landing Task; Pilot 3; 4-Axis Sidestick



a) Pitch Requirements



b) Roll Requirements

Figure 17(3.3.2.1). Bandwidth Results from Flight Experiments for UH-60 (Reference 5), XV-15 (Reference 45), and Boeing Vertol ADOCS Model (Reference 170)



The XV-15 has almost symmetric pitch and roll bandwidth characteristics. With SCAS on, the pilot rated it a 3 for fore/aft hover translation and a 4 for sideward hover translation. With the SCAS off, these ratings were 6.5-7 and 7. As Figure 17 shows, the ratings are consistent with the boundaries. The rating of 6.5-7 for pitch suggests that there should be a Level 3 pitch limit as well; as the NRC data of Figure 10 showed, however, it is difficult to determine the relative importance of pitch and roll bandwidth when both are very low.

#### 4. ADOCS Demonstrator Tests (Reference 170)

A UH-60A Black Hawk helicopter, converted to an Advanced Digital Optical Control System (ADOCS) demonstrator, has been developed by Boeing Vertol as a flying testbed. Preliminary handling qualities evaluations of this helicopter, utilizing a three-axis sidestick (pitch, roll and yaw) with a separate left-hand sidestick collective controller, are reported in Reference 170.

Handling qualities ratings (HQRs) were obtained from three pilots for low speed and hover tasks as shown in Figure 18 (taken from Reference 170). Pilots B and C flew an earlier version of the system which was an ACAH Response-Type, and had a very low frequency trim followup with no separate trim control for pitch and roll. Pilot A flew a later version with a significantly increased trim followup rate which causes the Response-Type to change to RCAH by the definitions in this specification. This is discussed, in considerable detail, in the backup material for Paragraphs 3.2.5 and 3.2.7. All of the tasks in Figure 18 involve fully attended operations in conditions of good visual cueing so the Response-Type (RCAH or ACAH) should not have an effect on the ratings (HQRs). Indeed, the rating spread in Figure 18 is consistent with most handling qualities flight testing. According to Reference 170, the liftoff and landing tasks had control mode switching problems during the Pilot B and C evaluations which would explain their poor ratings for these tasks. The ratings for the more demanding tasks (right side of Figure 18) average about 4, whereas the less demanding tasks are solid Level 1.\* The Level 2 ratings are consistent with the specification boundaries for roll, and indicate that the pitch boundary may be slightly lenient (see Figure 17).

The response is gain-margin-limited in both axes ( $\omega_{BW_{\text{gain}}} < \omega_{BW_{\text{phase}}}$ ), and the increase in bandwidth that would occur if only the phase margin limit were imposed is shown in Figure 17. As discussed earlier, such gain margin limiting would be cause to suspect the possibility of PIO tendencies for very tight tracking tasks (e.g., precision slope landing on a gusty day, or aggressive closed loop pilot technique). It is also important to note that if the Response-Type were labeled ACAH instead of RCAH, only the phase margin definition would apply. In fact, the final ADOCS is referred to as an Attitude system in

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\*The lateral jink is an exception to this conclusion as it is generally considered to be a demanding maneuver, and is rated solid Level 1.

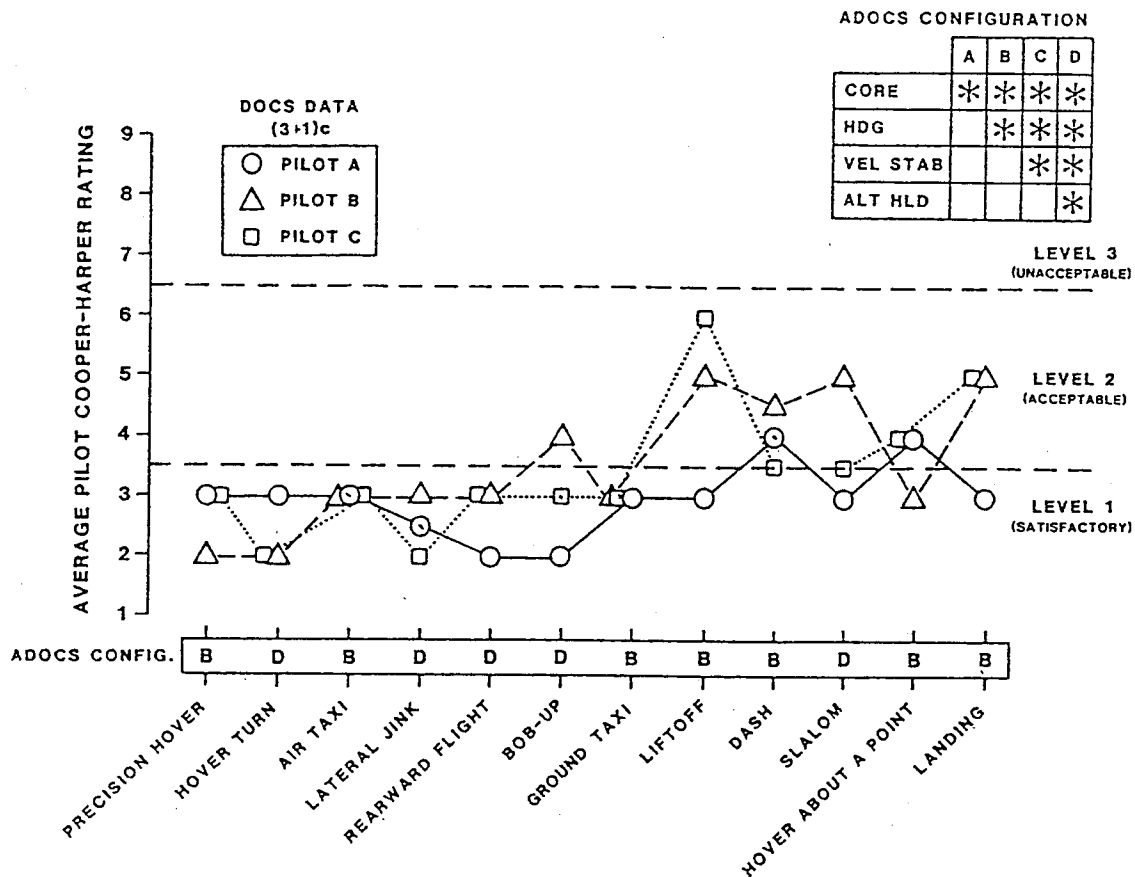


Figure 18(3.3.2.1). Evaluation of ADOCS Demonstrator Handling Qualities (from Reference 170)

Reference 170, probably because attitude feedback is a dominant feature of the SCAS. Nonetheless, it is a RCAH Response-Type by the definitions in this specification for the reasons discussed in the backup material for Paragraphs 3.2.5 and 3.2.7. Of course, the tendency for PIO is unrelated to the label we choose to put on the Response-Type, and is always a factor when  $\omega_{BW\text{gain}} < \omega_{BW\text{phase}}$ . This is discussed in detail in subsection 4 of Section b. Flight testing is currently planned to investigate the response characteristics of ADOCS for aggressive and precise tasks designed to expose any tendency for PIO.

#### DATA FOR CURRENT HELICOPTERS

Bandwidths of current operational helicopters were obtained from a variety of sources for comparison with the limits of Figures 1c(3.3) and 1d(3.3). Most of the bandwidth calculations were based on analytical models, although a small amount of flight data was available. Figure 19 summarizes this data in the form of crossplots of pitch and roll bandwidth versus airspeed. Data for more rapidly responding rotorcraft

such as the OH-6 and BO-105 are not shown because the rotor system time delay strongly affects the bandwidth and is not known with sufficient accuracy.

Open symbols on Figure 19 represent bandwidths calculated from analytical models of the helicopters. In these cases, a pure time delay of 125 ms was added to represent the effects of rotor dynamics and higher-order lags in the control system ( $\tau = 125$  ms was measured on the NRC Bell 205A).

Bandwidths obtained from flight data are indicated in Figure 19 by solid symbols. Frequency responses are available for the UH-60 references and the Bell 214ST (Reference 117), generated from frequency sweeps. The numbers for the UH-1H are based on maneuver performance plots of  $p_{pk}/\Delta\phi$  in Reference 93 (Supporting Data for Paragraph 3.3.3).

Handling qualities Levels drawn on Figure 19 are based on the bandwidth limits of Figures 1c(3.3) and 1d(3.3) for low values of  $\tau_p$ .

Several observations can be made from Figure 19: 1) most helicopters meet the Level 1 requirements with augmentation; 2) those helicopters that have bandwidths well into the Level 2 region -- i.e., the AH-1G, UH-1H, CH-53D, UH-60, and CH-47B -- all have some form of stability and control augmentation; 3) pitch bandwidth is always lower than roll bandwidth, with the pitch value typically about half the roll value, due to the inherently higher pitch inertia of typical rotorcraft configurations.

While the data of Figure 19 have not been used to define limits on bandwidth, they at least indicate that the bandwidth limits of Figures 1c(3.3) and 1d(3.3) are reasonable.

#### d. Guidance for Application

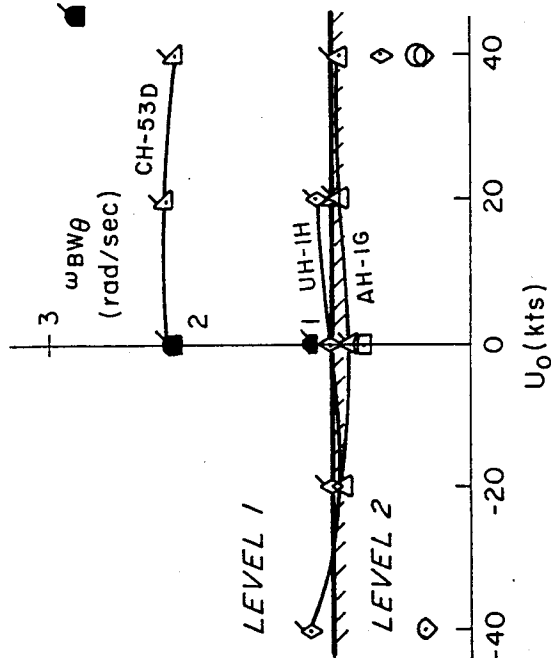
Procedures for obtaining the bandwidth parameters from flight data are documented in Appendix A.

Phase delay ( $\tau_p$ ) is conceptually related to the Lower Order Equivalent System (LOES) time delay parameter used in the fixed-wing military flying qualities specification (Reference 190) since they both describe a similar characteristic of the shape of the phase curve. Because of this, there is a strong tendency to think of phase delay as a pure transport delay in the time domain, and to use the parameters interchangeably. Such interpretations are incorrect, and it is important for the specification user to have a strong physical understanding of the phase delay parameter. The following discussion is intended to assist in such an understanding.

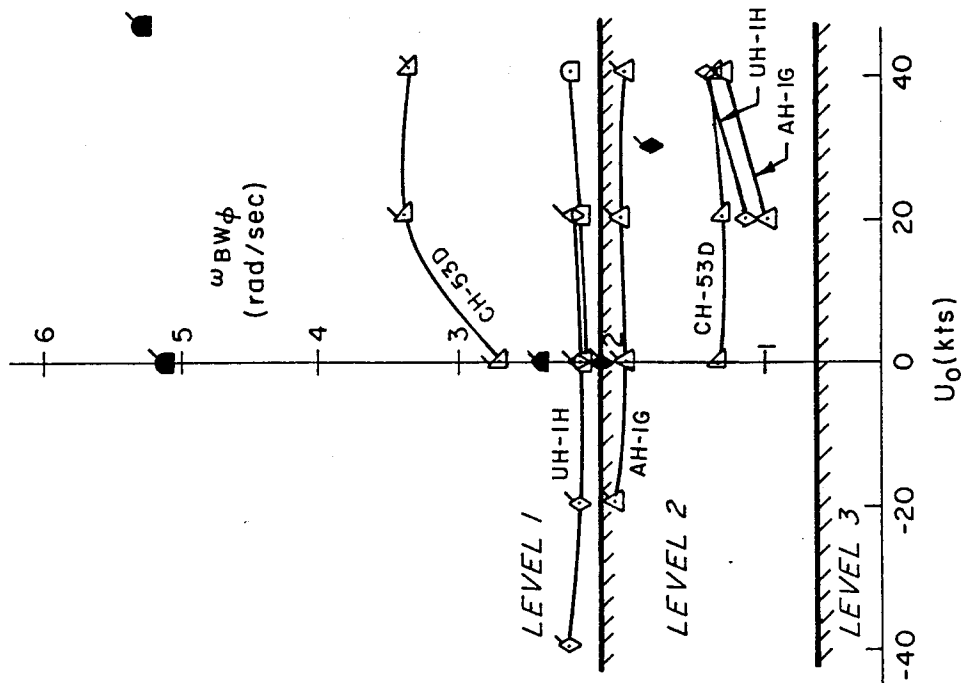
Experience has shown that handling qualities ratings (HQRs) tend to degrade rapidly with what would seem to be small values of phase delay, and hence it is an important parameter. For example, consider the two cases shown earlier in Figure 2 (from Reference 1). These configurations both have essentially the same Bandwidth (approximately

Aircraft	Ref.	
△ AH-IG	28	Analytical Models (derivatives and transfer functions)
◇ UH-IH	28	
△ CH-53D	28	
□ UH-60	145	125 ms delay added to simulate actuators, etc.
◇ CH-47B	146	
◇ UH-60	5	Flight Data (frequency sweeps)
■ Bell 214ST	117	
◇ UH-IH	93	Flight Data ( $P_{pk}/\Delta\phi$ )

Note: Flagged symbol indicates SCAS on



a) Pitch Bandwidth



b) Roll Bandwidth

Figure 19 (3.3.2.1). Pitch and Roll Bandwidths of Representative Helicopters at Low Speed

2.6 rad/sec), but one has significantly higher phase delay (0.014 sec vs. 0.17 sec), and significantly degraded pilot ratings (average rating of 4.1 vs. 8). It seems intuitively unlikely that this drastic degradation in pilot rating can be attributed to a 0.15 sec shift in the time response, or to the phase shift around the crossover frequency which is seen to be negligible (Figure 2). However, the shape of the phase curve is drastically steeper above the Bandwidth frequency, a factor which intuitively would be expected to lead to a significant difference in pilot rating. This is the only factor which yields a plausible explanation of the strong sensitivity of pilot rating to phase delay (and similarly to equivalent system time delay). This is important because it reveals the fact that phase delay is significant only because it is a measure of the shape of the phase curve around the instability frequency, not because of the transport delay properties in the time domain. Paradoxically, criteria which measure "time delay" in the time domain are of questionable validity. That is, the measurement of a transport delay resulting from a step input is sensitive to all the wrong things, e.g., minor variations in the shape of the input, initial conditions, etc. Because of this, such measurements may not be an accurate representation of the shape of the phase curve, which is the root cause of the problem. Likewise equivalent system time delay depends strongly on the form of the LOES, and therefore is not an independent measure of the shape of the phase curve, and consequently should not be used interchangeably with phase delay.

#### e. Related Previous Requirements

The longitudinal dynamic stability requirements of MIL-H-8501A, Paragraphs 3.2.11 and 3.6.1.2, express stability in terms of periods, cycles, and times to half or double amplitude for oscillatory modes. In addition, response characteristics at hover are specified in MIL-H-8501A as functions of moments of inertia (Paragraph 3.2.14 for longitudinal, 3.3.19 for lateral). The latter requirements effectively limit the minimum values of pitch and roll damping, i.e.,

$$-\frac{\partial M}{\partial q} \geq 8 (I_y) \cdot 7 \text{ ft-lb/rad/sec}$$

$$-\frac{\partial L}{\partial p} \geq 18 (I_x) \cdot 7 \text{ ft-lb/rad/sec}$$

For conventional helicopters in hover,  $M_q$  (or laterally,  $L_p$ ) describes the dominant pitch (roll) response dynamics, i.e.,

$$\frac{\theta}{\delta_B} = \frac{M\delta_B s e^{-\tau s}}{\left[s^2 - \frac{g M_u}{M_q}\right] (s - M_q)} = \frac{M\delta_B e^{-\tau s}}{s (s - M_q)}$$

Thus  $M_q$  and  $\tau$  determine the bandwidth of the response. With augmentation,  $M_q$  is increased in value, increasing the bandwidth frequency. Therefore, limits on bandwidth effectively specify the minimum augmented value of  $M_q$  for a fixed value of  $\tau$ .

The dramatic effect of  $\tau$  on bandwidth as  $M_q$  is increased is shown in Figure 20. Note that for large  $M_q$  a small increase in  $\tau$  results in a large decrease in bandwidth.

It is generally agreed (e.g., References 34 and 51) that the variation of required damping with moment of inertia (as shown in the above inequalities) is not appropriate; Figure 21, from Reference 51, illustrates the variation of  $M_q$  with  $I_y$ . Larger helicopters (such as the CH-53D) may typically not require high response bandwidth, but this is normally a function of mission, not size.

Reviewers with experience in preliminary design of helicopters have expressed concern that the familiar pitch (roll) damping vs. control sensitivity and/or control power boundaries no longer appear in the specification. These boundaries have historically been derived from MIL-H-8501A criteria as shown in Figure 22, taken from Reference 191 (Engineering Design Handbook, Helicopter Engineering, Part One-Helicopter Design, pages 6-30). As noted above, bandwidth is the augmented airplane equivalent of pitch (roll) damping, and indeed  $M_q$  and  $L_p$  are the bandwidths for an unaugmented helicopter for  $\tau = 0$ .

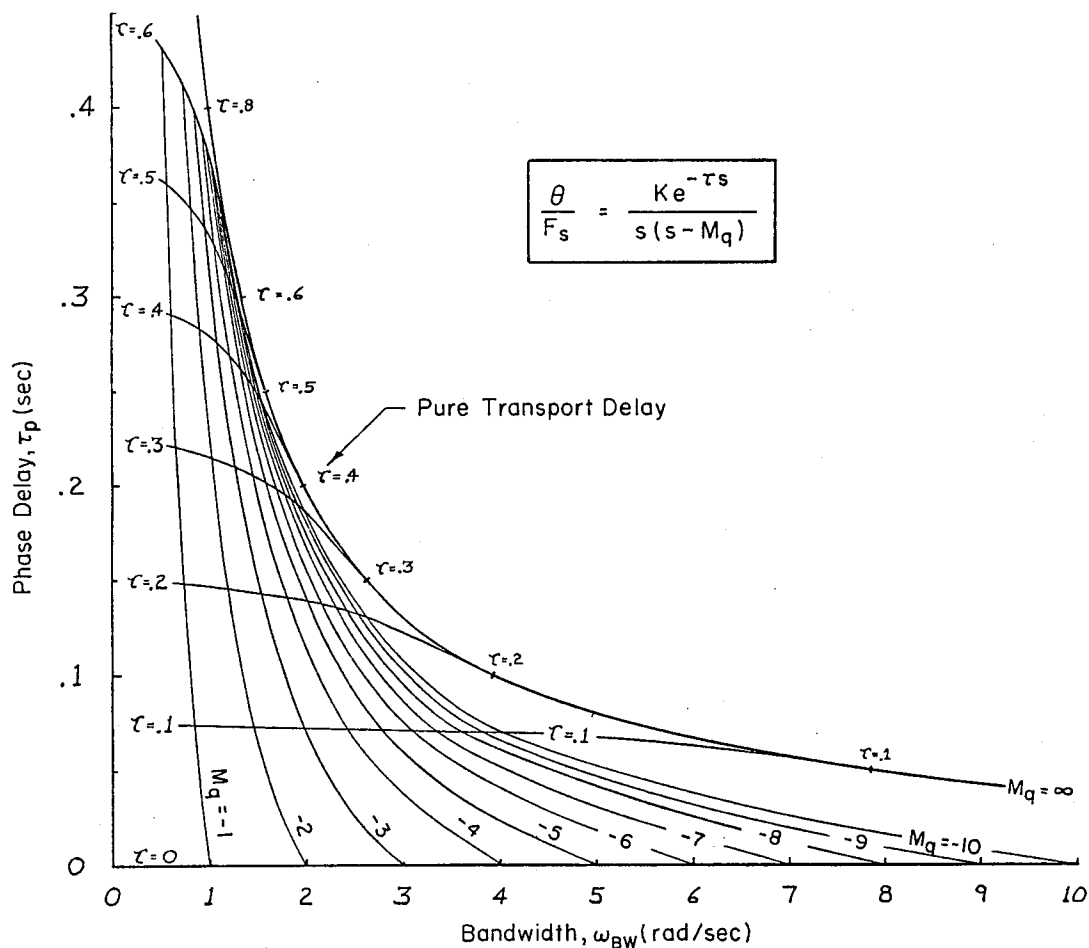


Figure 20 (3.3.2.1).  $\tau_p$  vs.  $\omega_{BW}$  For First-Order Rate Response-Types ( $\omega_{BW}$  Determined from Phase Plot, i.e., Not Gain Margin Limited)

However, it is incorrect to continue to use even the augmented values of these derivatives because the total response includes the effects of other modes and rotor system time delays that, when lumped together, result in an overall  $\tau_p$ . Note that for even moderate values of  $\tau_p$  and  $M_q$ ,  $\omega_{BW}$  is not well approximated by  $M_q$ .

Given that Bandwidth is a necessary replacement for angular damping, why not plot it vs. control sensitivity? The problem is that the control derivative ( $M/I$  in Figure 22) is not a valid measure of control sensitivity for a highly augmented helicopter. The key factor is the gain of the response in the region of piloted crossover. For conventional dynamics, the control derivatives are a measure of the magnitude of the response at all frequencies, including the region of crossover (as sketched in Figure 23a, noting that  $M_q$  is a factor in the magnitude of the Bode asymptotes at all frequencies). For an augmented aircraft, the magnitude of the response in the region of piloted crossover is not uniquely dependent on the control derivatives as it also depends on the values of the flight control system parameters, e.g.,  $K_q$  and  $T_q$  in Figure 23b. Therefore, a proper definition of control sensitivity for augmented helicopters would be the magnitude of the response in the region of crossover, say at the bandwidth frequency. Using that definition, the proper equivalent of the old damping vs. control sensitivity boundaries would be bandwidth vs. magnitude at the

#### MIL-H-8501A Requirement:

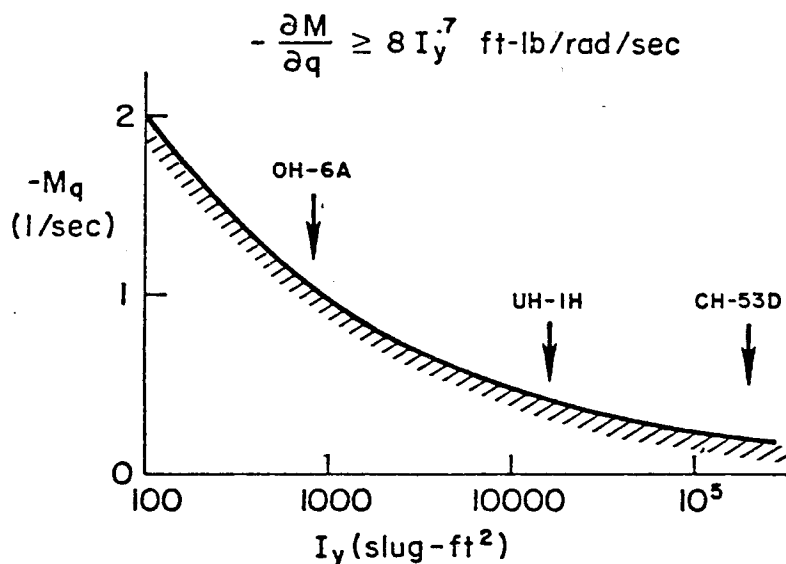


Figure 21(3.3.2.1). Pitch Damping Requirement from MIL-H-8501A (from Reference 51)

bandwidth frequency. Unfortunately, there is insufficient data from highly augmented aircraft to draw such boundaries. As with the old boundaries (e.g., Figure 22), the required control sensitivity will increase with increasing bandwidth (based on fixed wing experience, see Reference 192).

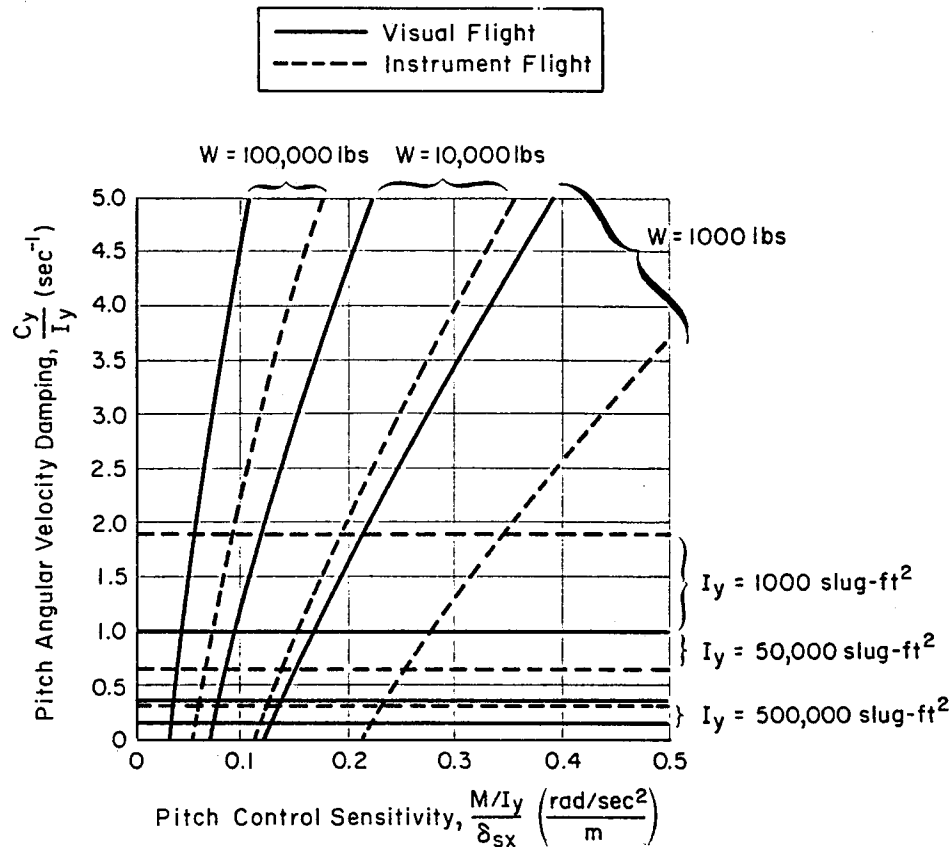
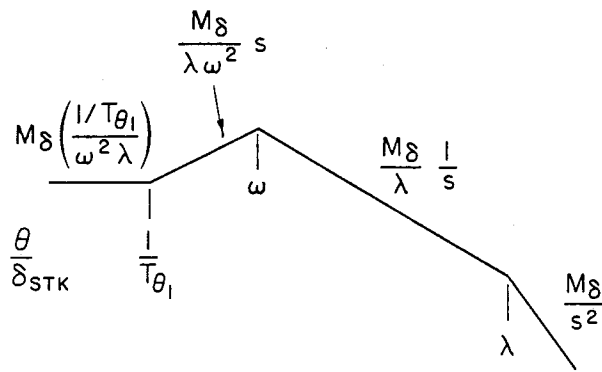


Figure 22(3.3.2.1). Minimum Control Sensitivity and Damping Requirements in Pitch from MIL-H-8501



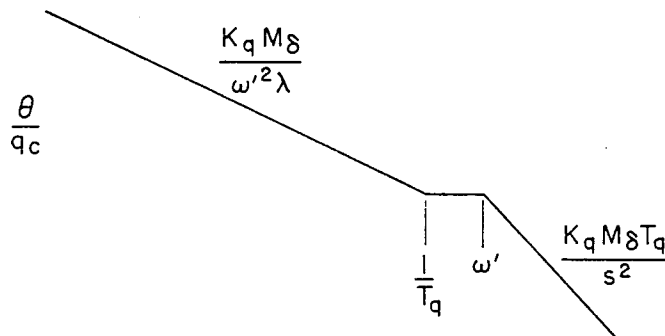


- Pitch acceleration,  $M_{\delta}$ , is a measure of the gain of the response at all frequencies

- Control sensitivity is

$$\left| \frac{\theta}{\delta_{STK}} \right|$$

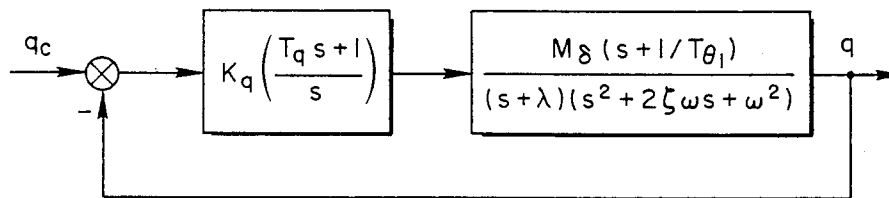
a) *Unaugmented Helicopter*



- The gain of the response in a given frequency range depends on  $M_{\delta}$ ,  $T_q$ , and  $K_q$

- Control sensitivity is

$$\left| \frac{\theta}{q_c} \right|$$



b) *Augmented Helicopter*

Figure 23(3.3.2.1). Illustration of the Difference Between Control Sensitivity for Augmented and Unaugmented Helicopter in Hover

a. Statement of Requirements

3.3.2.2 Mid-Term Response to Control Inputs. The mid-term response characteristics apply at all frequencies below the bandwidth frequency obtained in Paragraph 3.3.2.1. Use of an Attitude Hold Response-Type constitutes compliance with this paragraph, as long as any oscillatory modes following a pulse controller input have an effective damping ratio of at least 0.35. If attitude hold is not available, the applicable criterion (Paragraph 3.3.2.2.1 or 3.3.2.2.2) depends on the degree of divided attention necessary to complete the Mission-Task-Elements according to Paragraph 3.1.1.

3.3.2.2.1 Fully Attended Operations. For those Mission-Task-Elements specified in 3.1.1 as requiring fully attended operation, the mid-term response shall meet the limits of Figure 3(3.3).

3.3.2.2.2 Divided Attention Operations. For those Mission-Task-Elements specified in Paragraph 3.1.1 as requiring divided attention operations (Paragraph 2.5), the limits of Figure 3(3.3) shall be met, except that the Level 1 damping shall not be less than  $\zeta = 0.35$  at any frequency.

b. Rationale for Requirements

Consideration of divided attention is an essential element of a mission-oriented flying qualities specification. The requirements in this paragraph are based on the premise that the mid-term response characteristics represent the handling qualities most affected by mission requirements for divided attention. Simply stated, the pilot should be able to relinquish control of the helicopter for short periods without encountering significant excursions. If the mid-term response is unstable (as with many present-day helicopters), full attention to aircraft control is required on a continuous basis. Experience has shown that it is possible to conduct day-to-day operations with such helicopters with good pilot acceptance. However, operations which require division of attention, such as flying in IMC conditions, have exhibited a need for a more stable platform.

This requirement is intended to allow existing unstable helicopters to meet the specification, with the caveat that only tasks which do not involve divided attention are to be performed. This is accomplished by requiring the procuring activity to designate each Mission-Task-Element as a "fully attended" or a "divided attention" operation (see Paragraphs 2.5 and 3.1.1). If the MTEs are required to be performed in a divided attention environment, the mid-term responses must conform to Paragraph 3.3.2.2.2, which requires  $\zeta > 0.35$  at all frequencies for Level 1 (this eliminates many current-day unaugmented helicopters). On the other hand, if all of the MTEs can be accomplished with full pilot attention to aircraft control (e.g., other crew members handle the non-control tasks), certain mid-term divergences and oscillations are allowed (i.e.,  $\zeta \geq -0.20$  for  $\omega_n \leq 0.50$  rad/sec). This is covered by Paragraph 3.3.2.2.1, which is based on data obtained to investigate the flying

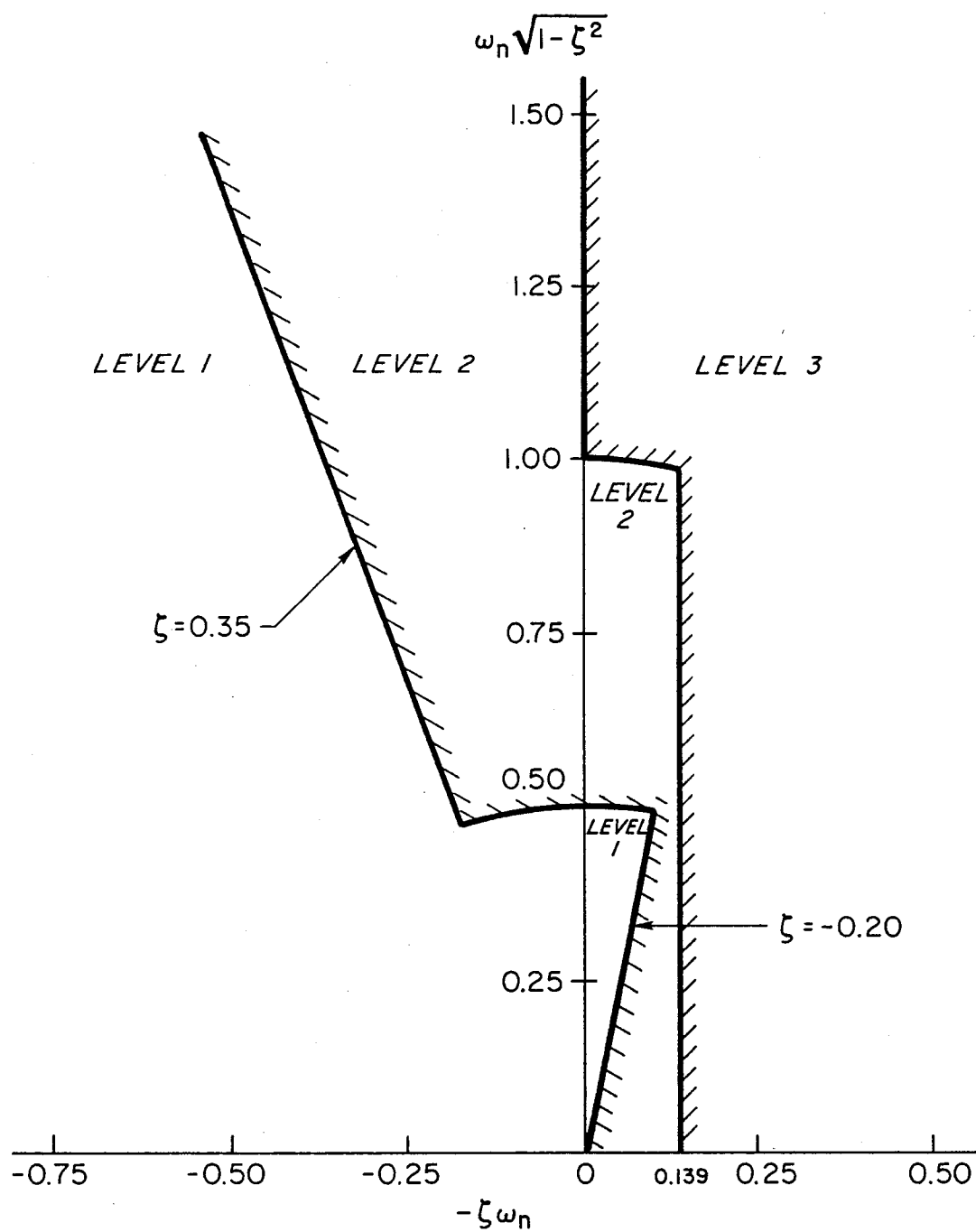


Figure 3(3.3). Limits on Pitch (Roll) Oscillations for Fully Attended Operations -- Hover and Low Speed

qualities of "conventional helicopters," such as were covered by the old MIL-H-8501A.

It is unavoidable that this requirement overlaps that of the short-term response requirements of Paragraph 3.3.2.1. In that paragraph, compliance with the short-term bandwidth limits does not assure mid-term stability, although the measured value of bandwidth is influenced somewhat by low-frequency modes. Conversely, proof that all modes of response exceed the Figure 3(3.3) limits does not guarantee that bandwidth will be Level 1. The intent of the bandwidth requirement is to determine that the vehicle's response around the frequencies to be used for high gain piloted closed-loop control is satisfactory; the intent of this paragraph is to guarantee that the low-gain pilot closed-loop control characteristics are not objectionable.

c. Supporting Data

There is a considerable data base concerning the influences of low-frequency modes on pilot ratings for low-speed operations. Unfortunately, there is a significant amount of scatter between various sources, resulting from differences in Response-Type, tasks, inflight as opposed to ground-based simulation, presence of winds, gust sensitivity, and so on. In addition, none of the studies has included a divided-attention task, i.e., they have all focused on fully attended operations. Most of the available data are reviewed in the Background Information and User Guide for MIL-F-83300 (Reference 30), and in Reference 12.

Three studies (References 142, 143, and 144) directly address the influence of low-frequency oscillations on handling qualities. All of the data from these three sources are presented in Reference 30. After considerable review of the data, it was decided that for the purposes of this specification, only one of these references could be used for development of handling qualities limits. The data of References 143 and 144 have been used only for qualitative interpretations and trends since the two evaluation pilots had private licenses only, and neither had a helicopter rating; the evaluations were performed on a fixed-base simulator with a display that test pilots with rotary-wing background were unable to use; and some of the pilot ratings are inconsistent with other data, especially that of Reference 142, where six qualified test pilots were used.

The discussions that follow will first present the supporting data for Paragraph 3.3.2.2.1 (fully attended operations), including the description of a brief simulation conducted in support of this requirement. The characteristics of current helicopters are then compared with the limits of Figure 3(3.3), and this will lead into a short discussion of the data supporting the more stringent requirement of Paragraph 3.3.2.2.2 (divided attention operations).

## REVIEW OF DATA FROM NORTHROP SIMULATION (REFERENCE 142)

The primary data source for the limits of Figure 3(3.3) was Reference 142. This reference used a moving-base simulator at Northrop Corporation to evaluate a wide range of vehicle dynamics. Evaluations were performed for pitch-axis variations alone, with the roll dynamics fixed at good values, and vice versa; and with both pitch and roll dynamics varied together. Vertical and directional axis responses were held fixed throughout the experiment. A 10-kt steady wind with a 3.4-ft/sec rms gust was included. Control sensitivity was optimized by the evaluation pilot prior to each run. Tasks consisted of vertical liftoff to 30 ft, a constant-heading square pattern, 25- and 40-ft-radius pirouettes, crosswind approaches and forward and sideward quickstops.

For the Reference 142 experiments, the classical hovering cubic was represented (in pitch) by the simple transfer function

$$\begin{aligned}\frac{\theta}{\delta_B} &= \frac{M_{\delta_B} (s - X_u)}{s^3 - M_q s^2 + (M_q X_u - M_{\theta} - X_u) s + M_{u\dot{g}} + X_u M_{\theta}} \\ &= \frac{M_{\delta_B} (s + 1/T_{\theta})}{(s + 1/T_{sp})(s^2 + 2\zeta_p \omega_p s + \omega_p^2)}\end{aligned}$$

The equivalent roll transfer function is

$$\begin{aligned}\frac{\phi}{\delta_A} &= \frac{L_{\delta_A} (s - Y_v)}{s^3 - L_p s^2 + (L_p Y_v - L_{\phi} - Y_v) s - L_{v\dot{g}} + Y_v L_{\phi}} \\ &= \frac{L_{\delta_A} (s + 1/T_{\phi})}{(s + 1/T_R)(s^2 + 2\zeta_d \omega_d s + \omega_d^2)}\end{aligned}$$

The primary variables were  $M_q$  and  $M_{\theta}$  (or, equivalently,  $L_q$  and  $L_{\phi}$ ). Two values of  $X_u$  ( $Y_v$ ) were used, -0.05 and -0.2 rad/sec; and, for most of the evaluations, two values of  $M_{u\dot{g}}$  ( $-L_{v\dot{g}}$ ), 0.33 and 1.0 rad/sec. Figures 1 and 2 show that these values are representative of a wide variety of current helicopters. In addition, Figure 1 indicates that, with the exception of the UH-60 and the CH-53D,  $X_u$  and  $Y_v$  are typically much less than -0.2 rad/sec. For the discussion that follows, only the data from Reference 142 with  $X_u$  and  $Y_v$  of -0.05 rad/sec will be used. Because these derivatives affect gust sensitivity and the amount of attitude required to trim in steady winds (see the discussion for Paragraph 3.3.1), the higher the value, the more sensitive the rotorcraft will be. The high  $X_u$  and  $Y_v$  data of Reference 142 would dictate much more stringent limits than those shown in Figure 3(3.3). Furthermore, a review of the Reference 142 pilot commentary reveals that the primary complaint for large  $X_u$  ( $Y_v$ ) was excessive trim attitudes. It would therefore be misleading to use this data to define dynamic limits in Figure 3(3.3).

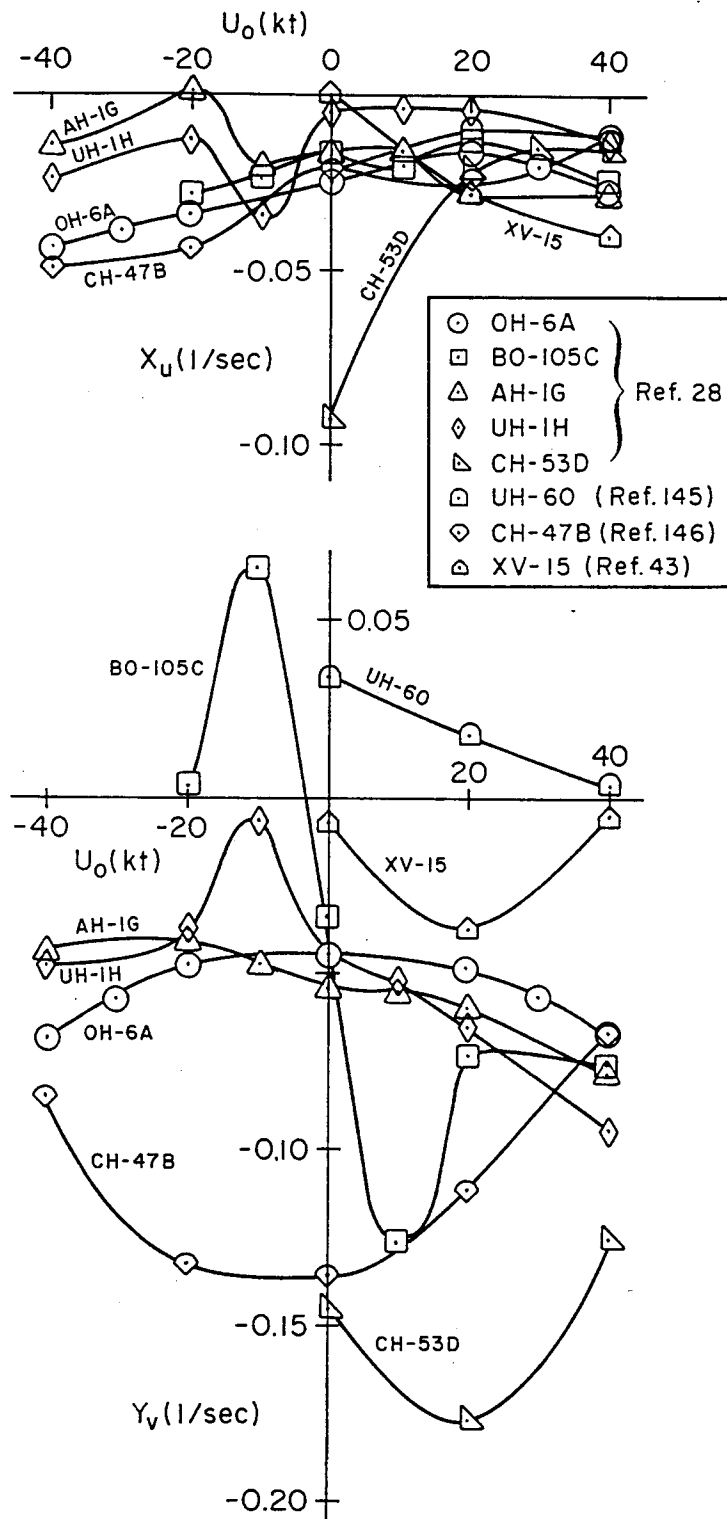


Figure 1(3.3.2.2). Values of the Derivatives  $X_u$  and  $Y_v$  for Several Helicopters

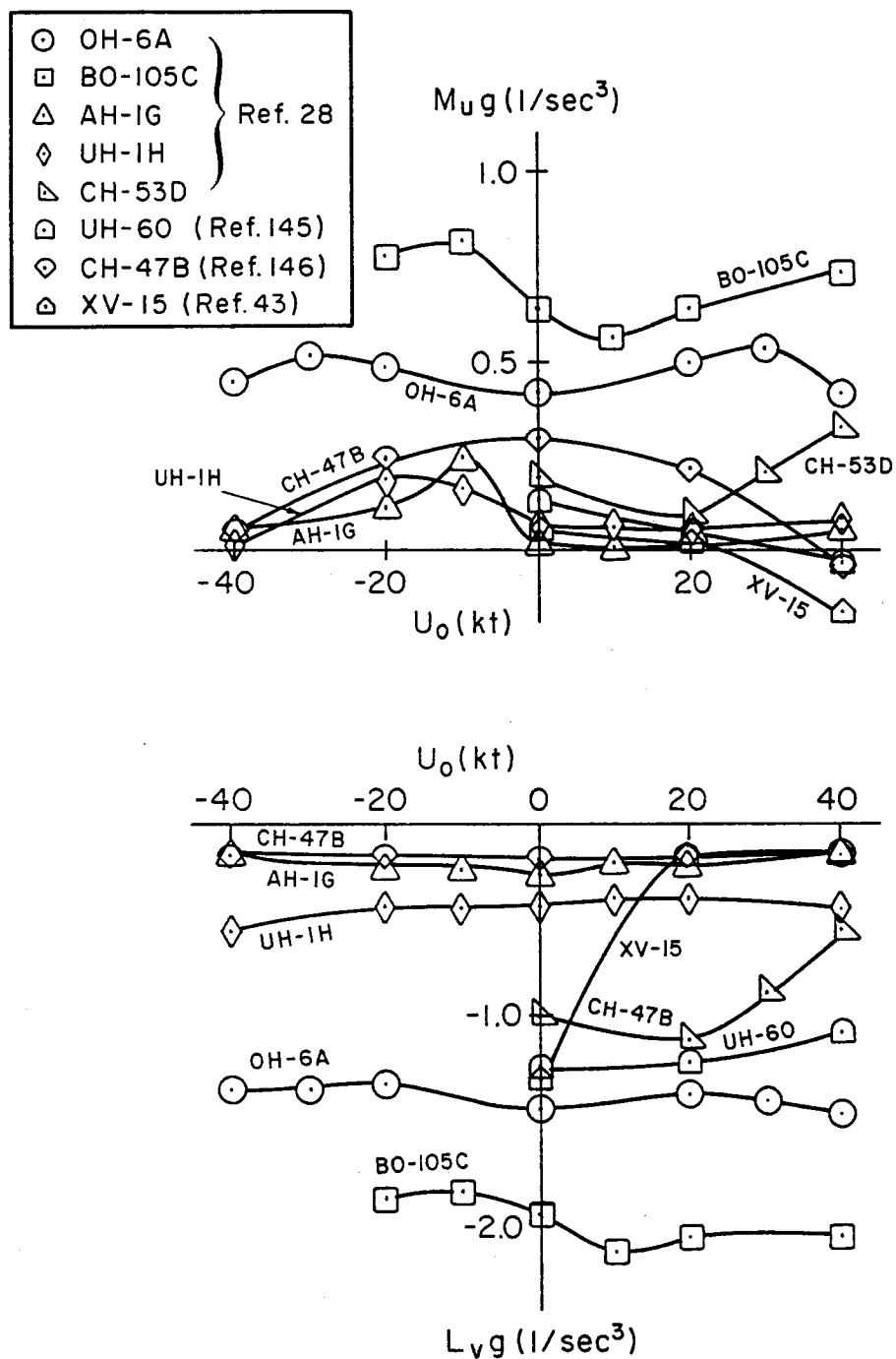


Figure 2(3.3.2.2). Values of the Terms  $M_{Ug}$  and  $L_{Vg}$  for Several Helicopters

Figure 3 summarizes the pilot ratings from Reference 142 for  $X_u$  and  $Y_v$  of  $-0.05$  rad/sec. Only the location of the oscillatory mode is shown for each configuration evaluated; the first-order root varies in value from  $0.5$  to  $6$  rad/sec. As a result, the short-term bandwidths for these configurations vary over a wide range of values between  $0$  and  $4$  rad/sec. It would be desirable to use only those configurations with Level 1 bandwidths to develop a requirement, but clearly this is impossible since bandwidth is influenced by the frequency and damping of the oscillatory root -- i.e., if we limited our analysis to those configurations with bandwidths of  $2$  rad/sec or greater, we would be restricted to only a small and, for the most part, stable subset of the available data.

The symbols on Figure 3 indicate the handling qualities Levels for bandwidth based on the requirements of Paragraph 3.3.2.1. The limits for pitch bandwidth are those for  $UCE=1$  because the visual projection system was considered to have provided very strong cues to the pilots (see Reference 12). The configurations in Figure 3 with  $\omega_n > 0.5$  rad/sec support a requirement for  $\zeta > 0.35$ .

The data of Figures 3a and 3b generally support the Figure 3(3.3) limits. Somewhat more stringent boundaries would be required for simultaneous degradation of the pitch and roll axes, as shown in Figure 3c. These data are presented for design guidance, and are felt to be inadequate for the development of an additional criterion. It seems important to note, however, that all configurations with a mode at  $\omega_n \leq 1$  rad/sec, in both the pitch and roll axes, are rated Level 2.

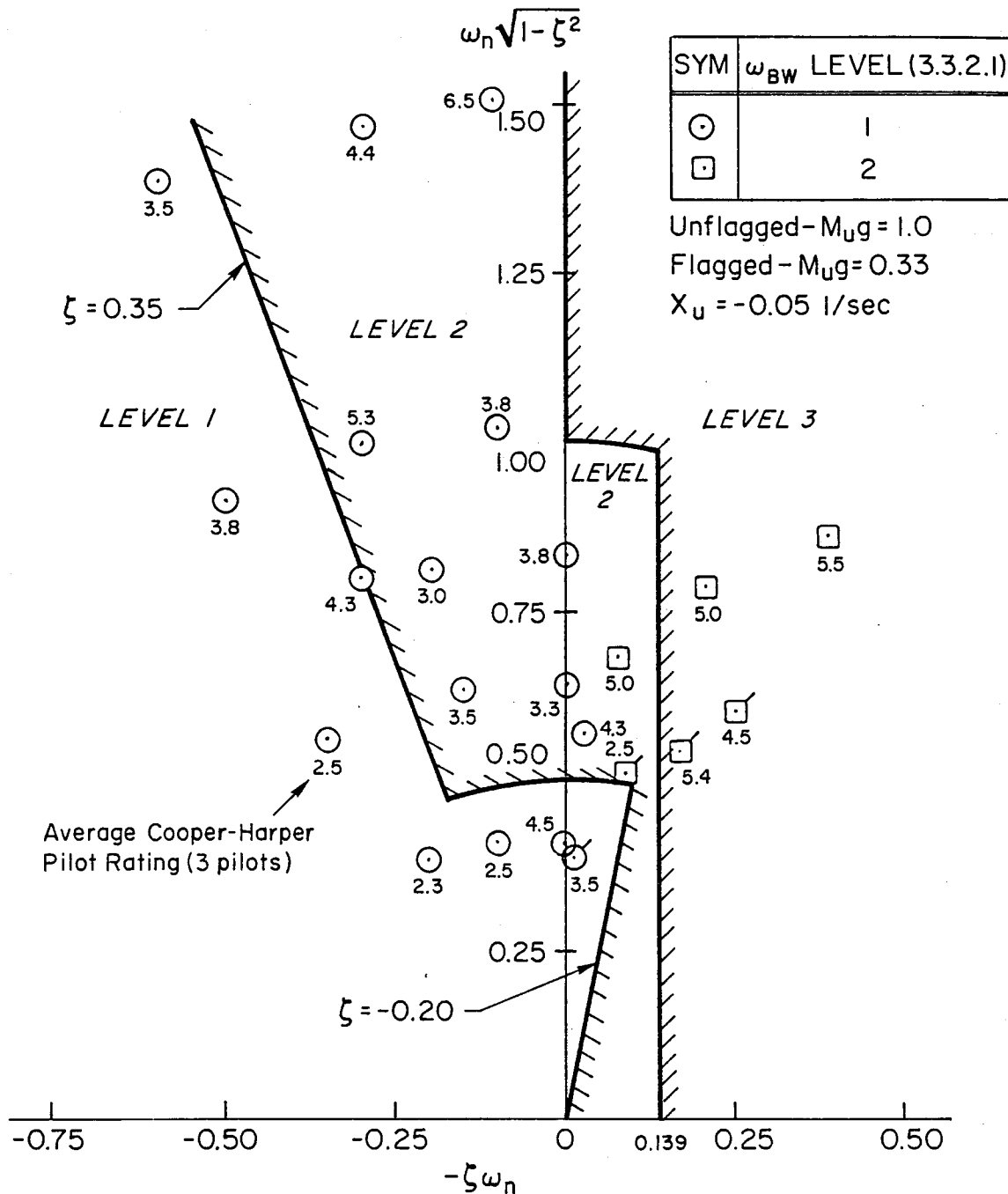
There are no cases in Figure 3 where the oscillatory mode was very low in frequency (i.e., below about  $0.3$  rad/sec). Limits for this region are based primarily on consideration of past requirements, MIL-H-8501A and MIL-F-83300, and the recommendations of Reference 12 (all discussed in the Related Previous Requirements section). The authors of both MIL-F-83300 (see Reference 30) and Reference 12 used Reference 142 as their primary source of data.

The Level 1 limit of Figure 3(3.3) for frequencies between  $0.5$  and  $1.0$  rad/sec is considerably more stringent than those of either MIL-H-8501A or MIL-F-83300: both of these specifications require only essentially neutral damping (i.e.,  $\zeta \geq 0$ ) in this range (as discussed in Related Previous Requirements). The data of Figures 3a and 3b support such a boundary in that the majority of the averaged ratings for  $\zeta > 0$  (for  $0.5 < \omega_n < 1.0$ ) are Level 1. It seems inconsistent, however, to require a bandwidth of at least  $1$  rad/sec in the short-term pitch requirement (Paragraph 3.3.2.1), and to allow zero damping for a mode at slightly less than  $1$  rad/sec. On this basis, a simple fixed-base piloted simulation experiment was conducted to test the region in question (i.e.,  $0 < \zeta < 0.35$  and  $0.5 < \omega_n < 1.0$ ).

#### RESULTS OF FIXED-BASE SIMULATION (REFERENCE 159)

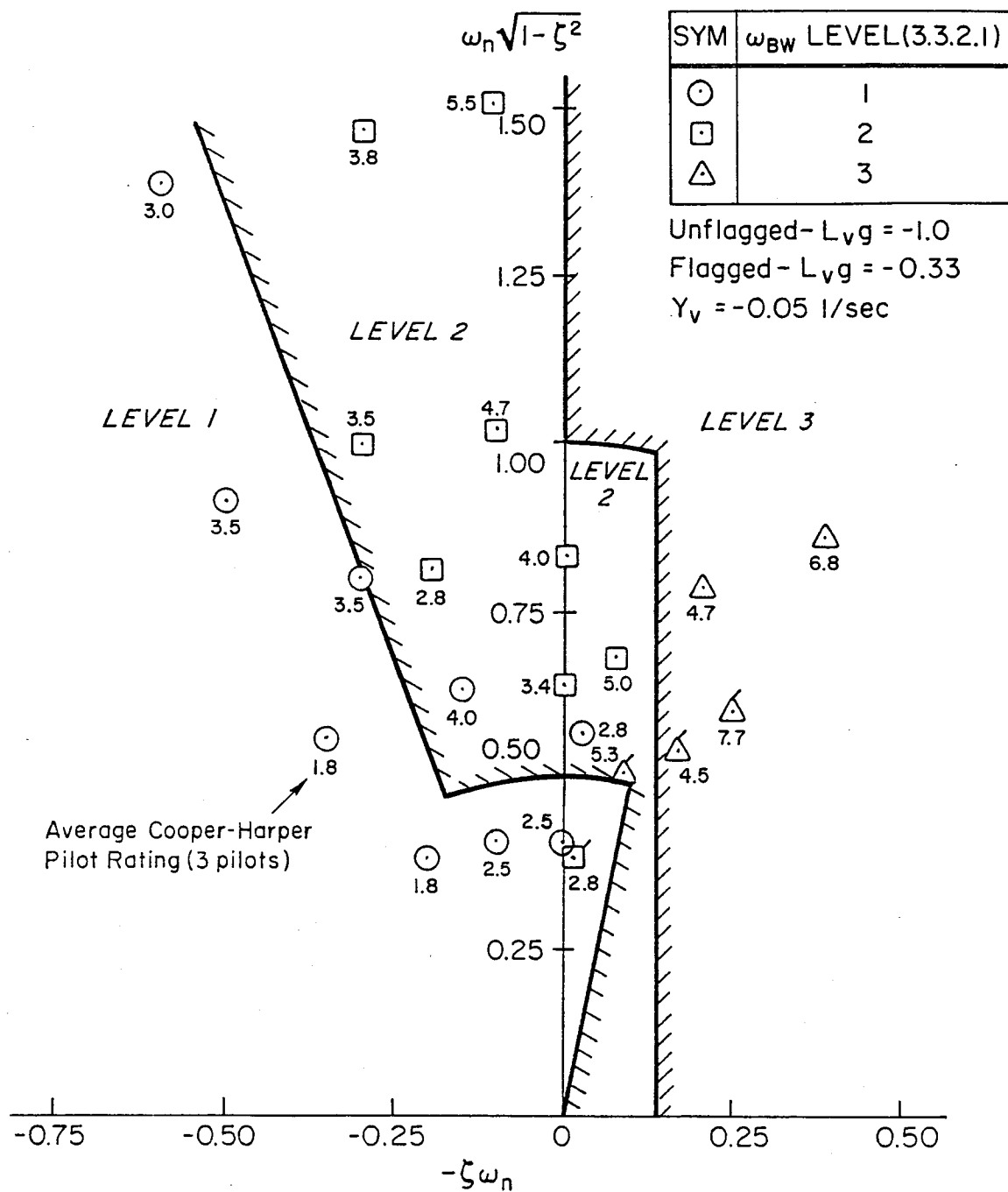
A very brief fixed-base piloted simulation was performed at Systems Technology, Inc., to evaluate the region noted above, and also to check





a) Longitudinal Variations (lateral fixed)

Figure 3(3.3.2.2). Pilot Rating Data from Moving-Base Simulation of Reference 142 (No Stick Force Gradients; with Steady 10-kt Wind and 3.4 ft/sec RMS Turbulence)



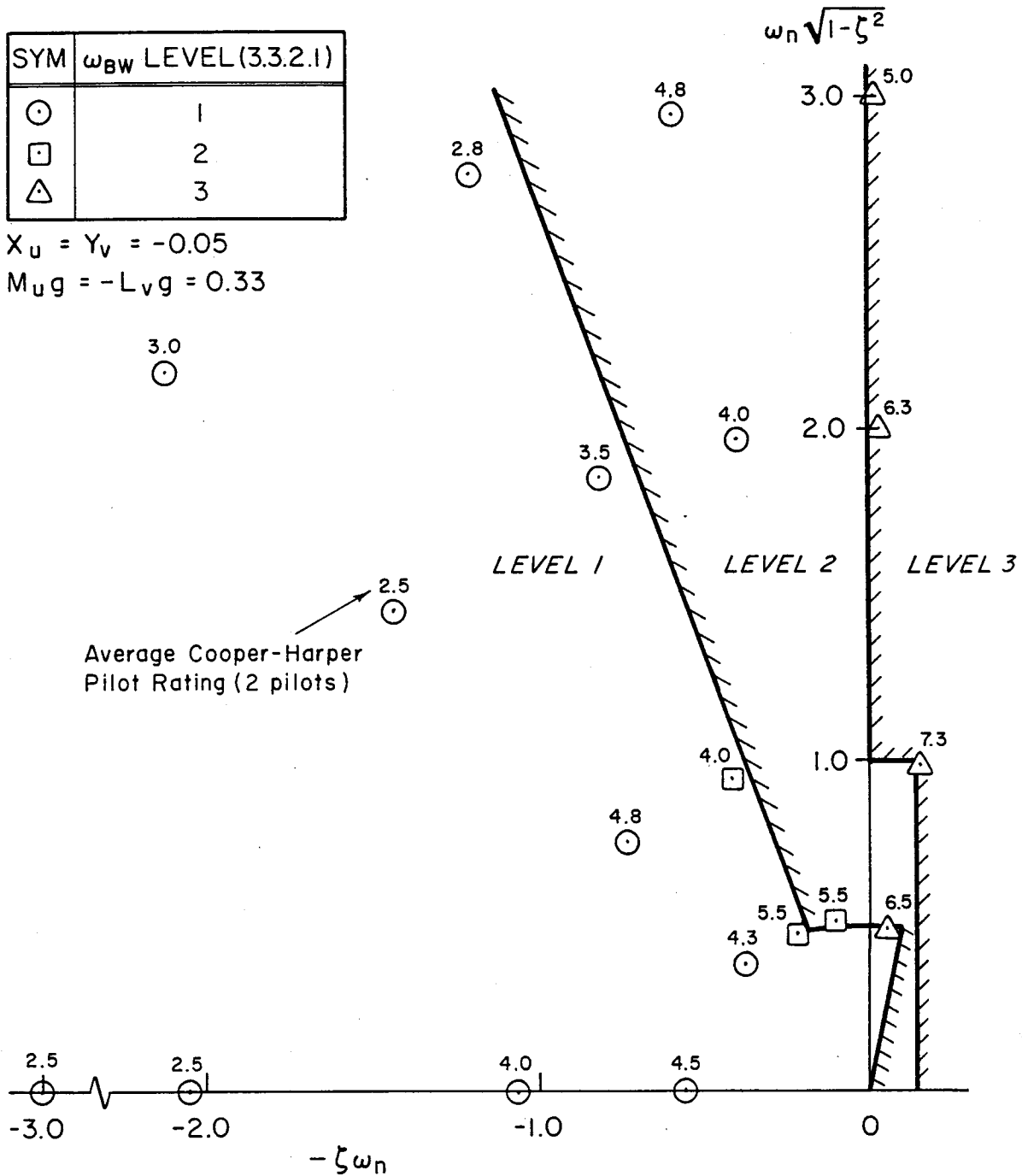
b) Lateral Variations (longitudinal fixed)

Figure 3(3.3.2.2). (Continued)

SYM	$\omega_{BW}$ LEVEL (3.3.2.1)
⊙	1
□	2
△	3

$$X_u = Y_v = -0.05$$

$$M_u g = -L_v g = 0.33$$



*c) Longitudinal and Lateral Varied Together (average PR does not include ratings from pilot F)*

Figure 3(3.3.2.2). (Concluded)

the effect of bandwidth in the presence of a negatively damped mode at  $\omega_n \leq 0.5$  rad/sec. The simulator display consisted of an oscilloscope devised to represent an attitude indicator (pitch bar). A centerstick controller mounted to a chair provided pitch control. The pitch attitude display was generated by an analog computer, thereby eliminating any undesirable time delays. The single-axis task consisted of tracking the pitch bar with a sum of high sinusoids driving the command bar. This setup has been used for a number of simulations (e.g., Reference 153). The controlled element for the simulation had the form of a conventional hovering cubic.

Since the task was a single-axis pitch tracking task, it would be expected to be somewhat easier to perform than the actual flying of a helicopter, where pitch attitude control is only an inner-loop function to achieve position control. Thus, it was expected that if any problems were uncovered, they would probably be worse in the real world.

Two pilots were used for the evaluations. Pilot A is an engineering test pilot with extensive ground based simulation experience and is a commercially rated helicopter pilot with experience in variable stability helicopters. Pilot B is a fixed-wing pilot with limited experience in simulations. Both had recently served as test subjects for other experiments with the same simulation setup. Each pilot was allowed to "fly" each configuration and select his control sensitivity before the formal evaluation. At least two 86-second runs were performed for each configuration before the pilot assigned Cooper-Harper pilot ratings and gave comments. The configurations were presented randomly and the pilots had no prior knowledge of the configurations. Repeat runs were performed in most cases.

The dynamics of the ten configurations are listed in Table 1 along with ratings and excerpts of pilot comments. Configurations 4 and 5 were not evaluated. Figure 4 shows the locations of the oscillatory modes (filled data points, configuration numbers are circled) in comparison with the data from Reference 142 (Figure 3a).

The control case for comparison with the data from Reference 142 was Configuration 8. This configuration was rated Level 1 by both pilots and this result agreed with that from Reference 142. In addition to having Level 1 phugoid characteristics, this configuration also had a pitch bandwidth of 3.9 rad/sec, which is well within the Level 1 bandwidth requirement of Paragraph 3.3.2.1.

Configuration 1 was given Level 2 ratings by both pilots, together with comments such as "tendency to PIO," "sluggish" and "unpredictable." This result is not surprising given the low bandwidth and the relatively high frequency of the second order mode, the combination of which gives rise to low open-loop phase margin. Describing functions taken during the simulation runs showed that this produced a closed-loop amplitude ratio peak of about 10 dB for both pilots (see Figure 5), corresponding to a closed-loop second order mode with a damping ratio of approximately 0.16. This was probably noticeable to the pilots as a PIO tendency. The ratings for a similar configuration in Reference 142 (Case 106, with an average rating of 3.8 in Figure 4), however, are

TABLE 1(3.3.2.2). LIST OF CONFIGURATIONS EVALUATED IN FIXED-BASE SIMULATION

CONFIGURATION	$\theta/\delta_\theta$	$\omega_{H\theta}$ (rad/sec)	DESCRIPTION	PILOT RATINGS		EXCERPTED PILOT COMMENTS
				A	B	
1	$\frac{(0.05)e^{-0.08s}}{(1.0)[0.1; 1.0]}$	1.4	Similar to Case 106 in Ref. 142 (Pilot Ratings of 3, 4, 4.5)	5,5,5,5	5	A: Tendency to PIO. Not crisp -- sluggish, cannot fly aggressively, unpredictable. B: Totally unpredictable. Can get adequate performance...If tighten up, I get PIOs.
2	$\frac{(0.05)e^{-0.08s}}{(1.2)[-0.17; 0.5]}$	0	Similar to Case 156 in Ref. (Pilot Ratings of 5, 5, 6)	4,5,5,5	3,3	A: Predictable but excessively sluggish...objectionable mid-frequency response problems -- acceptably crisp (not great). B: A little unpredictable. Crisp response but seemed to overshoot.
3	$\frac{(0.05)e^{-0.125s}}{(3.6)[0; 1.0]}$	2.0	$\zeta = 0$ , $\omega_h = 1$ , $\omega_{HW} = 2$ Case	4,5	-	A: Somewhat unpredictable and sluggish for tracking
4	$\frac{(0.05)e^{-0.125s}}{(1.0)[0; 1.0]}$	1.0	Same as 3 except $\omega_{HW} = 1$			
5	$\frac{e^{-0.125s}}{[0; 1.0]}$	1.0	Same as 4 except AGAH Response-Type			
6	$\frac{(0.05)e^{-0.125s}}{(4.7)[-0.2; 0.5]}$	2.0	$\zeta = -0.2$ , $\omega_h = 1$ , $\omega_{HW} = 2$ Case	4,3	2	A: A little loose B: No real deficiencies. Quite predictable, crisp, easy to correct errors.
7	$\frac{(0.05)e^{-0.08s}}{(1.85)[-0.125; 0.5]}$	1.0	OH-6A Hovering Cubic	4	3,2	A: Somewhat unpredictable but desired performance attainable. B: Slight predictability problem. Not overly crisp, but adequate.
8	$\frac{(0.05)e^{-0.08s}}{(6)[0.47; 0.43]}$	3.9	Similar to Case 113 in Ref. 142 (Pilot Ratings of 2, 2, 5)	3,3	3	A: Reasonably crisp response and very predictable. B: Little problem in keeping error low.
9	$\frac{(0.05)e^{-0.125s}}{(3.8)[0; 0.75]}$	2.0	$\zeta = 0$ , $\omega_h = 0.75$ , $\omega_{HW} = 2$ Case	5,4	4,3	A: Tendency to PIO -- unpredictable. Requires full attention. B: Between 3 and 4...annoying. Initially sluggish Not predictable.
10	$\frac{(0.05)e^{-0.125s}}{(2.0)[0.35; 0.75]}$	2.0	Same as 9 except $\zeta$ increased to 0.35	4,3	4,4	A: Moderately unpredictable. Tendency to PIO if I try to be aggressive. B: Lack of precision. Prone to overcontrol. Easy to get adequate performance.

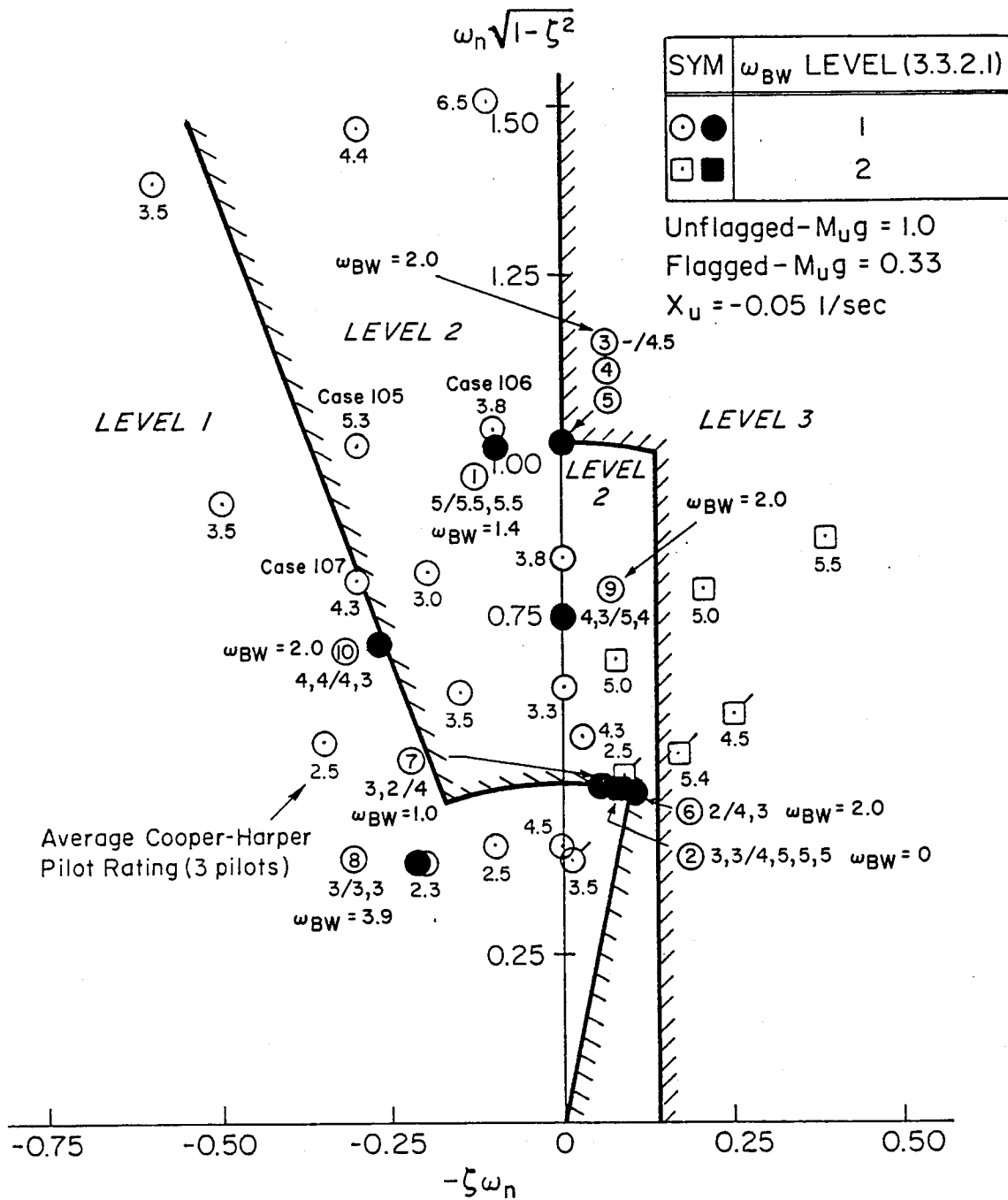


Figure 4(3.3.2.2). Configurations Evaluated on Fixed-Base Simulation  
 (Solid Symbols) Compared to Pitch Evaluations of Reference 142  
 (Figure 3a). Number in Circle is Configuration Number  
 (Table 1); Ratings are for Pilot B/Pilot A.  
 Repeat Ratings are Separated by Commas

somewhat better: 3, 4, and 4.5. An additional case evaluated in Reference 142 (Case 105, with the same  $\omega_n$  as Case 106 but with a higher  $\zeta$ ) received solidly Level 2 ratings in that simulation (ratings of 5, 5, 6 for an average of 5.3). Hence this case was rated Level 2 despite its relatively higher damped phugoid mode and Level 1 bandwidth. The discrepancy in the results of Reference 142 and the strong opinions of the pilots about Configuration 1 gave some merit to the investigation of a more restrictive Level 1 boundary in the positively damped region of Figure 4.

Configurations 9 and 10 were developed specifically to test more restricted limits with  $\omega_n = 0.75$  rad/sec. Both had a Level 1 pitch bandwidth of 2.0 rad/sec. These were given marginally Level 2 ratings by the pilots. The ratings for Configuration 10 are consistent between pilots as well as with Reference 142 data as evidenced by the point in Figure 4 with a slightly higher frequency (Case 107 in Reference 142) and an average pilot rating of 4.3. Comments for Configuration 9 were similar to those given for Configuration 1, but the ratings show that desired performance could be obtained in this case.

The ratings and comments for Configurations 9 and 10 indicated that a lowering of the allowable frequency of the phugoid mode might be appropriate. This categorizes some current unaugmented helicopters as having Level 2 handling qualities at low speeds. While acknowledging the differences in the simulation and task setup between Reference 142 and this experiment, it was the opinion of the pilots that a helicopter with pitch characteristics similar to those exhibited by Configurations 1, 9, and 10 would definitely be outside the Level 1 boundary. The closed-loop describing function data in Figure 5 confirms this observation for Configuration 1.

Configurations 3, 4, and 5 were not expected to yield results considerably different from those for Configuration 1 and hence, these configurations were not investigated in any detail. It is interesting to note, however, that Pilot A ascribed a better rating to Configuration 3 than he did for Configuration 1. This is probably due to the higher bandwidth of Configuration 3.

The remaining three configurations (2, 6, and 7) differed in bandwidth but had very similar (unstable) phugoid modes on the Level 1 boundary defined by  $\omega_n = 0.5$  rad/sec. Figure 6 shows the trend of pilot rating with bandwidth for these configurations, and indicates that they are marginally Level 1, if  $\omega_{BW\theta} \geq 1$  rad/sec. These data, on average, support the bandwidth limit of 1 rad/sec as well as the  $\omega_n = 0.5$  rad/sec boundary in Figure 3(3.3). The low bandwidth of Configuration 7 was noted by both pilots in their comments on the lack of crispness and unpredictability of the response. The ratings for these three configurations show bandwidth to be the deciding factor (Figure 6). The phugoid mode, though unstable, seemed to have been of sufficiently low frequency to have been stabilized, with little compensation by the pilots. There is a 1-1/2 point rating bias between Pilots A and B for Configurations 2, 6, and 7, although the trends are identical.

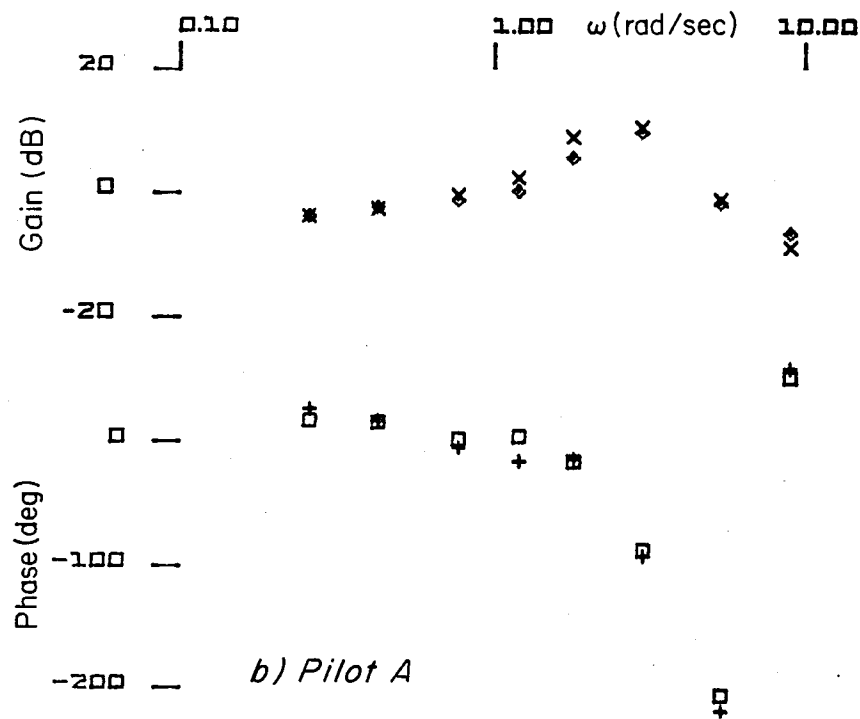
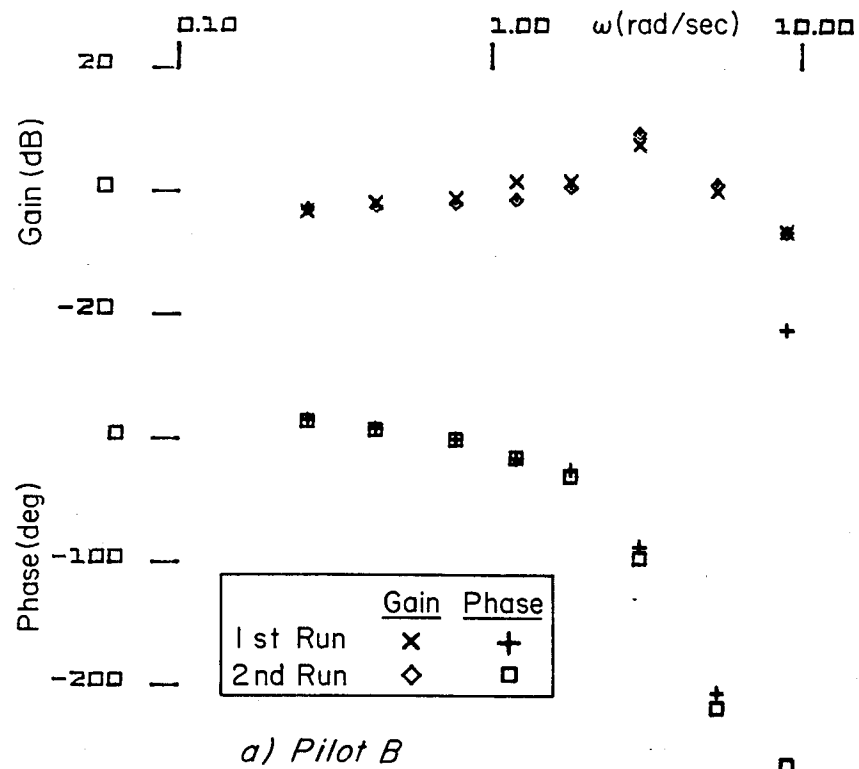


Figure 5(3.3.2.2). Measured Closed-Loop Describing Functions for Configuration 1



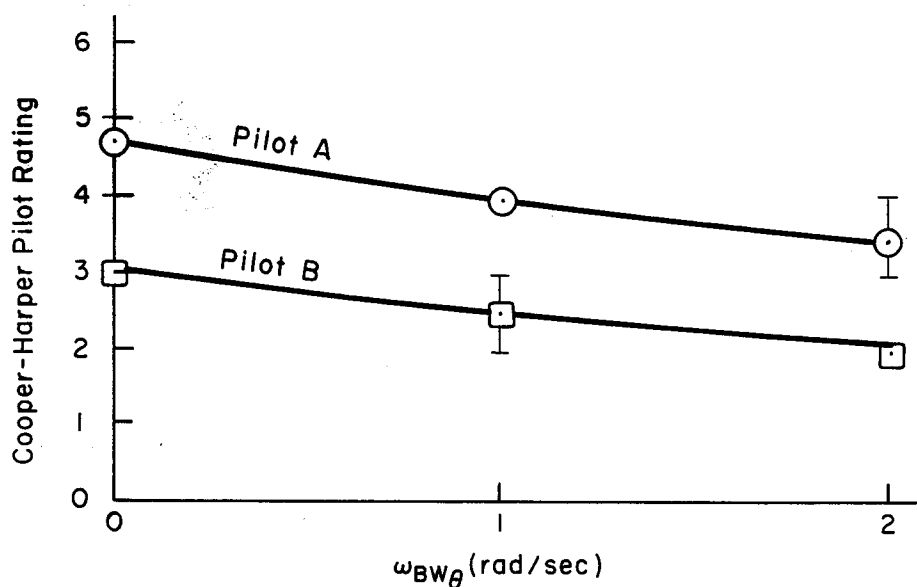
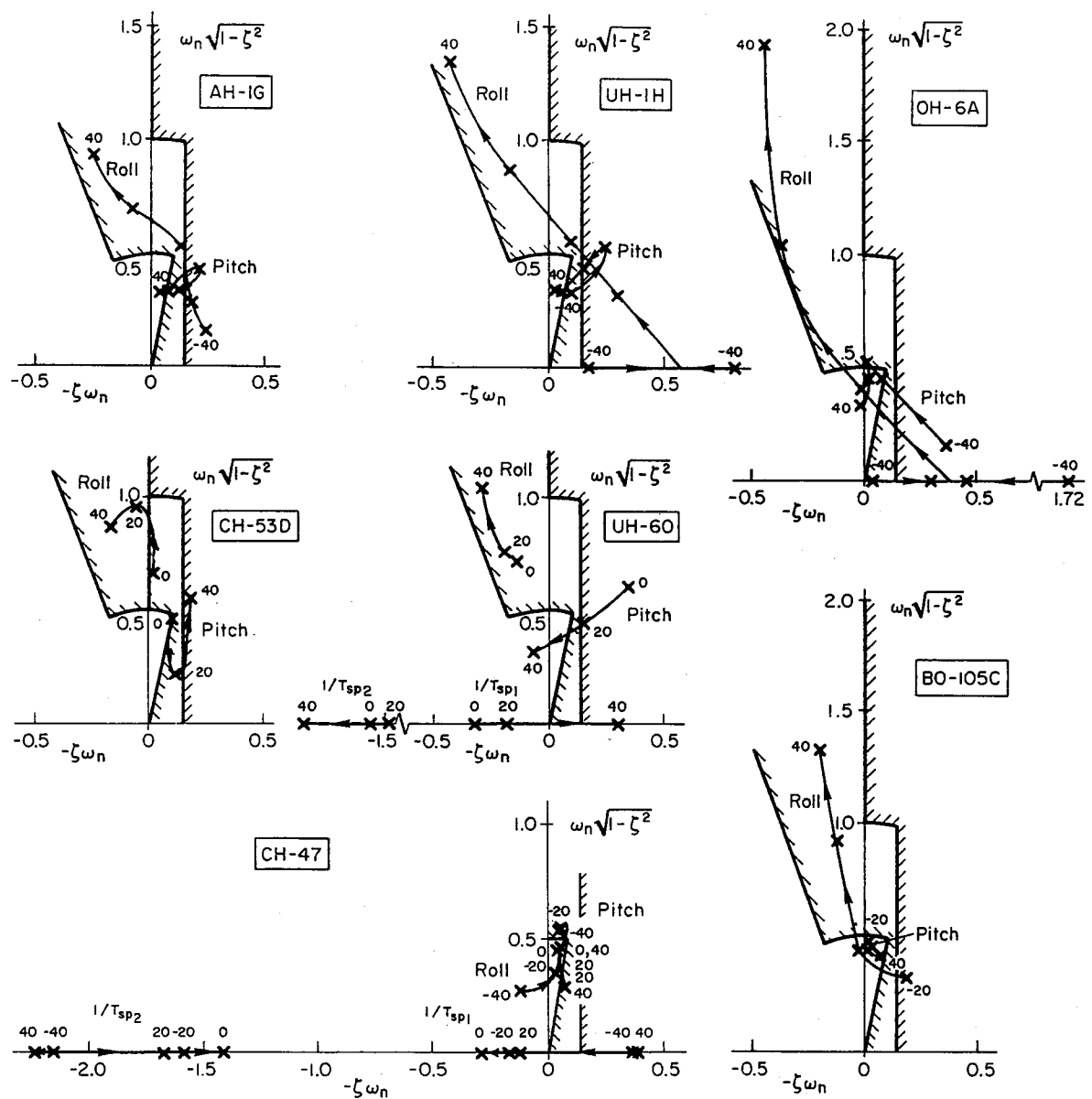


Figure 6(3.3.2.2). Effect of Bandwidth on Configurations with Marginal Oscillatory-Mode Characteristics at  $\zeta \doteq -0.2$ ,  $\omega_n = 0.5$  rad/sec (Configurations 2, 6, and 7)

#### COMPARISON WITH OPERATIONAL HELICOPTERS

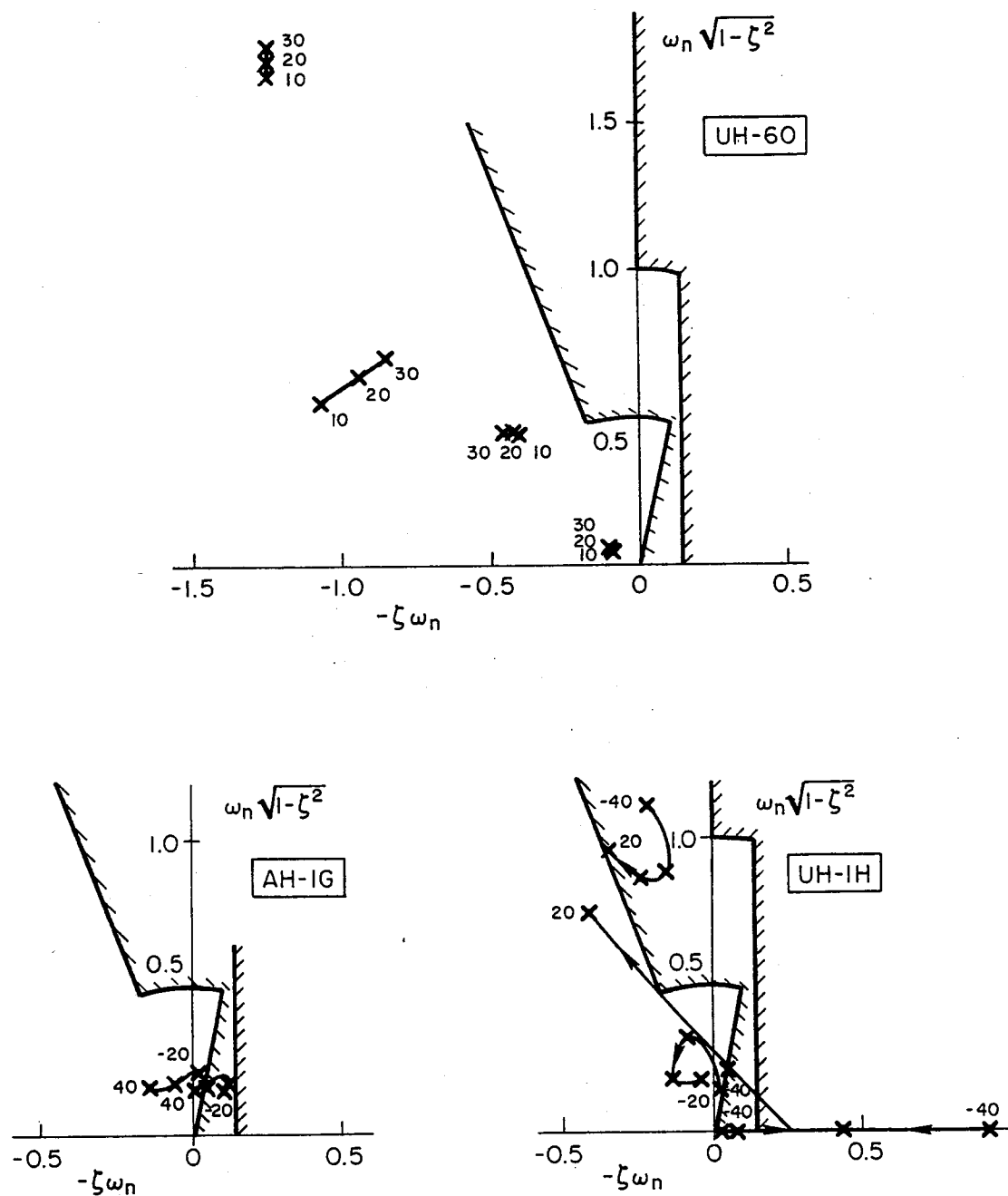
There is little quantitative data available for currently operational helicopters. For example, review of flight test reports generated by the Army (especially References 67 through 81) did not provide quantitative information for comparison with the requirements of Figure 3(3.3). It was therefore necessary to turn to analytical models to determine the oscillatory-mode dynamics of several helicopters, as shown in Figure 7. These modes are shown in the form of root loci, where the variable in the loci is airspeed. End points are labeled on each locus; each X between end-points represents a 20-kt increment. Figure 7 indicates that nearly all of the helicopters shown would be Level 2 or worse in rearward flight, but would improve in forward flight.

While no quantitative measures of pitch and roll dynamics could be found in the flight test literature for operational helicopters, some interesting comments can be found; as an example, Table 2 has excerpts from References 69, 151, and 152, describing the low-speed handling qualities of the unaugmented CH-47C, YUH-60A, and OH-58C, respectively. The CH-47C was considered to be Level 2 (pilot rating of 4) in VFR conditions with the SAS off. Figure 7 indicates that this is not too surprising, considering that both the pitch and roll modes are near the Level 1 limit at low speeds. In addition, the bandwidth of the CH-47 is Level 2 SAS-off. Similarly, the YUH-60A (essentially the UH-60) was given a pilot rating of 6 for precision hovering, and Figure 7 shows that the pitch mode lies in the Level 3 region. The dynamics of the OH-58C are not available, but the comments in Table 2 describe a divergent pitch oscillation that "became uncomfortable after the first cycle" but was still acceptable (pilot rating of 3). This supports the observation that some level of divergence is not objectionable.



a) SCAS Off

Figure 7(3.3.2.2). Comparison of Oscillatory Characteristic Modes of Several Helicopters with Requirements of Figure 3 (3.3)



*b) SCAS On (all pitch and roll modes; individual modes are coupled and are not identified)*

Figure 7(3.3.2.2). Concluded

TABLE 2(3.3.2.2). EXCERPTS FROM FLIGHT TEST REPORTS

<u>AIRCRAFT</u>	<u>COMMENTS</u>
CH-47C (Ref. 69)	Under VFR conditions (SAS off), the pilot was able to retain control of the helicopter with moderate effort (HQRS 4). The SAS-OFF flight characteristics are such that extended flight and safe landings could be accomplished under VFR conditions. Flight under simulated IFR conditions with the SAS OFF required considerable pilot effort to satisfactorily control the helicopter and left no time to devote to other tasks (HQRS 8).
YUH-60A (Ref. 151)	The short-term response characteristics of the helicopter with AFCS ON were essentially deadbeat.... The deadbeat short-term characteristics were also evident for all axes during flight through moderate turbulence. The pilot was able to correct for attitude disturbances in level flight and hover with minimal effort (HQRS 3).... (With AFCS off) pilot-induced oscillations occurred about all axes during all flight phases due to the loss of rate damping and attitude hold. Aircraft control during precision hovering and simulated IMC tasks with AFCS OFF required extensive pilot compensation (HQRS 6).
OH-58C LCH (Light Combat Helicopter) (Ref. 152)	Lateral and directional control pulses generated an essentially deadbeat aircraft response. Forward and aft longitudinal control inputs were accompanied by a divergent pitch oscillation which became uncomfortable after the first cycle. However, the motion was slow to develop [and] easily controlled (HQRS 3).... Low-speed forward flight was easily accomplished (HQRS 2). Rearward flight was accomplished from zero to 30 KTAS with considerable pilot compensation required to limit pitch and yaw excursions to 15 degrees (HQRS 5).... Right sideward flight was accomplished from zero to 35 KTAS with minimal pilot compensation to control pitch, roll, and yaw oscillations (HQRS 3).... Left sideward flight at 20 and 25 KTAS could not be satisfactorily performed due to excessive pitch, roll, and yaw excursions (HQRS 7).... Left sideward flight at 25 KTAS required only minimal pilot compensation (HQRS 3).

#### SUPPORT FOR PARAGRAPH 3.3.2.2.2

Since piloted simulations of low-speed flight have historically not included divided-attention tasks, it is impossible to determine how stringent a requirement on such operations should be. The specification simply states that all modes must have a damping ratio of 0.35 or greater. The comments excerpted in Table 2 indicate that a divergence that is otherwise acceptable or not overly objectionable in fully-attended flight becomes totally unacceptable in a divided-attention environment. For the CH-47C, SAS-off, a helicopter that was manageable (pilot rating of 4) in VFR became Level 3 in simulated IFR conditions (rating of 8) with "no time to devote to other tasks," i.e., no time for divided attention. Similarly, the unaugmented YUH-60A developed "pilot-induced oscillations... about all axes" and control during simulated IMC "required extensive pilot compensation (HQRS 6)."

#### d. Guidance for Application

None.

#### e. Related Previous Requirements

Both the rotary-wing specification, MIL-H-8501A (Reference 31), and the fixed-wing VSTOL specification, MIL-F-83300 (Reference 29), contain limits on the allowable oscillations in pitch and roll. In MIL-H-8501A, there are requirements for both VFR and IFR operations, the latter being more stringent. The MIL-F-83300 limits are also given for VFR and IFR operations, and are divided into handling qualities Levels -- a distinction lacking in the helicopter specification. The helicopter limits may be interpreted as "pass/fail" rather than as Level 1 limits, and thus they correspond to some average Cooper-Harper pilot rating greater than 3-1/2. This is important to keep in mind because the MIL-H-8501A limits are more lenient, in general, than MIL-F-83300, and considerably more lenient than the requirements of Figure 3(3.3).

Figure 8 compares the Level 1 limits of Figure 3(3.3) with the Level 1 limits from MIL-F-83300 and the IFR requirements from MIL-H-8501A. At low frequencies (below 0.5 rad/sec), the current limit is not as strict as MIL-F-83300, but it does not allow the level of very-low-frequency instabilities permitted by MIL-H-8501A. At higher frequencies (between 0.5 and about 1 rad/sec), the helicopter and fixed-wing VSTOL specifications are close, while the current requirement is considerably more stringent. Above 1 rad/sec, the current limit is close to MIL-F-83300.

The similarities between the current limits and MIL-F-83300 are not surprising, considering that they are based on essentially the same supporting data. The increase in damping required at moderate frequencies is a result of the limited fixed-base simulation described in the Supporting Data section. The authors of Reference 12, reviewing the same data, recommended requirements very similar to those in MIL-F-83300.

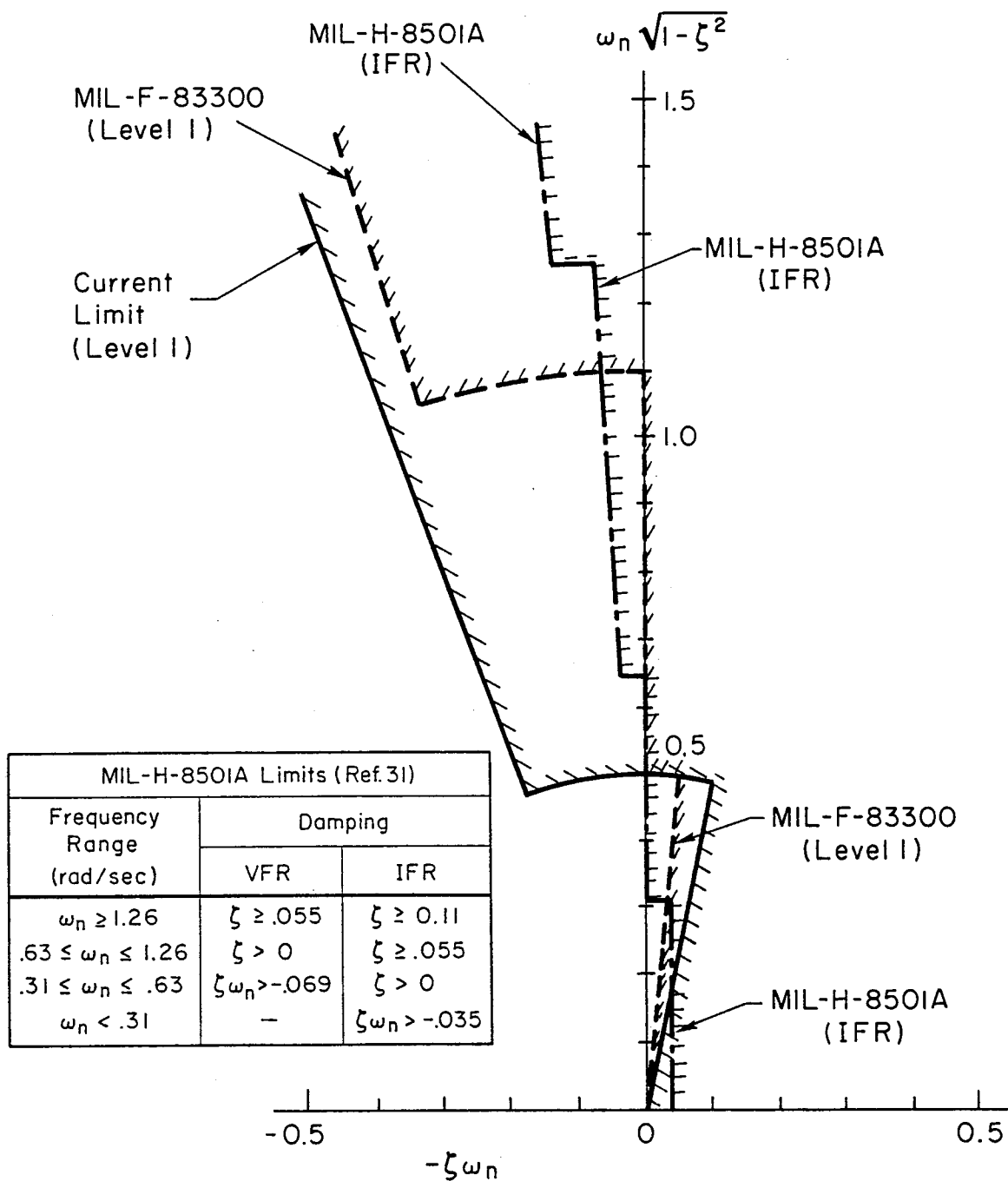


Figure 8(3.3.2.2). Comparison of Level 1 Limits of Figure 3(3.3) with Requirements from MIL-H-8501A (Reference 31) and MIL-F-83300 (Reference 29)

a. Statement of Requirement

3.3.2.3 Short-Term Pitch and Roll Responses to Disturbance Inputs. Pitch and roll responses to inputs directly into the control surface actuator shall meet the bandwidth limits based on cockpit controller inputs as specified in Paragraph 3.3.2.1. If the bandwidth and phase delay parameters based on inputs to the control surface actuator can be shown to meet the cockpit control input bandwidth requirements by analysis, no testing is required. This requirement shall be met for Level 1, and relaxed according to Paragraph 3.2.12 for Levels 2 and 3.

b. Rationale for Requirement

It is possible, through command shaping, to achieve bandwidths in response to disturbances (i.e., turbulence) that are different from those for control inputs. This paragraph requires that, for Level 1, the disturbance-input bandwidths meet the control-input limits. It allows a relaxation for degraded operations (Levels 2 and 3), where command shaping may be employed to improve the control-response bandwidth with no corresponding improvement in the disturbance-rejection bandwidth.

c. Supporting Data

See the discussion for Paragraph 3.2.12.

d. Guidance for Application

See the discussion for Paragraph 3.2.12.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.3 Moderate-Amplitude Pitch (Roll) Attitude Changes (Attitude Quickness). The ratio of peak pitch (roll) rate to change in pitch (roll) attitude,  $q_{pk}/\Delta\theta_{pk}$  ( $p_{pk}/\Delta\phi_{pk}$ ), shall exceed the limits specified in Figure 4(3.3). The required attitude changes shall be made as rapidly as possible from one steady attitude to another without significant reversals in the sign of the cockpit control input relative to the trim position. The initial attitudes, and attitude changes, required for compliance with this requirement, shall be representative of those encountered while performing the required Mission-Task-Elements (Paragraph 3.1.1). It is not necessary to meet this requirement for Response-Types which are designated as applicable only to UCE = 2 or 3.

b. Rationale for Requirement

The parameters  $q_{pk}/\Delta\theta$  and  $p_{pk}/\Delta\phi$  are analytically shown below to be directly related to bandwidth, so this paragraph effectively allows decreasing bandwidth with increasingly large maneuvers. As the amplitude increases from small values, this interpretation becomes less appropriate, and the boundaries are better interpreted as a measure of agility. The boundaries of Figure 4(3.3) are specifically applied above attitude changes normally associated with fine tracking (5 deg in pitch and 10 deg in roll); below these attitude changes, the bandwidth requirements, Paragraph 3.3.2.1, apply. They are extended to attitudes where the angular rate (or attitude for Attitude Response-Types) requirements (Paragraph 3.3.4) are more relevant, 30 deg in pitch and 60 deg in roll.

The details of this requirement have undergone constant review and improvement as experience with its application has been gained. Numerous comments and critiques from both private industry and government have revealed various aspects of the realities of applying, and complying with, all of the attitude quickness requirements in the specification. These important aspects of the attitude quickness criteria are discussed at length in section d, Guidance for Application. The specific rationale for each of the requirements of Figure 4(3.3) is as follows:

- a. Target Acquisition and Tracking (pitch): There is no information for setting pitch limits for this requirement. The Level 1 limit in Figure 4a(3.3) is based on the roll Level 2 limit in Figure 4b(3.3), discussed below. The Level 2 limit is equivalent to a requirement for an approximately constant bandwidth around 0.5 rad/sec (the Level 2 limit in Paragraph 3.3.2.1). This is considered to be a very low bandwidth even for fine tracking, so there is no reason to allow further reductions as maneuver amplitude increases.



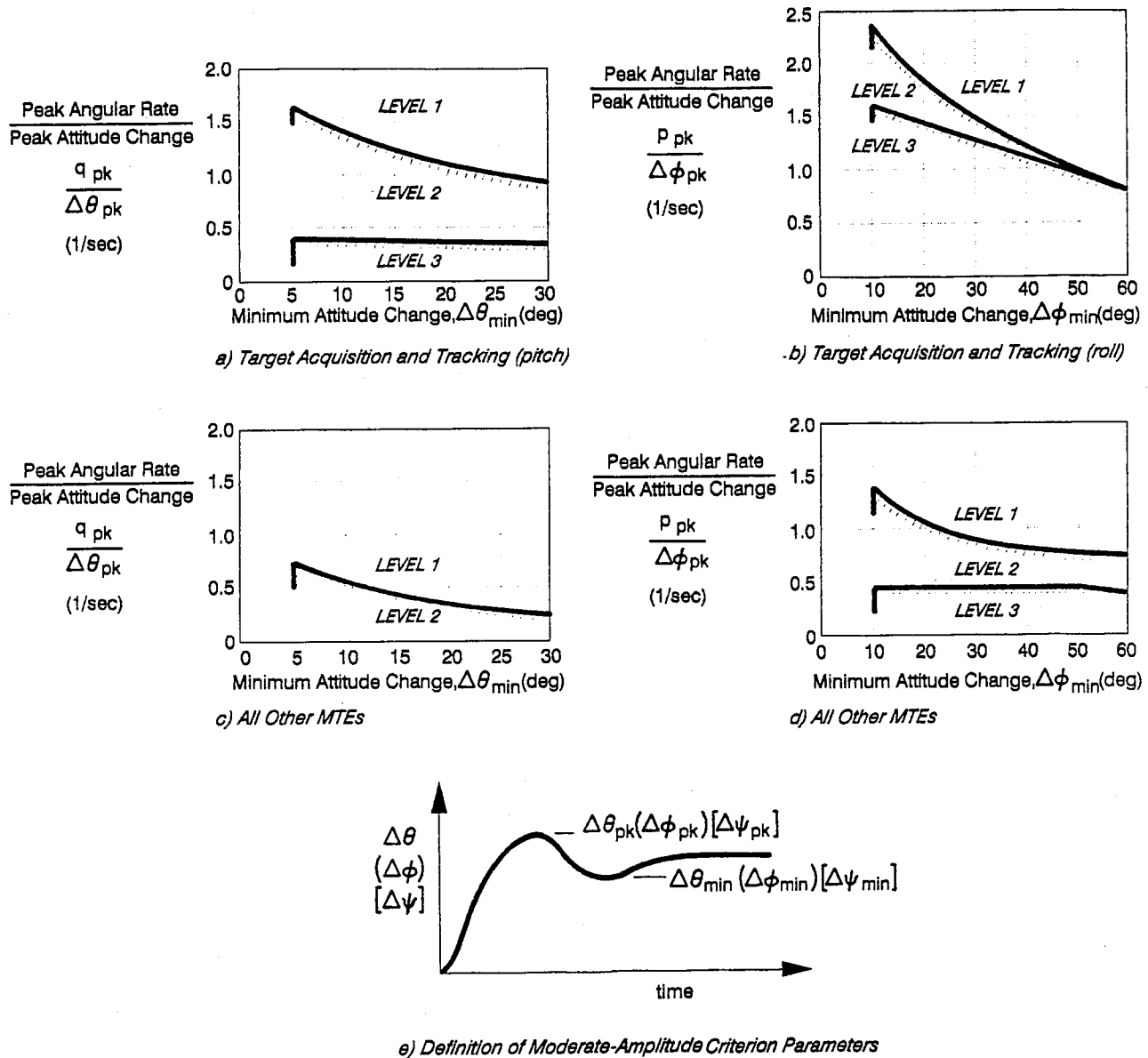


Figure 4(3.3). Requirements for Moderate-Amplitude Pitch (Roll) Attitude Changes -- Hover and Low Speed

- b. Target Acquisition and Tracking (roll): The lower ends of the Level 1 and 2 limits in Figure 4b(3.3) are based on the forward-flight requirement, Paragraph 3.4.5.2, which was in turn developed from HUD tracking simulation data in Reference 93. These data are presented in the Supporting Data discussion for Paragraph 3.4.5.2. The upper ends of the boundaries in Figure 4b(3.3) were set so that they meet the angular-rate requirement for Aggressive Maneuvering (Table 1(3.3)) at the larger bank angles.
- c. All Other MTEs (pitch): The lower end of the Level 1 limit in Figure 4c(3.3) has been set to allow a natural reduction in bandwidth from the small-amplitude limit of 1 rad/sec (at  $\Delta\theta \doteq 0$ ) to a value consistent with the angular rate requirement for moderate maneuvering (Table 1(3.3)) at the upper end.
- d. All Other MTEs (roll): The boundaries in Figure 4d(3.3) are based on simulation data for sidestep maneuvers, as discussed in Supporting Data.

Background for this requirement can be found in the roll control study of Reference 93, which was the initial source for all of the moderate-amplitude criteria in the specification. In the flight and simulation programs of Reference 93, a number of discrete lateral maneuvering tasks were devised for both low speed and forward flight, from which "maneuver performance" diagrams were constructed. The technique for constructing such plots is sketched in Figure 1. For a maneuver that requires discrete control inputs (i.e., inputs that closely resemble square waves, Figure 2) the cross-plot of peak roll rate,  $p_{pk}$ , versus bank angle change,  $\Delta\phi$ , for the entire maneuver represents a "task signature" (Figure 3) related to the pilot's demands on the vehicle. For example, a high ratio of  $p_{pk}/\Delta\phi$  means that the pilot requires high agility in the roll axis, while a low ratio (for the same bank angle change) indicates a correspondingly low roll performance demand.

For ideal Rate and Attitude Response-Types (no time delays, high-order modes, control position or rate limiting, etc. -- see Guidance for Application), the relationships between bandwidth and  $p_{pk}/\Delta\phi$  can be derived for ideal inputs as shown in Table 1. The relationships between bandwidth frequency and  $p_{pk}/\Delta\phi$  are plotted in Figure 4 for the Rate Response-Types. For a simple first-order approximation (i.e., assuming pure roll damping,  $1/T = -L_p$ ), the ratio  $\omega_{BW}/p_{pk}/\Delta\phi$  is identically equal to one. If flapping and lag terms are included, the ratio is much greater than one for low effective damping ratios and approaches one at higher damping ratios. When rate augmentation is added to produce a response with a first-order numerator term, the relationship is somewhat more complicated, as Table 1 shows. For a representative value of the zero  $1/T_p = 0.75$  rad/sec, and for bandwidth frequencies between 2 and 5 rad/sec, the ratio  $\omega_{BW}/(p_{pk}/\Delta\phi)$  is approximately equal to one, within

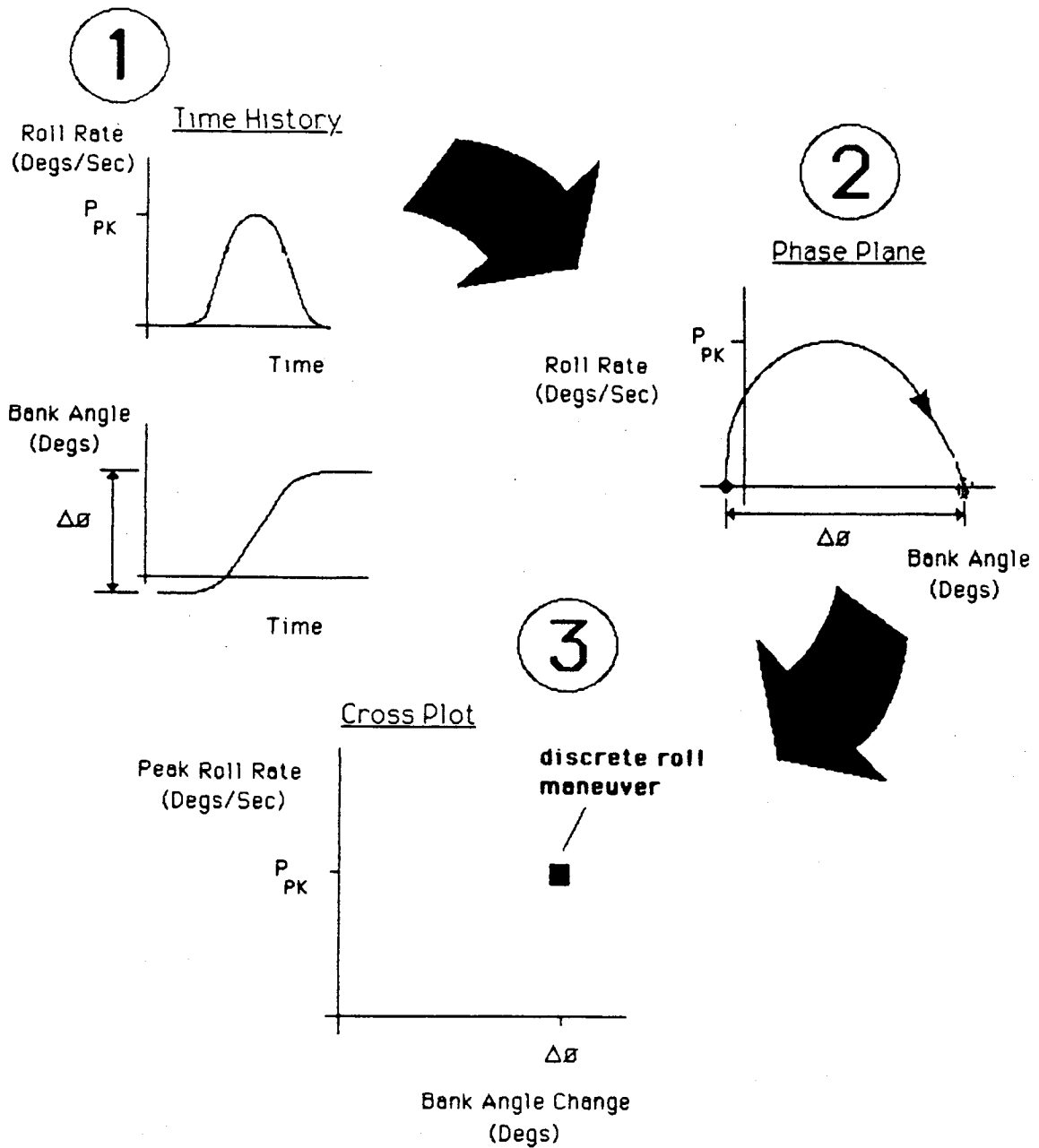


Figure 1(3.3.3). Analysis Technique for Discrete Roll Maneuver Data (from Reference 93)

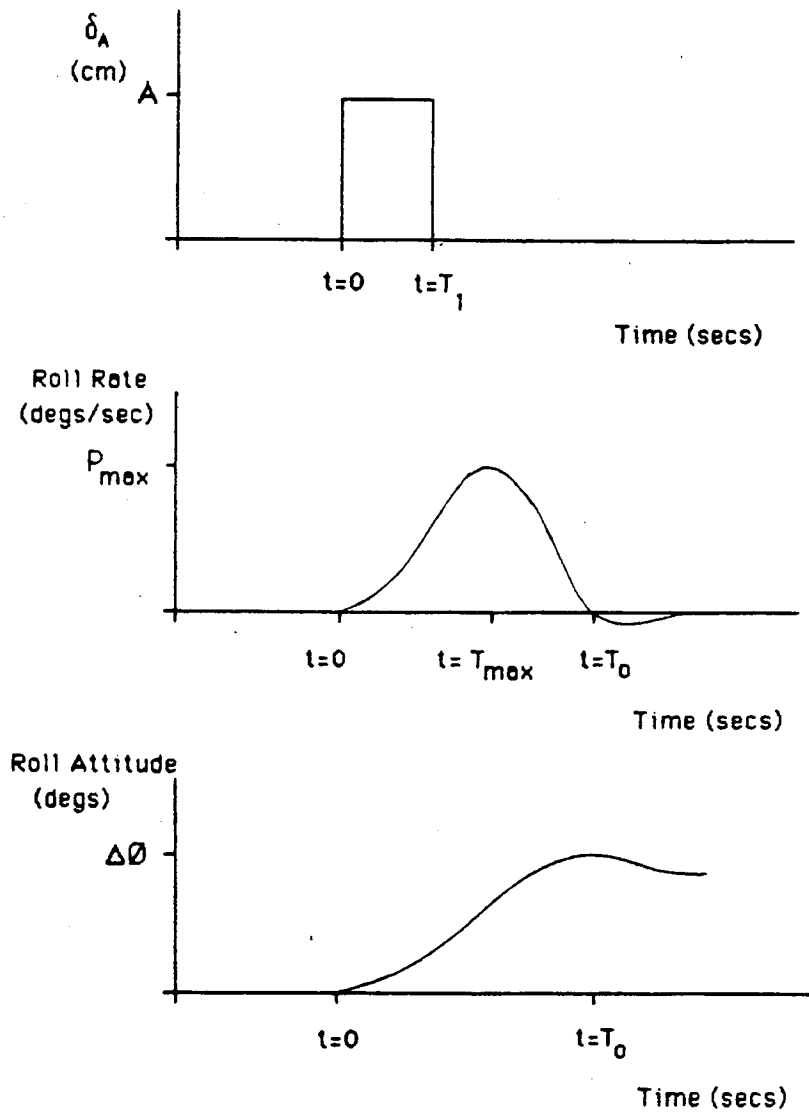


Figure 2(3.3.3). Characteristics of Square Wave Input Response for a Rate Response-Type (from Reference 93)

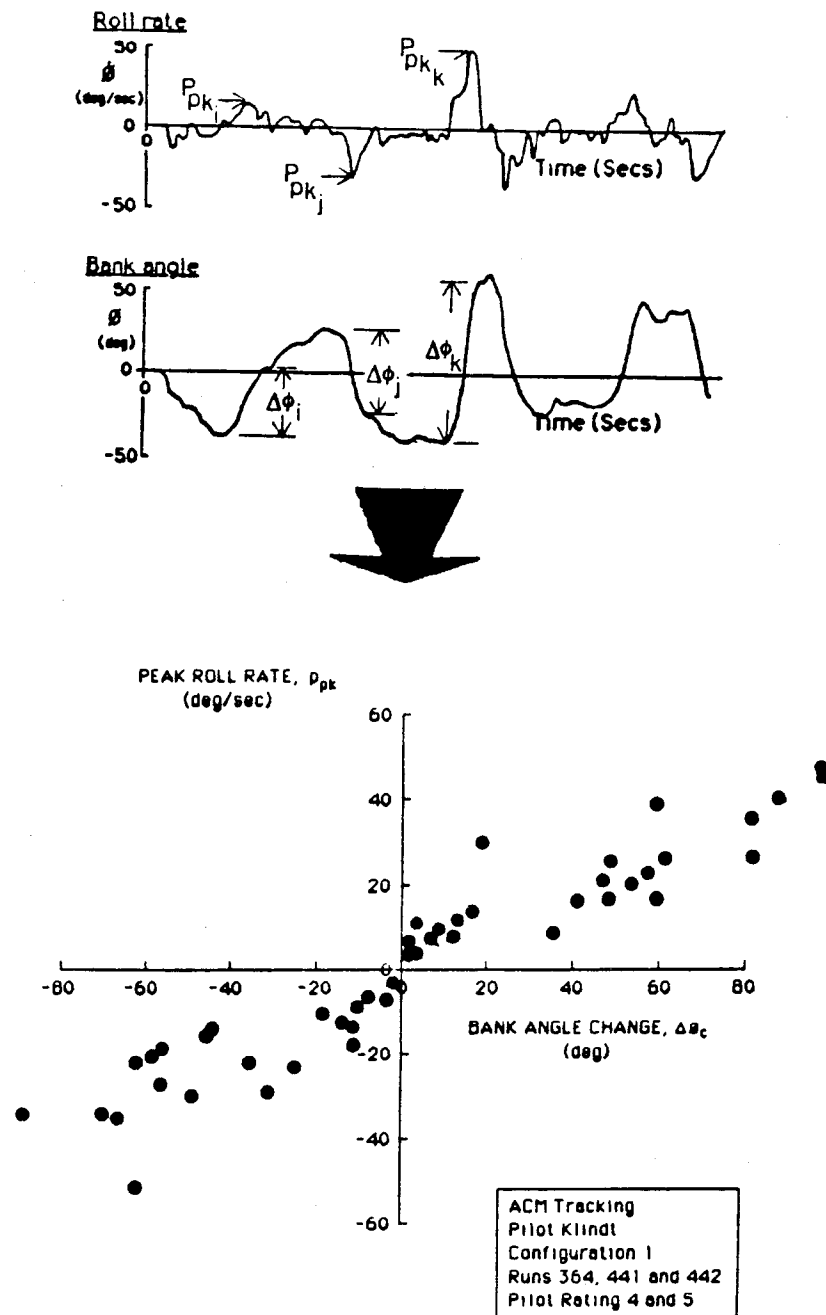


Figure 3(3.3.3). Definition of the "Task Signature" from Discrete Maneuver Time Histories (from Reference 93)

TABLE 1(3.3.3). LIMITING MANEUVER PERFORMANCE CHARACTERISTICS  
FOR RATE AND ATTITUDE RESPONSE-TYPES

Description	Input	Model	$p_{pk}/\Delta\phi_{pk}$	$\omega_n/(p_{pk}/\Delta\phi_{pk})$	$\omega_{BW}/(p_{pk}/\Delta\phi_{pk})$
1) Rate Response-Types	Impulse				
a) Roll Damping Approx.		$\frac{p_c}{p_c} = \frac{1/T}{s+1/T}$	$1/T$	N/A	1.0
b) Flap/Lag Approx.		$\frac{p_c}{p_c} = \frac{\omega_n^2}{s^2+2\zeta\omega_n s+\omega_n^2}$	$\frac{\omega_n \exp(-\alpha \sin^{-1} \sqrt{1-\zeta^2})}{1+\exp(-\pi\alpha)}$	$\frac{1+\exp(-\pi\alpha)}{\exp(-\alpha \sin^{-1} \sqrt{1-\zeta^2})}$	$\frac{(-\zeta+\sqrt{\zeta^2+1}) [1+\exp(-\pi\alpha)]}{\exp(-\alpha \sin^{-1} \sqrt{1-\zeta^2})}$
c) Rate-Augmented		$\frac{p_c}{p_c} = \frac{\omega_n^2 T_p (s+1/T_p)}{s^2+2\zeta\omega_n s+\omega_n^2}$	$\frac{(\omega_n/\sqrt{1-\zeta^2}) A \sin\psi}{1+A \exp(-\alpha\psi)}$	$\frac{1+A \exp(-\alpha\psi)}{(A/\sqrt{1-\zeta^2}) \sin\psi}$	$\frac{(K\zeta+\sqrt{K^2\zeta^2+1}) [1+A \exp(-\alpha\psi)]}{(A/\sqrt{1-\zeta^2}) \sin\psi}$
2) Attitude Response-Type	Step	$\frac{\phi}{\phi_c} = \frac{\omega_n^2}{s^2+2\zeta\omega_n s+\omega_n^2}$	$\frac{\omega_n \exp[-\alpha(\pi/2-\tan^{-1}\alpha)]}{1+\exp(-\pi\alpha)}$	$\frac{1+\exp(-\pi\alpha)}{\exp[-\alpha(\pi/2-\tan^{-1}\alpha)]}$	$\frac{(\zeta+\sqrt{\zeta^2+1}) [1+\exp(-\pi\alpha)]}{\exp[-\alpha(\pi/2-\tan^{-1}\alpha)]}$

Note:  $\alpha = \frac{\zeta}{\sqrt{1-\zeta^2}}$ ;  $A = (1-2T_p\zeta\omega_n+T_p^2\omega_n^2)^{1/2}$ ;  $\psi = \tan^{-1} \frac{T_p\omega_n\sqrt{1-\zeta^2}}{T_p\zeta\omega_n-1}$ ;  $K = \frac{\omega_{BW}-1/T_p}{\omega_{BW}+1/T_p}$

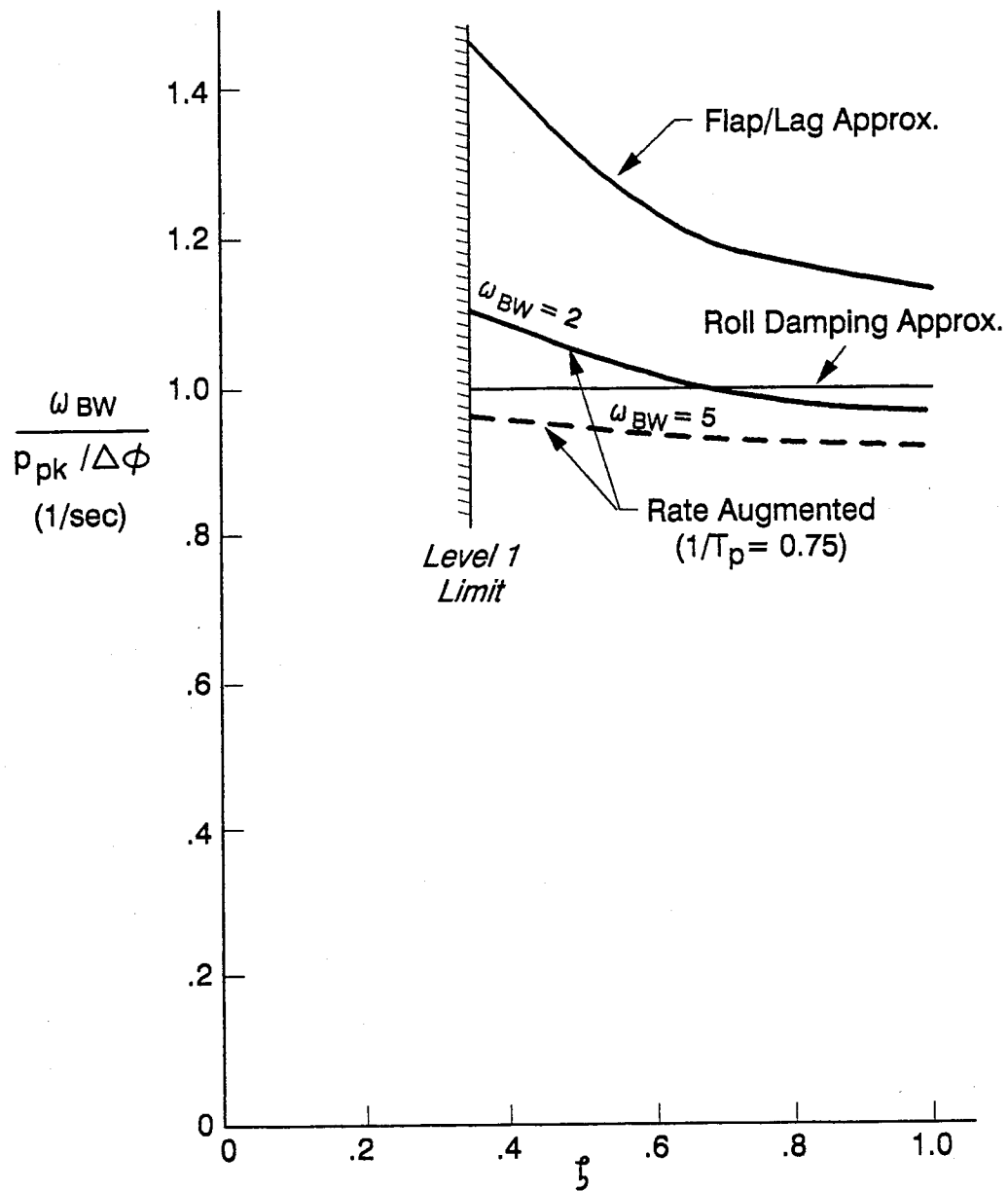


Figure 4(3.3.3). Analytical Relationships for Rate Response-Types (Impulse Input)

about 10 percent. Hence, requirements on attitude quickness are, to a first approximation, requirements on bandwidth.

Ideally, the maneuver to show compliance with this requirement would have the characteristics shown on Figures 1 and 2. As a practical matter, however, the "steady" attitude achieved may be difficult to define due to overshoots and drifting that may be imperceptible to the pilot. Therefore, the attitude change between initiation of the maneuver and first peak is taken as the denominator of the  $q_{pk}/\Delta\theta$  and  $p_{pk}/\Delta\phi$  parameters. The required change from "one steady attitude to another" is taken as the first minimum following the first peak of the response, i.e.,  $\Delta\theta_{min}$  or  $\Delta\phi_{min}$ . This value is plotted as the abscissa on the Figure 4(3.3) requirements. These parameters are defined in the sketch in Figure 4e(3.3). If there is no attitude overshoot, the peak and steady attitude changes are identical. The definitions in Figure 4e(3.3) penalize large attitude overshoots; large attitude overshoots are not consistent with the intention of the requirement, where a crisp attitude change without overshoot is the desired goal.

An exemption from this requirement is given for Response-Types that are provided specifically for use in degraded Usable Cue Environments (UCE = 2 or 3), where the attitudes and attitude changes will be much smaller and the commands more gentle.

#### c. Supporting Data

Most of the supporting data for the attitude quickness requirements come from the flight and simulation study of roll control power, Reference 93. Since this was specifically a roll control study, there is no information for setting pitch limits, and some assumptions have been made for the pitch requirements in Figure 4(3.3). As discussed in Rationale for Requirement, only the roll boundaries of parts b and d of Figure 4(3.3) are fully supported by data -- and in the case of part b, the supporting data are for forward flight (see Supporting Data for Paragraph 3.4.5.2). This section reviews the data upon which Figure 4d(3.3) is based. Following this is a brief comparison of the moderate-amplitude requirements with the small- and large-amplitude requirements to illustrate the interrelationship of these criteria.

#### 1. Supporting Data for Figure 4d(3.3), All Other MTEs (roll)

These limits were developed from maneuver performance data generated in the Vertical Motion Simulator program of Reference 93. Several Rate and Attitude Command/Attitude Hold Response-Types were flown through a lateral sidestep task at hover. Because of the known shortcomings of hovering Rate Response-Types on a ground simulator (see, e.g., the discussions for Paragraphs 3.2.2 and 3.3.2.1), only the Attitude data were used here.

The task performance plots for five ACAH configurations are shown in Figure 5. These plots show characteristics typical of such maneuver performance data: a relatively high performance requirement at small



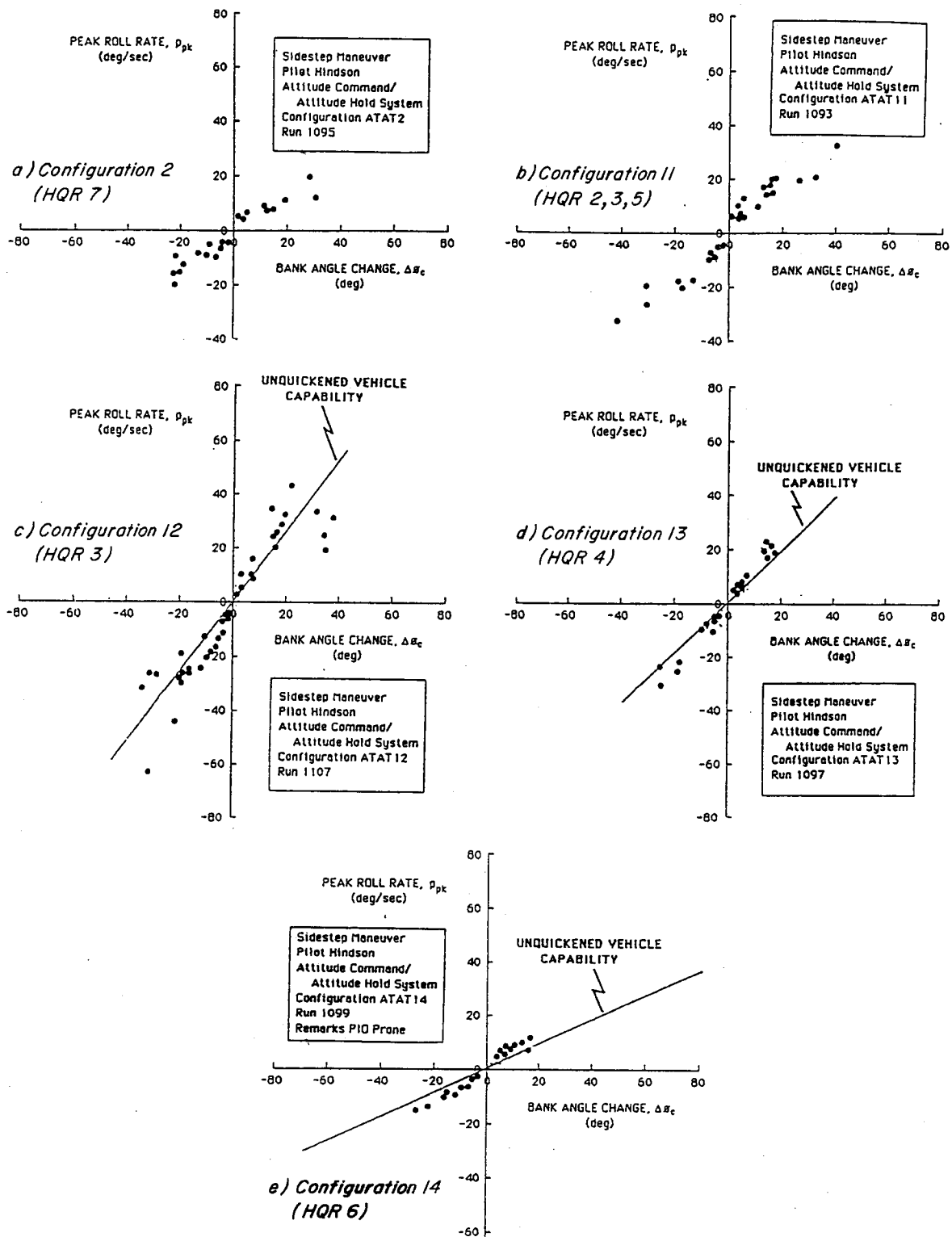


Figure 5(3.3.3). Sidestep Task Performance for Attitude Response-Types from Simulation (Reference 93)

attitude changes (i.e., the effective slope of  $p_{pk}$  vs.  $\phi$  is steep), with a decreasing demand as the bank angle change increases.

The maneuver data from Figure 5 are replotted in Figure 6 using the criterion parameters (points in Figure 5 with  $\phi < 10$  deg are not shown in Figure 6 since it is assumed this range is covered by the bandwidth requirements). The Level 1 limit as drawn corresponds to the range of points for Configurations 11, 12, and 13. All of the configurations exceeded the Level 2 limit; this limit is based more on the small-amplitude bandwidth requirement in Paragraph 3.3.2.1 (0.5 rad/sec) and the peak roll rate requirement of 21 deg/sec, as sketched in Figure 6.

## 2. Comparison with Small- and Large-Amplitude Requirements

Figure 7 shows the requirements of Figure 4(3.3) in comparison with the small-amplitude (bandwidth) requirements of Paragraph 3.3.2.1 and the large-amplitude limits of Paragraph 3.3.4. The bandwidth requirements are re-interpreted here in terms of the attitude quickness criteria; in reality, as Figure 4 showed, the relationships between the bandwidth and attitude quickness parameters are not exactly one-to-one, so the solid circles representing bandwidth limits are only approximations for illustrative purposes. The dashed lines representing the large-amplitude maneuvering requirements have been sketched assuming equal values of minimum and peak attitude changes -- i.e., no overshoots in attitude (see Figure 4e(3.3)).

As Figure 7 illustrates, the moderate-amplitude requirements of this paragraph effectively connect with the bandwidth limits at low amplitudes, and with the peak angular rate limits at very high amplitudes.

### d. Guidance for Application

Because there is a lack of familiarity with this requirement, several examples have been developed to illustrate methods of compliance, as well as some of the issues involved in demonstrating compliance.

#### 1. Examination of Control Inputs Required

Three simple aircraft configurations were mechanized on an analog computer for evaluation with a fixed-base piloted simulation. The models used made no assumptions about such effects as actuator rate limiting, which is discussed separately below. The configurations were as follows:

- Rate, with large roll-rate overshoot (factor of 1.8) which is allowed by the specification, but was found to be troublesome (albeit still rated Level 1 by most pilots) in the Reference 157 experiment. This was included because it might be difficult to

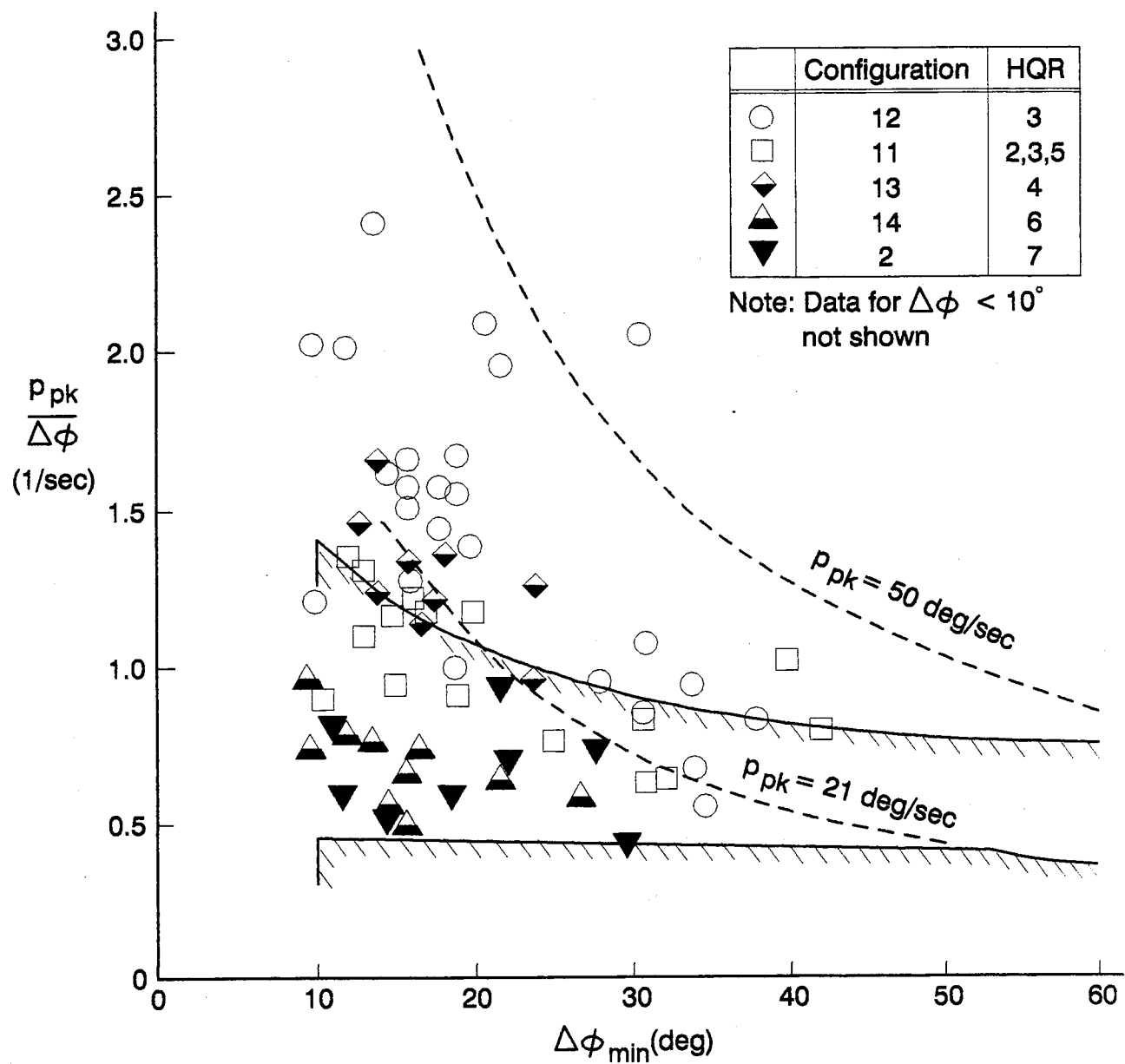


Figure 6(3.3.3). Maneuver Data from Figure 5  
(Supporting Data for Figure 4c(3.3))

- Small-Amplitude Requirement (bandwidth) -- Para. 3.3.2.1
- Moderate-Amplitude Requirement (attitude quickness) -- Para. 3.3.3
- - - Large-Amplitude Requirement -- Para. 3.3.4 (rate response-types)

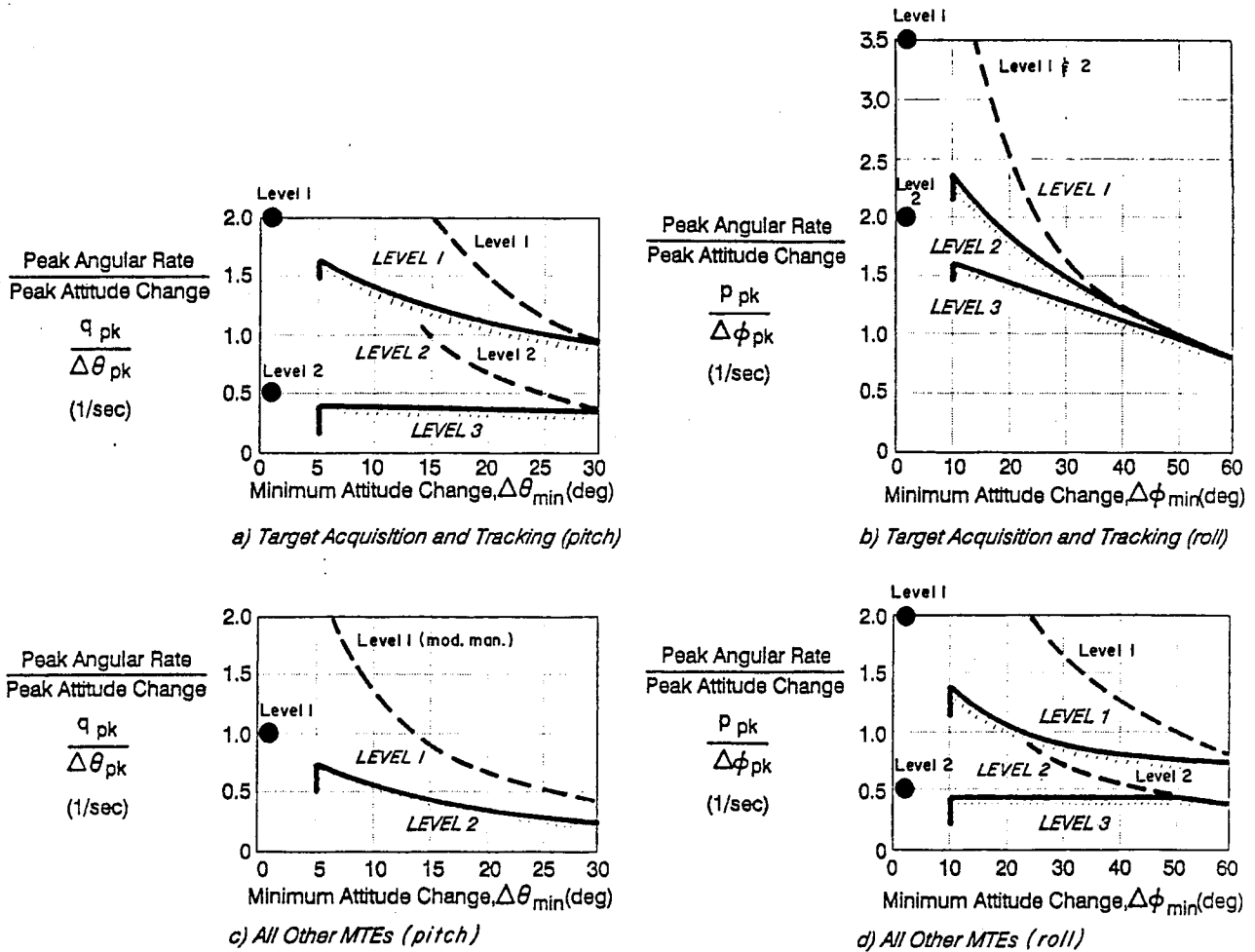


Figure 7(3.3.3). Comparison of Moderate-Amplitude Requirements with Small- and Large-Amplitude Requirements

perform a pure attitude change due to the attitude drop-back characteristics that occur with large angular-rate overshoot. The bandwidth of this configuration is 2.2 rad/sec, therefore it barely meets the bandwidth requirement of 2.0 rad/sec.

$$\frac{\phi}{\delta_{as}} = \frac{2.43}{(0)[0.7, 1.56]} \times \underbrace{\frac{100}{[0.7, 10]}}_{\text{actuator}}$$

- Attitude Command/Attitude Hold (ACAH) with a bandwidth of 2.2 rad/sec (barely meets the bandwidth requirements).

$$\frac{\phi}{\delta_{as}} = \frac{2.43}{[0.7, 1.56]} \times \underbrace{\frac{100}{[0.7, 10]}}_{\text{actuator}}$$

- Rate with an unstable mode at the maximum allowable frequency and minimum allowable damping from the Figure 3(3.3) criterion. This would be representative of a conventional unaugmented helicopter that barely meets the specification. The bandwidth of this configuration is 2.2 rad/sec.

$$\frac{\phi}{\delta_{as}} = \frac{23.5(0.05)}{(4.7)[-0.2, 0.5]} \times \underbrace{\frac{100}{[0.7, 10]}}_{\text{actuator}}$$

The phase delay was 0.12 sec in all cases.

The fixed-base simulator time histories of the maneuvers conducted to demonstrate specification compliance are shown in Figure 8, and the corresponding values of  $p_{pk}/\Delta\phi_{pk}$  are plotted on the criterion boundaries in Figure 9.

The first input for the Rate Response-Type in Figure 8a results in a value of  $p_{pk}/\Delta\phi_{pk}$  that falls below the Level 1 limit, whereas the last two inputs are not only well above the Level 1 boundary, but are close to the actual bandwidth of this simulated configuration (1.9 vs. 2.2 rad/sec). The last two inputs were more like the impulse which was shown to approximate the system bandwidth in Figure 4. Therefore, for Rate Response-Types the recommended control strategy for specification compliance testing is to utilize pulse-like inputs of varying magnitude to produce the necessary range of attitude changes. Note that the change from one steady attitude to another was less than ideal (in spite of maximum pilot effort) due to the attitude drop-back characteristics.

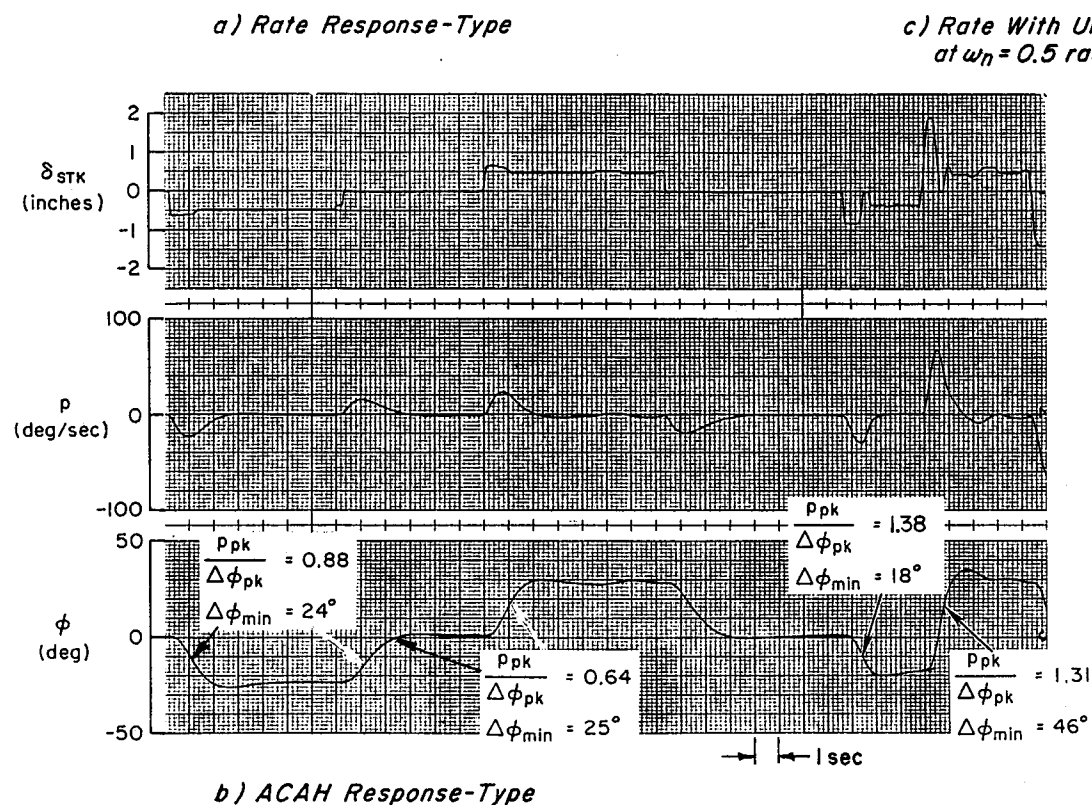
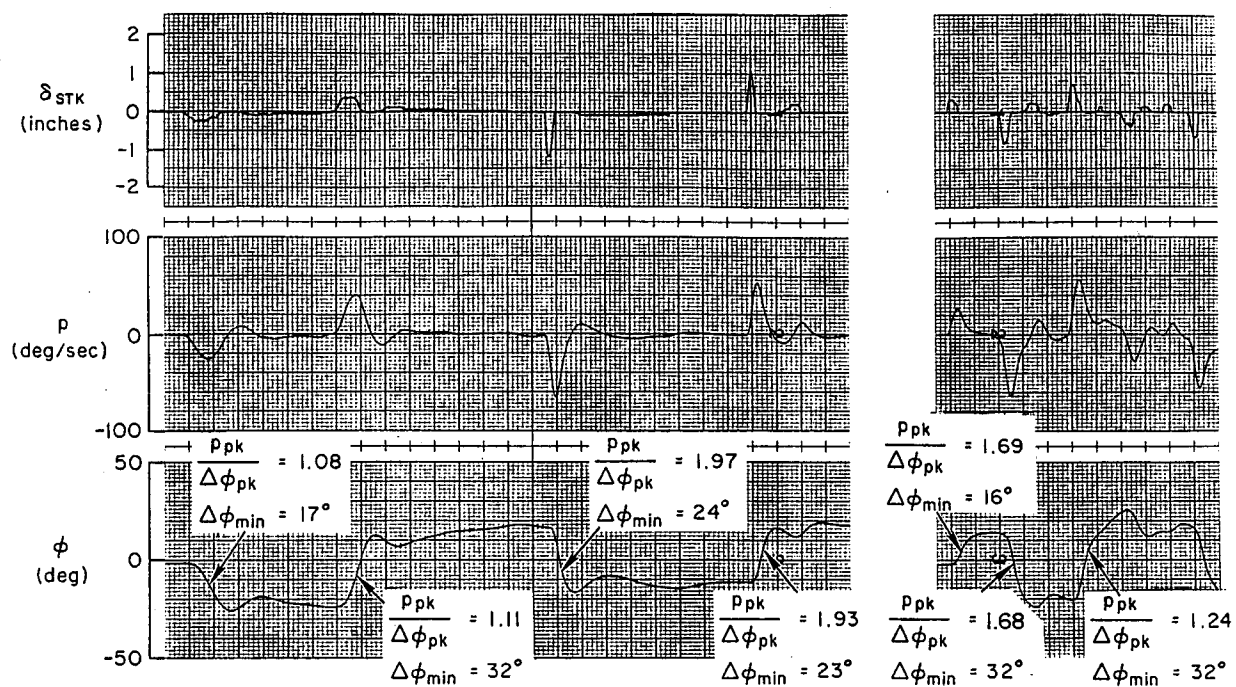


Figure 8(3.3.3). Example of Specification Compliance Maneuvers on a Fixed Base Simulation

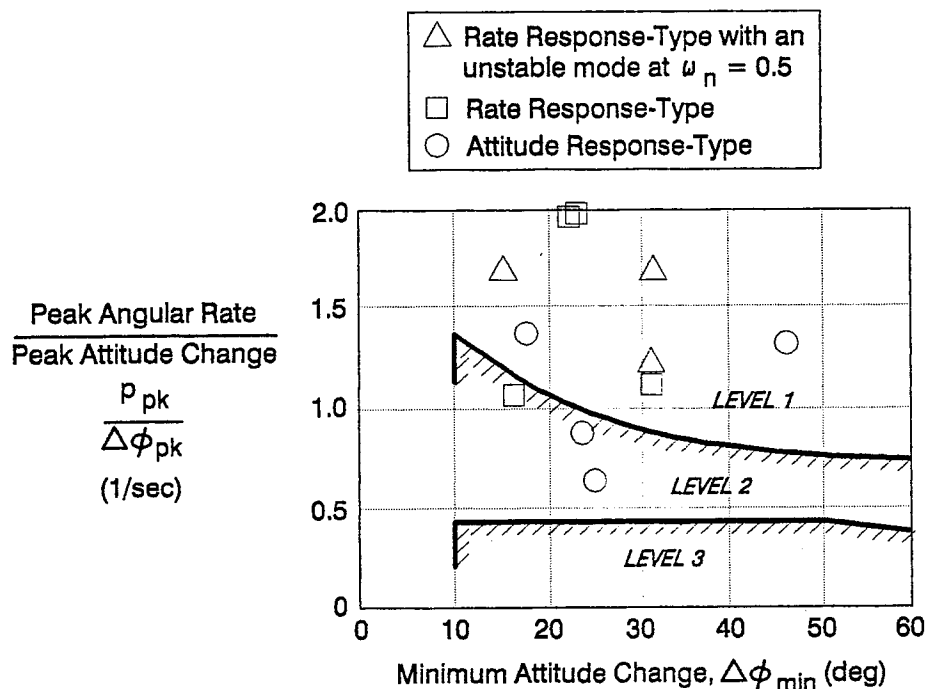


Figure 9(3.3.3). Example Maneuvers from Figure 7  
Plotted on Criterion Boundaries  
(Figure 4d(3.3))

The control inputs all meet the requirement for being essentially one sided.

The first two inputs for the ACAH Response-Type in Figure 8b result in low values of  $p_{pk}/\Delta\phi_{pk}$ . These responses fall below the specification boundaries in Figure 9. When flying an ACAH Response-Type, it is common pilot technique to overdrive the commanded attitude to quicken the initial response, and this technique is demonstrated by the last two inputs in Figure 8b. Significantly increased values of  $p_{pk}/\Delta\phi_{pk}$  are achieved by this method, and the configuration meets the specification in Figure 9. The recommended input for ACAH is to initially overdrive the commanded attitude followed by an essentially steady value of stick consistent with the commanded attitude, as shown in the last two runs in Figure 8b. The purpose of this control strategy is not to provide lead equalization (control changes proportional to perceived angular rate), but simply to overcome the inherent stability of ACAH Response-Types. This is an important distinction because the former control strategy can result in increased workload whereas the latter usually does not.

For fully attended operations in UCE=1, the specification allows some instability as long as the mode is at a frequency less than 0.50 rad/sec and a damping ratio of greater than -0.20 (see Figure 3(3.3) of the specification). Such a configuration was investigated to determine if special control techniques would be required to meet this requirement, which was derived with more highly augmented rotorcraft in mind. A sample of the results is shown in Figure 8c. The most successful

strategy was found to consist of performing a series of back-to-back rolls followed by a stabilized period. It was easier to accomplish the maneuver if the bank angles were not sustained, thereby minimizing the effects of the low frequency instability. Holding the bank angle for any significant period of time (say over two seconds) results in lateral translations (for near hover flight conditions), and hence a new flight condition. If linear transfer functions are being used (as might be the case during the initial tests for compliance on a new design), the region of validity is rapidly exceeded as linear velocities build. Hence the recommended input consists of holding the attitude just long enough to show that the rotorcraft is stabilized, and then rolling (or pitching) in the opposite direction to keep the translational rates small. These observations are equally true for more highly augmented configurations, but the change in flight condition is not so important as the augmentation tends to dominate the response (i.e., the aerodynamics become transparent), which is, in fact, the objective of high gain, high authority feedbacks. A few small, low-frequency control reversals are necessary to stabilize the low-frequency unstable mode. Such reversals are not "significant" in the context of the requirement. Examples of the types of reversals that would be deemed significant are demonstrated in the following discussion.

## 2. Example of "Significant Reversals" in Control Input

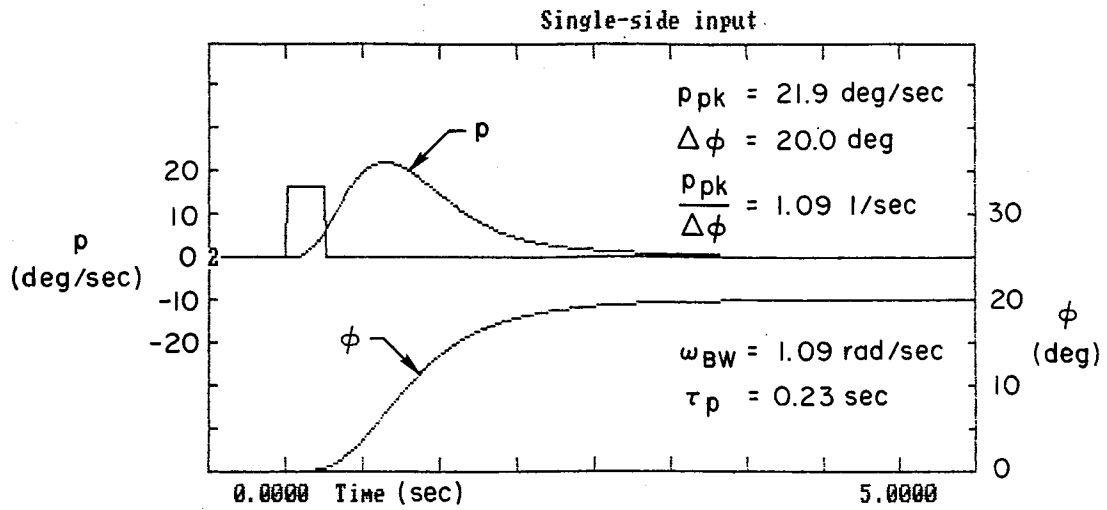
As noted in Section b, Rationale for Requirement, the fundamental relationship between  $p_{pk}/\Delta\phi_{pk}$  and bandwidth is based on simple one-sided control inputs. Misleading results can be obtained if significant control reversals from the trim position are allowed, as illustrated by the following example. Consider the Rate system shown below.

$$\frac{p}{\delta_a} = \frac{50}{(0)(2)[.7,5]}$$

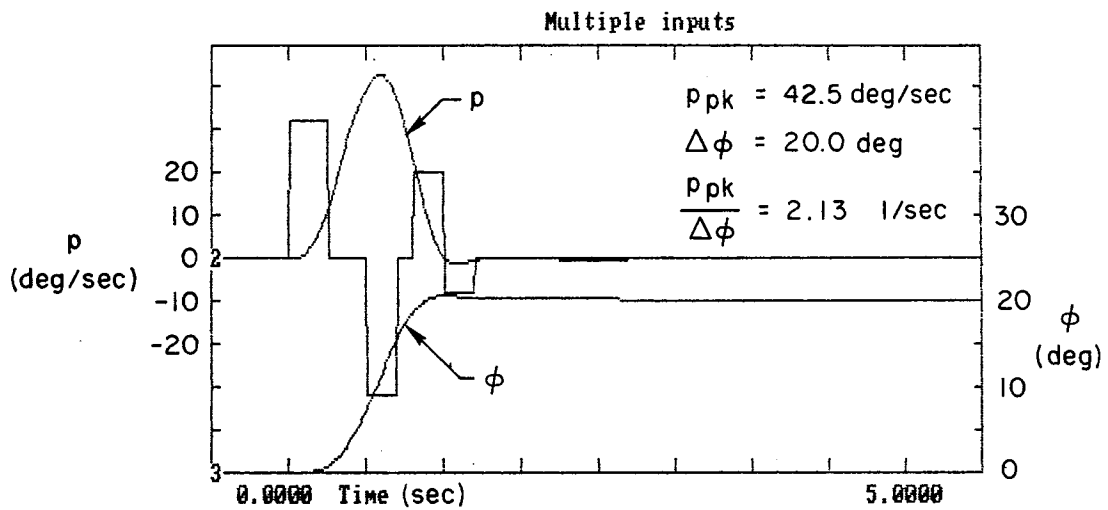
The bandwidth of this system is 1.06 rad/sec, and the phase delay is 0.23 sec, i.e., a sluggish response. The value of  $p_{pk}/\Delta\phi_{pk}$  following the recommended one-sided pulse-like input is 1.09 1/sec (Figure 10a), i.e., it is approximately equal to the bandwidth, as it should be. If the pilot were to employ rapid control input reversals as shown in Figure 10b, the value of  $p_{pk}/\Delta\phi_{pk}$  is increased to 2.13 1/sec. This is not representative of the rotorcraft-alone dynamics, and is more a measure of pilot skill in timing the inputs. In fact, the pilot is closing roll rate and attitude loops just like a SCAS, and of course, paying the penalty in terms of workload. Since the purpose of this requirement is to specify the rotorcraft dynamics without pilot equalization, significant control reversals are not allowed during compliance demonstration maneuvers. Note that the control reversal in this example is the same magnitude as the original input. In general, the control reversals are not significant if the control reversals are significantly less than the initial input.

Obviously there will be cases where some judgement will be required to determine if the control reversals constitute augmentation of bandwidth. Fortunately, such reversals are not necessary for even modest





a) Correct Input



b) Incorrect Input

Figure 10(3.3.3). Example of "Significant" Control Reversal

levels of augmentation, and will only be an issue for simple helicopters with undemanding mission requirements. For modern rotorcraft with a full authority SCAS, control reversals should not be allowed for compliance with Level 1 boundaries.

### 3. Effects of Control Rate and Position Limiting

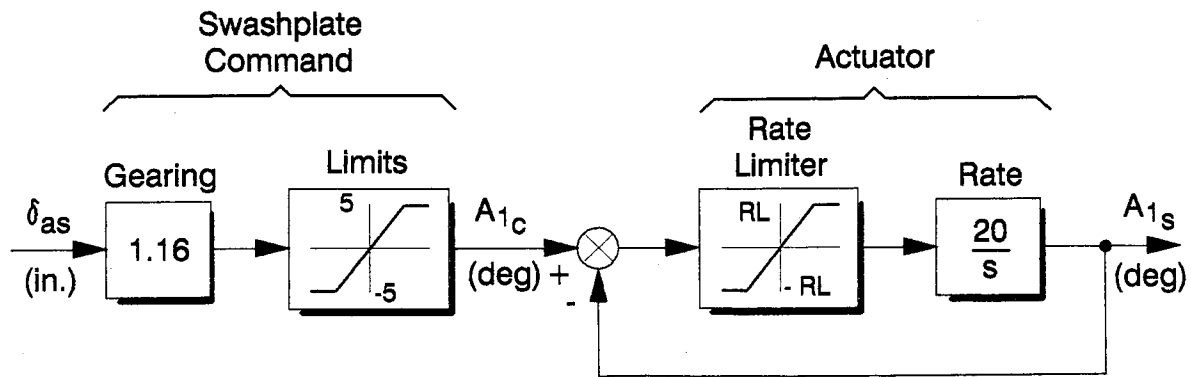
The relationships between bandwidth and  $p_{pk}/\Delta\phi$  illustrated in Figure 4 were derived for a pure impulse input -- i.e., an input with infinite amplitude, applied in zero time. In reality, of course, such an input is impossible; and for some rotorcraft, physical limitations on swashplate travel and actuator rate may greatly affect the rotorcraft's response capabilities. Swashplate travel is an important design consideration, not directly related to handling qualities, that (in combination with other rotor design details) sets the maximum achievable roll rate for the rotorcraft. Actuator rate response is dictated in part by weight and structural considerations, since an extremely rapid, powerful actuator also means an extremely large, heavy piece of equipment, and hence a potentially unnecessary weight penalty.

Control position and rate limiting -- whether it is due to cockpit control design or swashplate limits -- will reduce the maximum achievable  $p_{pk}/\Delta\phi$  capability of a rotorcraft. As an example, a simple roll response model, using the flap/lag approximation of Table 1, was mechanized on a non-real-time simulation with varying actuator rate limits. The dynamics of the model, representative of the B0-105C (from Ref. 93), are documented in Figure 11. Step control inputs ( $\delta_{as}$  in Figure 11a) were applied through a swashplate gearing and position limiter (assuming neutral stick, i.e., stick and swashplate centered), then through a variable rate limiter with a 20-rad/sec first-order actuator lag. The rotor model (Figure 11b) has very high flapping stiffness, resulting in a high-bandwidth response (assuming no limiting, Figure 11c\*) that is Level 1 by Paragraph 3.3.2.1. The swashplate travel limit of  $\pm 5$  deg results in a relatively low maximum achievable roll rate of approximately 58 deg/sec (this is still Level 1 by Paragraph 3.3.4).

Actuator rate limits of 50, 100, and 200 %/sec were simulated (100 %/sec means the actuator travels stop-to-stop, or 100% of authority, in one second). Figure 12 shows typical responses from the simulation for the three actuator rate limits for a 0.75-sec pulse input (full cockpit control applied at  $t = 1$  sec and released at  $t = 1.75$  sec). For the most sluggish actuator, Figure 12a, the pulse is too short to achieve maximum roll rate of 58 deg/sec; a peak roll acceleration of 60 deg/sec<sup>2</sup> is achieved on initiation of the input and an even larger acceleration of -66 deg/sec<sup>2</sup> results when the input is removed; the total bank angle change is 31 deg with negligible overshoot.

---

\*Rate limiting has only a very slight effect on the bandwidth frequency listed in Figure 11c; it greatly increases the phase delay to a value of about 0.2 sec for the largest rate limiting considered here.



a) Swashplate Model

$$\frac{p}{A1_s} = \frac{\frac{1}{\tau_b} L_{b1}}{s^2 + \frac{1}{\tau_b} s + L_{b1}}$$

$$\frac{1}{\tau_b} = \text{Regressive flap mode frequency (rad/sec)}$$

$$L_{b1} = \text{Flapping stiffness (rad/sec}^2\text{/rad)}$$

$$\text{From Ref. 93, for B0-105C: } \frac{1}{\tau_b} = 10.8, L_{b1} = 89.1$$

$$\frac{p}{A1_s} = \frac{962.28}{s^2 + 2(0.57)(9.4)s + 9.42}$$

b) Rotor model

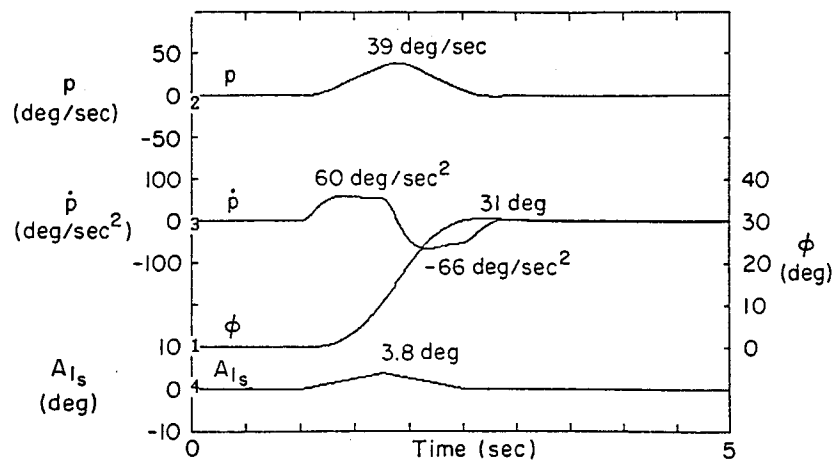
$$\frac{p}{\delta_{a_s}} = \frac{22,325}{(20)[0.57, 9.4]} \quad (\text{deg/sec/in.})$$

$$\omega_{BW_\phi} = 4.3 \text{ rad/sec}$$

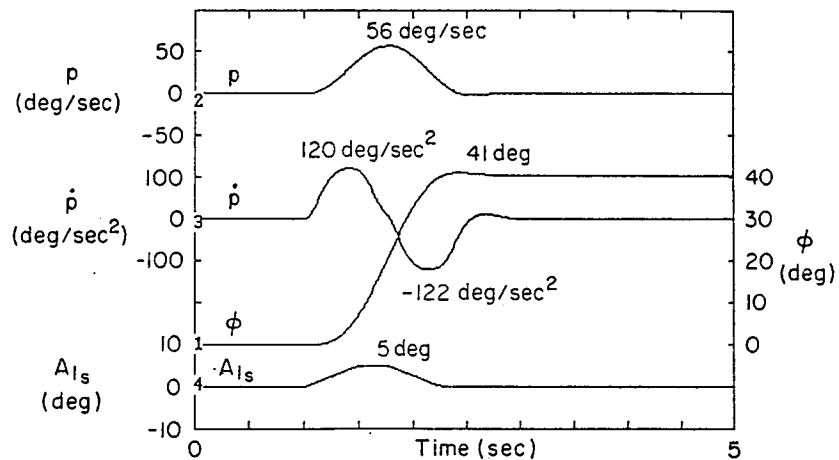
$$\tau_{p_\phi} = 0.09 \text{ sec}$$

c) Linear Rotorcraft Model (No Limiting)

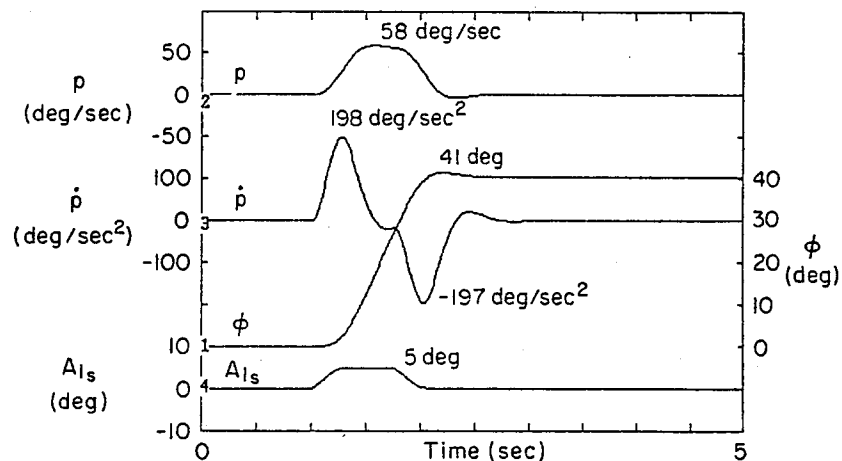
Figure 11(3.3.3). Model of B0-105C for Rate Limiting Evaluation



a) 50 %/sec



b) 100 %/sec



c) 200 %/sec

Figure 12(3.3.3). Example of Response to 0.75-sec Pulse Control Input (Full Control Applied at  $t = 1$  sec)

With a 200 %/sec actuator rate, Figure 12c, the maximum achievable roll rate of 58 deg/sec is reached, but the total bank angle change is almost identical to the 100 %/sec case: the more responsive actuator allows a higher initial acceleration and hence a higher roll rate, but it also provides for a more rapid return to zero roll rate when the input is removed, and thus the roll angle is about the same as for the 100 %/sec case. This situation would, of course, be very different if the swashplate travel limits were increased so a higher roll rate could be achieved.

A full series of pulse inputs was applied to the Figure 11 model for the three actuator rate limits, with the results as shown in Figure 13. Maximum roll acceleration values are noted next to each symbol (a negative sign indicates cases where the maximum acceleration occurred at the removal, rather than the initiation, of the input, as for the examples in Figures 12a and 12b). The dashed line represents the maximum roll rate capability of 58 deg/sec, and the Level 1 limit from Figure 4b(3.3) of the specification is shown for comparison.

Figure 13 shows a significant reduction in roll capability as actuator rate limiting is increased; for the 200 %/sec system, the specification is met easily but at the possible expense of very high roll accelerations that may not be desired, or even achievable in the real world. For this actuator rate, the fundamental limitation in roll response for bank angle changes above about 25 deg is the maximum roll rate capability. With a 100 %/sec actuator, the overall response (in terms of  $p_{pk}/\Delta\phi$ ) is reduced, with a commensurate reduction in peak acceleration, but this system is still Level 1. For the 50 %/sec actuator, the response is simply too slow to meet the Level 1 limit; by 53 deg of bank (the rightmost circle symbol on Figure 13), this rotorcraft has still not reached peak roll rate.

The effect of actuator rate limiting, as illustrated by the foregoing examples, is to reduce the relationship between  $p_{pk}/\Delta\phi$  and bandwidth that was developed earlier. It is not the intent of this Paragraph of the specification to design actuation systems. It is possible, however, that a rotorcraft with the combination of a marginal bandwidth and a sluggish actuator will fail this requirement, and it is the intent of this Paragraph to make certain that such marginal responses do not degrade the overall handling qualities, no matter what the cause.

e. Related Previous Requirements

None.

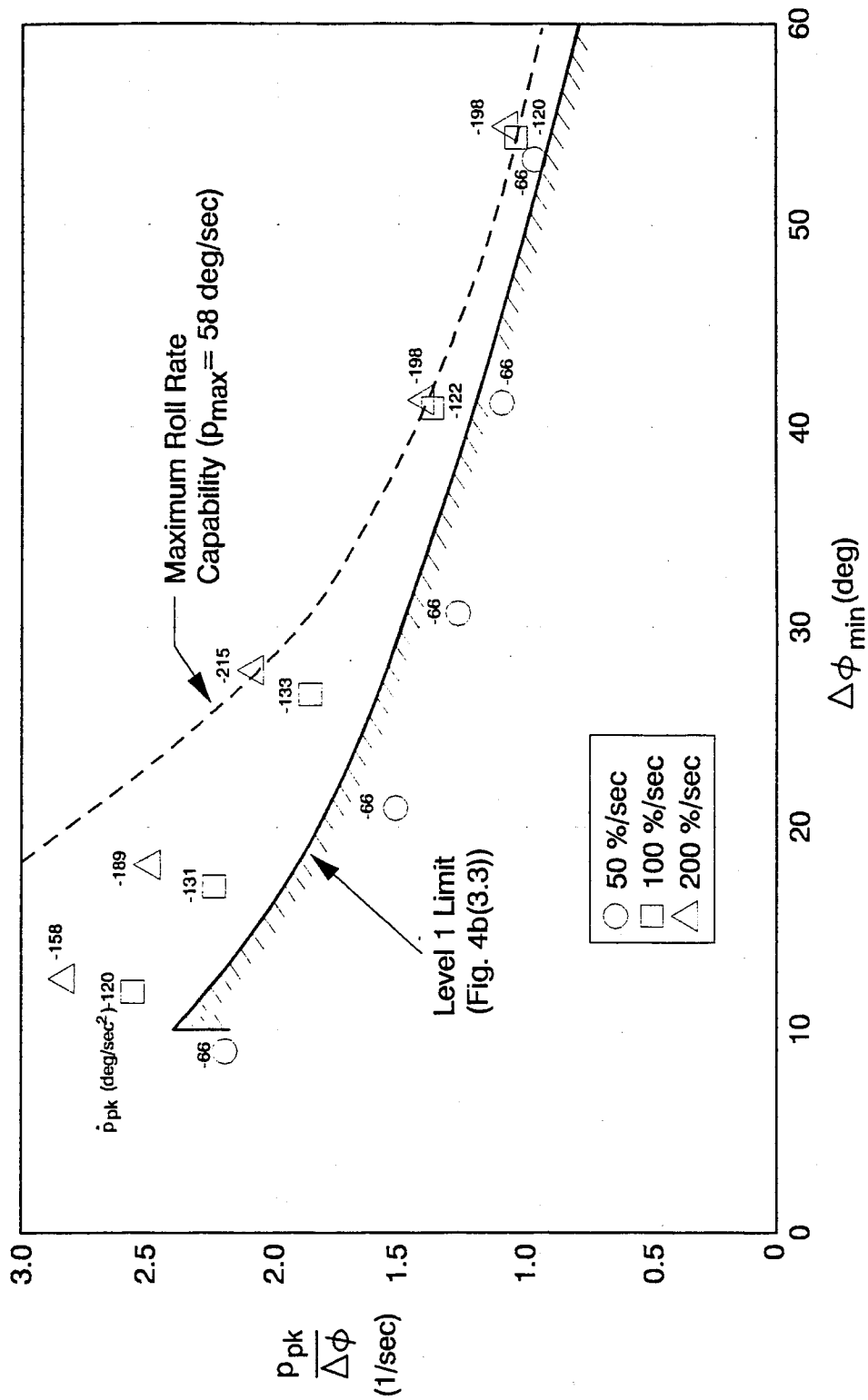


Figure i3(j.3.3). Effect of Actuator Rate Limits on Compliance with Moderate-Amplitude Requirements

a. Statement of Requirement

3.3.4 Large-Amplitude Pitch (Roll) Attitude Changes. The minimum achievable angular rate (for Rate Response-Types) or attitude change from trim (for Attitude Response-Types) shall be no less than the values specified in Table 1(3.3). The specified rates or attitudes must be achieved in each axis while limiting excursions in the other axes with the appropriate control inputs. Response-Types which are designated as applicable exclusively to UCE = 2 or 3 are only required to meet the Limited Maneuvering requirements.

TABLE 1(3.3). REQUIREMENTS FOR LARGE-AMPLITUDE ATTITUDE CHANGES

MISSION-TASK-ELEMENT	RATE RESPONSE-TYPES						ATTITUDE RESPONSE-TYPES			
	MINIMUM ACHIEVABLE ANGULAR RATE (deg/sec)						MINIMUM ACHIEVABLE ANGLE (deg)			
	LEVEL 1			LEVELS 2 AND 3			LEVEL 1		LEVELS 2 AND 3	
	q	p	r	q	p	r	$\theta$	$\phi$	$\theta$	$\phi$
<u>Limited Maneuvering</u> All MTEs not otherwise specified	(a) $\pm 6$	(b) $\pm 21$	$\pm 9.5$	(c) $\pm 3$	(b) $\pm 15$	$\pm 5$	(b) $\pm 15$	(b) $\pm 15$	(b) $\pm 7$	(b) $\pm 10$
<u>Moderate Maneuvering</u> Rapid transition to precision hover Slope landing Shipboard landing	(d) $\pm 13$	(d) $\pm 50$	$\pm 22$	(e) $\pm 6$	(e) $\pm 21$	$\pm 9.5$	(d) $\pm 20$ $-30$	(d) $\pm 60$	(e) $\pm 13$	(e) $\pm 30$
<u>Aggressive Maneuvering</u> Rapid accel and decel Rapid sidestep Rapid hovering turn Rapid slalom Target acquisition and tracking Pullup/pushover Rapid bobup-bobdown	(f) $\pm 30$	(g) $\pm 50$	$\pm 60$	(d) $\pm 13$	(d) $\pm 50$	$\pm 22$	(f) $\pm 30$	(g) $\pm 60$	(d) $\pm 20$ $-30$	(d) $\pm 30$

Note: Letters in parentheses denote subparagraphs in Supporting Data where the limits are discussed.

b. Rationale for Requirement

This requirement is intended to be a measure of control power, specified here in terms of the minimum angular rate or attitude that can be attained. The requirement is divided into three levels of aggressiveness corresponding to the needs of the missions. The Limited Maneuvering requirements of Table 1(3.3) apply to tasks involving precise small-amplitude maneuvering. The Moderate Maneuvering and Aggressive Maneuvering categories apply when a high level of agility is required. A limit on angular rate is not specified for ACAH Response-Types, the philosophy being that this is adequately covered by the moderate-amplitude requirement in Paragraph 3.3.3. The reader is referred to Reference 93 for a more indepth discussion of roll control power.

c. Supporting Data

Many of the control power limits are based on the Reference 85 flight tests conducted in 1964 by Bell Helicopter. These results are summarized in Table 1. The Reference 85 study, while quite old, is one of the few test programs that directly address the issue of aggressive NOE maneuvering in an actual flight environment, and it presents data as to the required angular rates and displacements. Reference 93 represents a more recent simulation, flight test, and analysis program to investigate roll control power and short term response characteristics for large amplitude maneuvering. As pointed out in Reference 93, in most cases the maneuverability limits are more a function of the pilot than of the rotorcraft. It is therefore felt that many of the results obtained in the Reference 85 tests are still valid, even though the tests were performed in a UH-1B and a modified H-13. This is supported by the fact that almost none of the limits are achieved with a 100% cockpit control input. On the other hand, more aggressive maneuvering probably would have been achieved if the handling qualities were better. There is really no effective way to account for this without additional data.

The supporting data and rationale for each of the pitch and roll limits in Table 1(3.3) are discussed in the following subsections. Each subsection is labeled according to the corresponding label found in Table 1(3.3) in this BIUG. Support for the yaw rate limits is given in the supporting data for Paragraph 3.3.8.

a) The pitch rate control power for this group of MTEs is based on the rationale that even large transport and cargo rotorcraft will have to perform a reasonably aggressive flare and recovery to accomplish the "approach to hover" MTE. The Reference 85 tests showed that pitch rates of +5.8 and -5.5 deg/sec were required for flare and recovery (see Table 1).



TABLE 1(3.3.4). SUMMARY OF NOE MANEUVER  
DATA FROM REFERENCE 85

a) SUMMARY OF H-13 NAP-OF-THE-EARTH MANEUVER DATA  
Average Gross Weight 2640 Lbs.

MANEUVER	AIR SPEED	g's	ROLL ATT.	ROLL RATE	LAT. STICK	PITCH ATT.	PITCH RATE	F/A STICK	COLL. STICK
	Knots		Deg.	°/Sec	% From Left	Deg.	°/Sec	% From Aft	% From Down
Hit The Deck	82 39	1.50 .30	25	5	52.5 29.1	-10 -32	10.5	100 58	60 0
Evasive Action	77 50	1.55 .25	60	51	84.1 4.1	- 2 -28	13	92 42	80 10
Reversal of Heading (Fixed Collective)	74 35	1.90 1.00	65	33	72.5 8.3	3 -26	14	77 35	47 47
Reversal of Heading (Variable Collective)	70 49	2.00 1.00	60	21	54.1 8.3	- 9 -25	8	81 42	65 35
Quick Squat	65 0	1.80 .89	53	24	58.3 17.5	2 -19	12	77 15	50 15
Jump-Fire-Run	10 0	2.16 0	5	8	50 18.3	- 9 -13	7	73 42	100 15

b) SUMMARY OF UH-1B NAP-OF-THE-EARTH MANEUVER DATA  
Average Gross Weight 7200 Lbs.

MANEUVER	AIR SPEED	g's	ROLL ATT.	ROLL RATE	LAT. STICK	PITCH ATT.	PITCH RATE	F/A STICK	YAW RATE	PEDAL	RPM (ROTOR)	COLL. STICK
	Knots		Deg.	°/Sec	% From Left	Deg.	°/Sec	% From Aft	°/Sec	% From Left		% From Down
Fort Benning Confidence Course	97 69	1.62 .82	62	49	74 25	+ 5.8 -12.5	+14 - 8	78 45	12	73 37	324 311	65 42
Break and Dive	100 30	1.30 .70	52	12.5	57 35	+15 -24	+ 8.8 - 3.0	61 37	18	90 35	330 299	57 36
Pull Up	93 53	1.88 1.00	37.5	15	61 40	+12 -24	+18.5 - 6.9	59 38	14.2	72.5 61	338 310	50 22
Maximum Descent	90 64	1.2 .5	19	7.0	62 55	+ 6.0 -13.0	+ 4.0 - 5.3	80 48	6.0	93 67	333 324	52 2
Flair and Recovery	88 12	1.55 .95	10	10	68 39	+24.5 - 5.0	+ 5.8 - 5.5	54 28	9.5	94 18	343 309	58 1
Rapid Pedal Reversals for Area Fire	10 0	1.10 .90	26	27	69 34	+13 -10	+ 8.5 - 6.5	58 23	42	73 5	319 316	63 24
Tactical Takeoff and Climbout	85 0	1.25 .85	21	10	58 40	+ 7 -29	+ 9 - 5.5	69 30	13	74 29	328 314	80 12
Quick Stop and Turn into Wind	86 0	1.40 .85	60	17	75 38	+22 -20.5	+18 - 9	70 17	22.2	92 28	333 289	62 0

b) The roll rate and roll angle limits are based on an experiment conducted at NASA Ames in the S01 moving-base simulator (Reference 120). This simulator is ideally suited for hover experiments in that it operates with the pilot looking out at the real world (a parking lot) and with one-to-one motion. The problems associated with visual displays and motion washouts are therefore eliminated. Of course, the restricted motion (effectively a cube with 18 ft sides) did not allow highly aggressive maneuvering. Within that context, the simulation task was to aggressively move laterally from one hover reference to another and back. The task of maneuvering aggressively, but in a very limited area, was taken as representative of the limited maneuvering MTEs. The Reference 120 roll rate data for varying levels of control power for Rate Response-Types is plotted in Figure 1 vs. Cooper pilot rating. These results suggest a 21 deg/sec roll rate limit for Level 1 and a 15 deg/sec limit for Level 2; these values are necessarily approximate because of the scatter in the data.

The Attitude Response-Type data from the Reference 120 experiment are plotted in Figure 2 in terms of the maximum achievable roll angle vs. pilot rating. These data indicate that a Level 1 limit of 13 deg and a Level 2 limit of 6 deg (0.23 radians and 0.1 radians, respectively, in Figure 2) are appropriate. These values have been increased slightly, to 15 deg and 10 deg, for the Table 1(3.3) requirements.

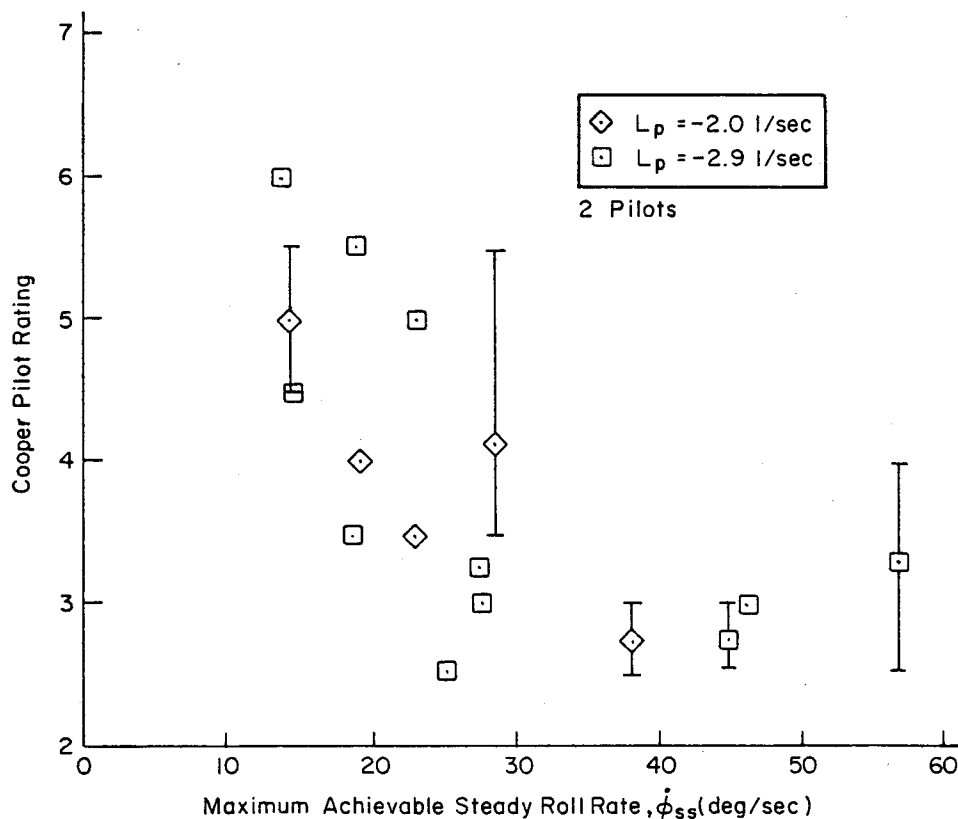


Figure 1(3.3.4). Maximum Achievable Steady State Roll Rate vs. Pilot Ratings from Reference 120 Simulation

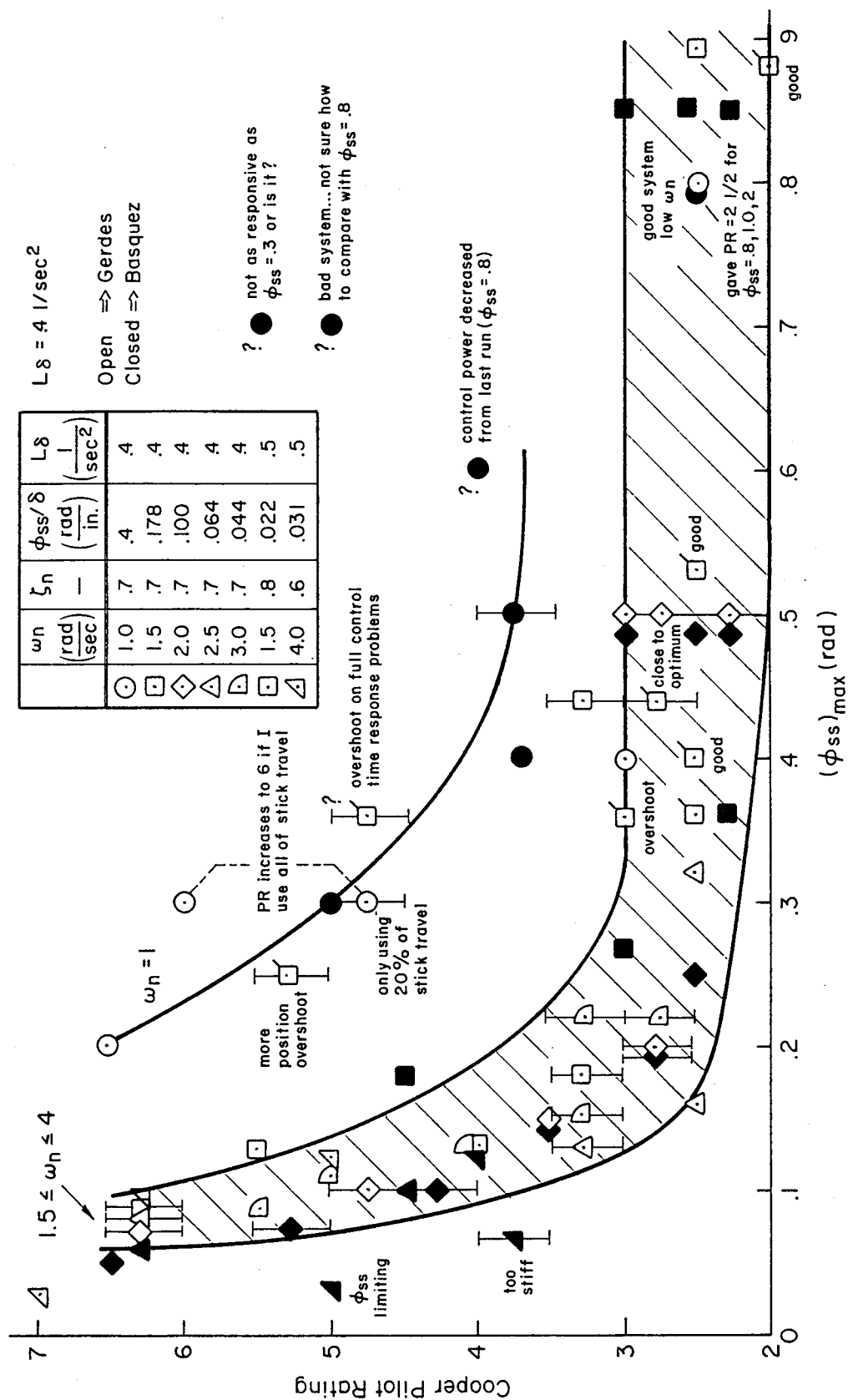


Figure 2(3.3.4). Bank Angle Control Power Data for Maneuvering in a Limited Area.  
Data from Simulation of Reference 120; also see Reference 12

c) This Table 1(3.3) limit is based on converting the MIL-F-83300 VSTOL requirements from time to achieve an attitude in 1 second to an angular rate. This was done assuming a minimum (conservative in terms of control power) level of pitch damping,  $M_q$ , of -2.5 1/sec and proceeding as shown below.

$$\frac{\dot{\theta}(\max)}{\theta(1)} = \frac{-1}{T \left( 1 - e^{-1/T} - \frac{1}{T} \right)}$$

where  $\frac{1}{T} \Rightarrow -M_q$  for conventional helicopters

$$\dot{\theta}(\max) = 1.58 \theta(1)$$

Using this approximation, the MIL-F-83300 requirements for control power convert to an angular rate of 3.2 deg/sec.

d) The Level 1 pitch and roll rate limits, the negative pitch attitude limit, and the roll angle limits are based on the evasive action maneuver data in Table 1a. The +20 deg attitude limit is based on the ability to do a quickstop and turn into the wind (Table 1b), on the basis that this maneuver is an element of an assault landing such as would be required of a utility rotorcraft in combat.

The Level 2 limits for aggressive maneuvering MTEs are based on the Level 1 limits for moderate maneuvering developed above, on the basis that this would allow sufficient agility for the mission to be continued, albeit with degraded performance.

e) The Level 2 requirements for moderate maneuvering MTEs are taken as the Level 1 requirements for the limited maneuvering MTEs, on the basis that the mission could be continued, albeit with degraded effectiveness. However, the Level 2 bank angle for ACAH Response-Types was increased to 30 deg on the basis that it represents a reasonable minimum requirement for survival in a combat environment.

f) This requirement is based on unpublished maneuverability requirements developed by Sikorsky test pilots for advanced attack helicopters. Specifically, the requirement recommends 0.5 g of longitudinal acceleration in 1.5 sec from a steady hover. This translates to a pitch attitude change of -30 deg in 1.5 sec. For a typical rotorcraft attitude response with 130 ms of time delay, and a 20 rad/sec servo actuator, the peak pitch rates required to accomplish the maneuver vary with damping ratio as plotted in Figure 3. A peak pitch rate of 29 deg/sec is required for a closed-loop damping (pilot plus aircraft) of 0.7, which would assure no overshoot. A value of 30 deg/sec is specified in Table 1(3.3).

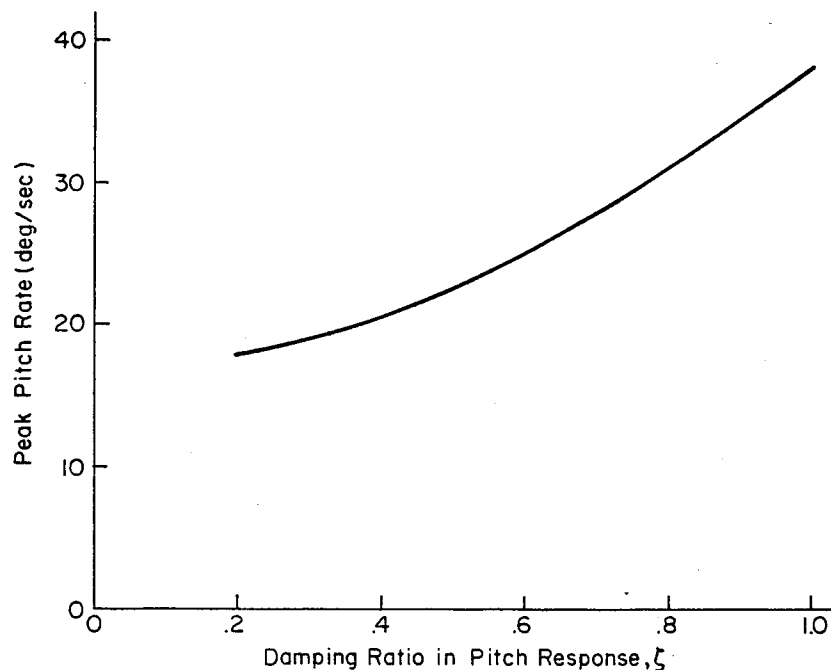


Figure 3(3.3.4). Peak Pitch Rates Required to Change Pitch Attitude by 30 deg in 1.5 sec

g) The required roll rate is based on the Reference 93 study, conducted specifically to examine roll control power. Data for an aggressive sidestep maneuver were obtained in flight using a UH-1H, and in simulation. The UH-1H data indicated a requirement for a roll rate of 40 deg/sec while the simulation (conducted on the NASA Ames Vertical Motion Simulator) indicated 50 deg/sec would be required (see Figure 4). The 50 deg/sec value from simulation is used in Table 1(3.3) on the basis that the handling qualities of the simulated aircraft were probably somewhat better than the UH-1H, and that the pilots were probably willing to be somewhat more aggressive in simulation, as they would also be in actual combat. This result is in agreement with simulated air combat reported in Reference 85 where the highest recorded roll rate was 51 deg/sec. The highest recorded change in roll attitude was 60 deg, which would support a roll attitude control power requirement of  $\pm 30$  deg. This was increased to 60 deg on the basis that bank angles of 55 deg were achieved in low-altitude aggressive maneuvering by the Army Aviation Engineering Flight Activity (AEFA), using a Blackhawk in support of development of the Section 4 maneuvers.

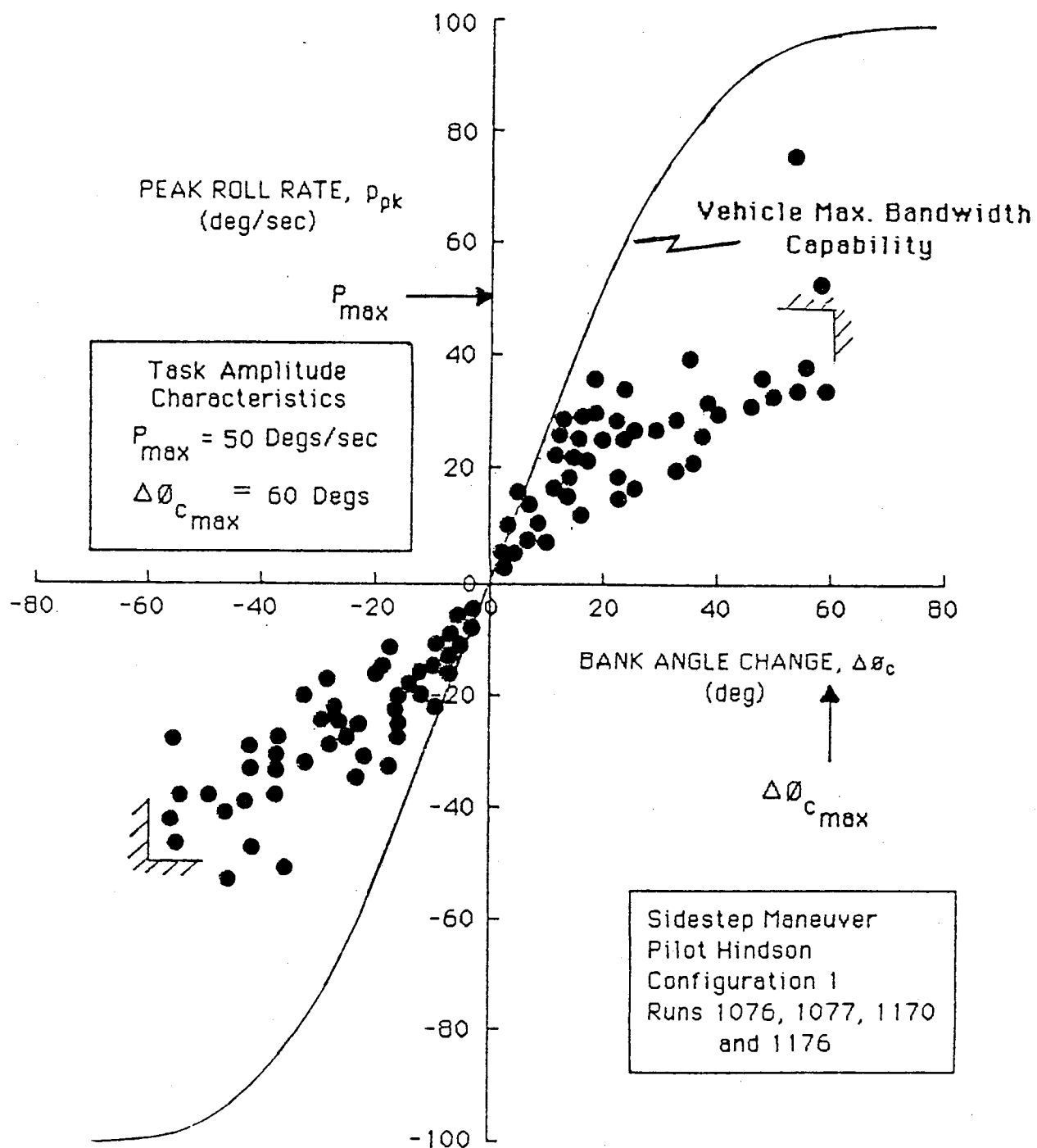


Figure 4(3.3.4). Sidestep Task Performance (from Reference 93)

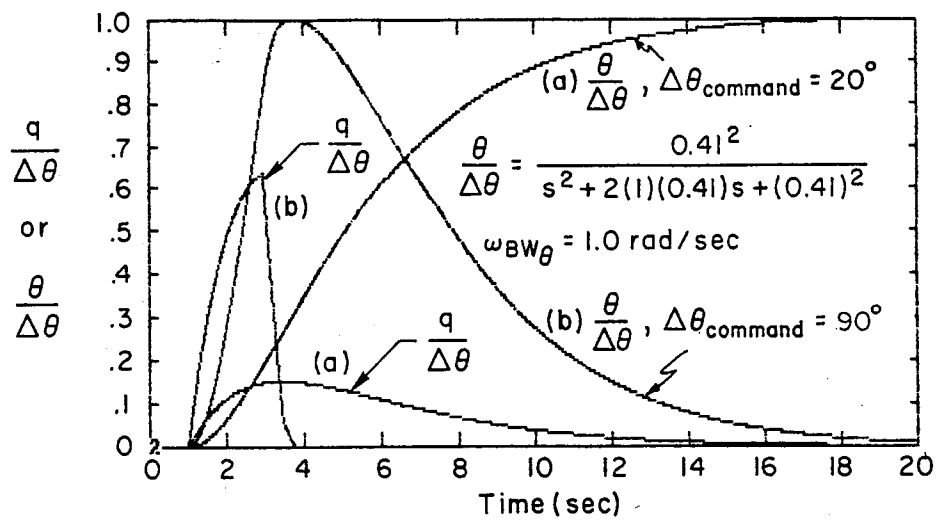
d. Guidance for Application

It is important to remember that the rates and angles specified in Table 1(3.3) are minimum values. Other requirements may, in fact, dictate more agility. For example, an ACAH design that can only achieve the limit attitude given in Table 1(3.3), and no more, will probably have trouble meeting the moderate-amplitude requirements of Paragraph 3.3.3. Consider a rotorcraft operating in an ACAH mode which is limited to 20 deg nose-up pitch attitude at full aft stick, and which is trimmed at an attitude of 0 deg. Furthermore, assume that this helicopter has the minimum allowable bandwidth of 1 rad/sec (Paragraph 3.3.2.1) and a damping ratio of 1.0. The  $q/\Delta\theta$  time history following a full aft-stick input is shown in Figure 5a, curve a, where  $q_{pk}/\Delta\theta = 0.15$ , considerably less than the limit of 0.35 in Figure 4c(3.3). If the pitch attitude limits were increased to allow a total travel of, for example, 90 deg, the pilot could overdrive the initial response to easily meet the requirement. This is illustrated by curve b in Figure 5a, except that in order to achieve the desired 20 deg attitude change, the pilot must remove the input when  $q/\Delta\theta$  exceeds the peak value. Such large inputs could be provided through nonlinear stick shaping or through a dual-mode function that switched to a Rate Response-Type for large inputs.

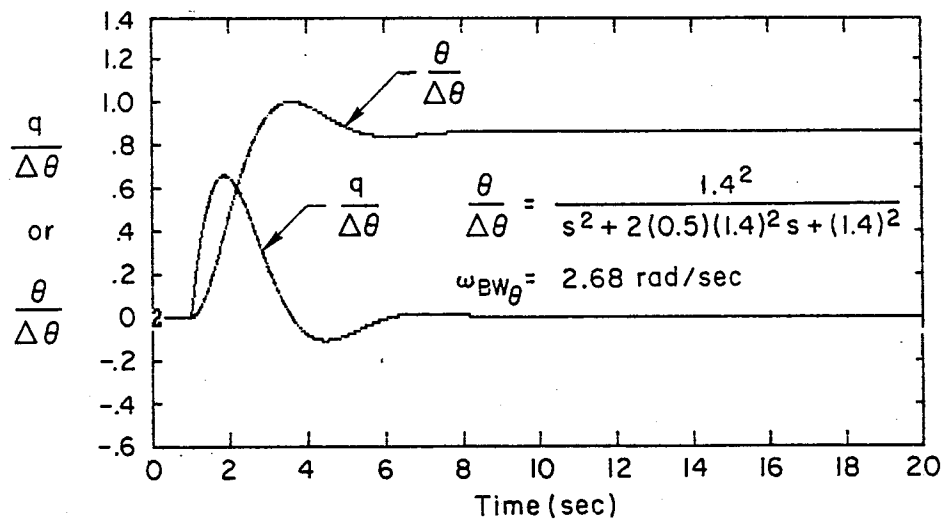
An alternative solution would be to quicken the ACAH response, through either an increased natural frequency, decreased damping, or both, that is, a higher bandwidth; Figure 5b indicates that this could be achieved with a bandwidth of 2.68 rad/sec and a damping ratio of 0.50. This example illustrates the trade-offs available to the flight-control system designer to meet the requirements for small-, moderate-, and large-amplitude maneuvering. Taken together, these criteria are intended to insure that sufficient agility exists, even for flight-control-system modes primarily designed to provide stabilization.

e. Related Previous Requirements

MIL-F-83300 (Reference 29), 3.2.3.1, specifies control power in terms of attitude change in one second or less. The current approach more directly addresses control power from the pilot's viewpoint: the maximum achievable rate or attitude.



a) Achieve  $q_{pk}/\Delta\theta$  by Over Driving  $\Delta\theta$



b) Achieve  $q_{pk}/\Delta\theta$  by Increasing Bandwidth

Figure 5(3.3.4). Response of ACAH Response-Types to Step Controller Inputs at  $t=1$  sec



a. Statement of Requirement

3.3.5 Small-Amplitude Yaw Attitude Changes

3.3.5.1 Short-Term Response to Yaw Control Inputs (Bandwidth). The heading response to directional cockpit control force or position inputs shall meet the limits specified in Figure 5(3.3). The bandwidth ( $\omega_{BW\psi}$ ) and phase delay ( $\tau_{p\psi}$ ) parameters are obtained from frequency responses as defined in Figure 2(3.3).

It is desirable to meet this requirement for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish.

b. Rationale for Requirement

This requirement has been formulated for a Rate Response-Type, typical of most helicopters and required by Paragraph 3.2.2. For unaugmented helicopters at low speeds, heading bandwidth is directly related to yaw damping. Hence, this requirement essentially specifies the yaw damping boundaries. As bandwidth decreases -- e.g., for low yaw damping -- the response becomes more acceleration-like.

This requirement was developed from data using a conventional pedal directional controller. There is no reason to believe, however, that use of a sidestick directional controller would have an effect on the bandwidth required.

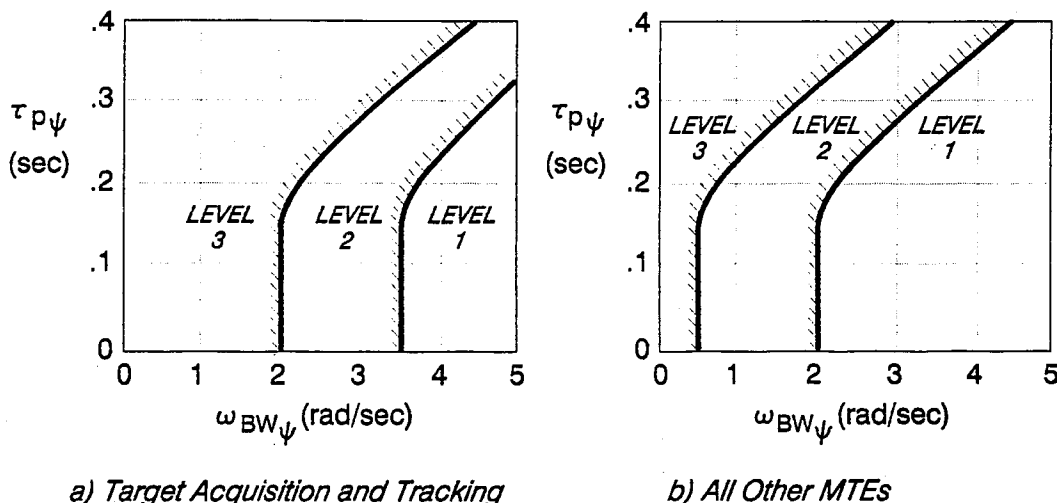


Figure 5(3.3). Requirements for Small-Amplitude Heading Changes -- Hover and Low Speed

c. Supporting Data

The limits for Target Acquisition and Tracking, Figure 5a(3.3), are identical to the limits in roll (Figure 1b(3.3)). The supporting data for these limits are given in the background for Paragraph 3.4.7.1. An investigation into mission-oriented directional control requirements by the Army Aeroflightdynamics Directorate (AFDD) (Reference 106) served as the primary source of data in the formulation of this requirement. Two other investigations into directional control requirements by the National Research Council of Canada (References 113 and 114), together with a study of simulator fidelity (Reference 5), served as secondary sources of data.

The AFDD simulation (Reference 106) was conducted on the Vertical Motion Simulator (VMS) at the NASA Ames Research Center. The effects of different levels of yaw damping ( $N_r$ ) and weathercock stability ( $N_y$ ) on five particular phases of a scout/attack helicopter mission (Mission-Task-Elements) were examined. These elements were as follows:

- Nap of the Earth (NOE) flight
- Deceleration/Quickstop (DECL)
- Low Hover Turn (LHT) (in-ground-effect hover)
- High Hover Turn (HHT) (out-of-ground-effect hover)
- Target Acquisition and Tracking (support for Figure 5a(3.3))

All simulations were run under day VMC conditions, and the helicopter model was sufficiently augmented in pitch, roll, and heave to demonstrate Level 1 handling qualities in those axes. The yaw axis remained unaugmented.

Each test configuration was run with and without turbulence, making it possible to investigate independently -- without the contamination of turbulence -- the effect of yaw rate control/response on pilot ratings. A separate criterion (Paragraph 3.3.7) has been formulated to deal with the effect of gusts on pilot performance.

For analysis, the Mission-Task-Elements have been sorted into three basic groups: hover (deceleration, LHT and HHT), low speed flight (NOE), and target acquisition. Pilot ratings for deceleration, LHT and HHT were averaged and correlated with directional bandwidths ( $\omega_{BW_y}$ ) for the hover flight condition (Figure 1) while the ratings for the NOE phase were correlated with bandwidths at 40 kts (Figure 2). Ratings for the target acquisition and tracking task, which required a relatively higher pilot workload when compared to the other hover tasks, were correlated independently (Figure 3). The calculated bandwidths include corrections for time delays in the VMS resulting from computer frame time and motion and visual system delays, obtained from Reference 49.

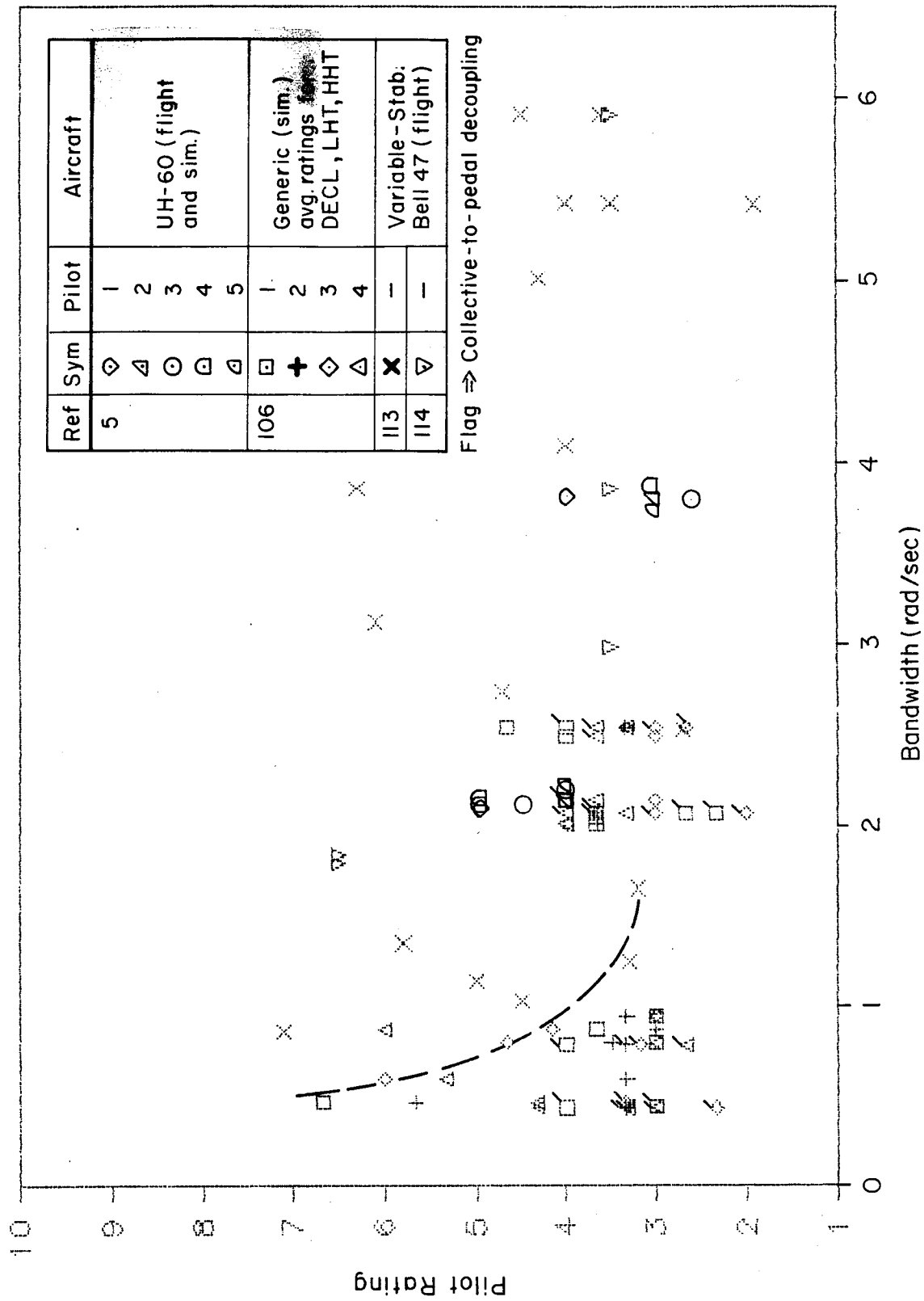


Figure 1(3.3.5.1). Pilot Ratings vs. Heading Bandwidth ( $\omega_{BW_H}$ ) for Hover Tasks. Ratings from References 113 and 114 are Based on the Cooper Scale; All Others Are Based on the Cooper-Harper Scale

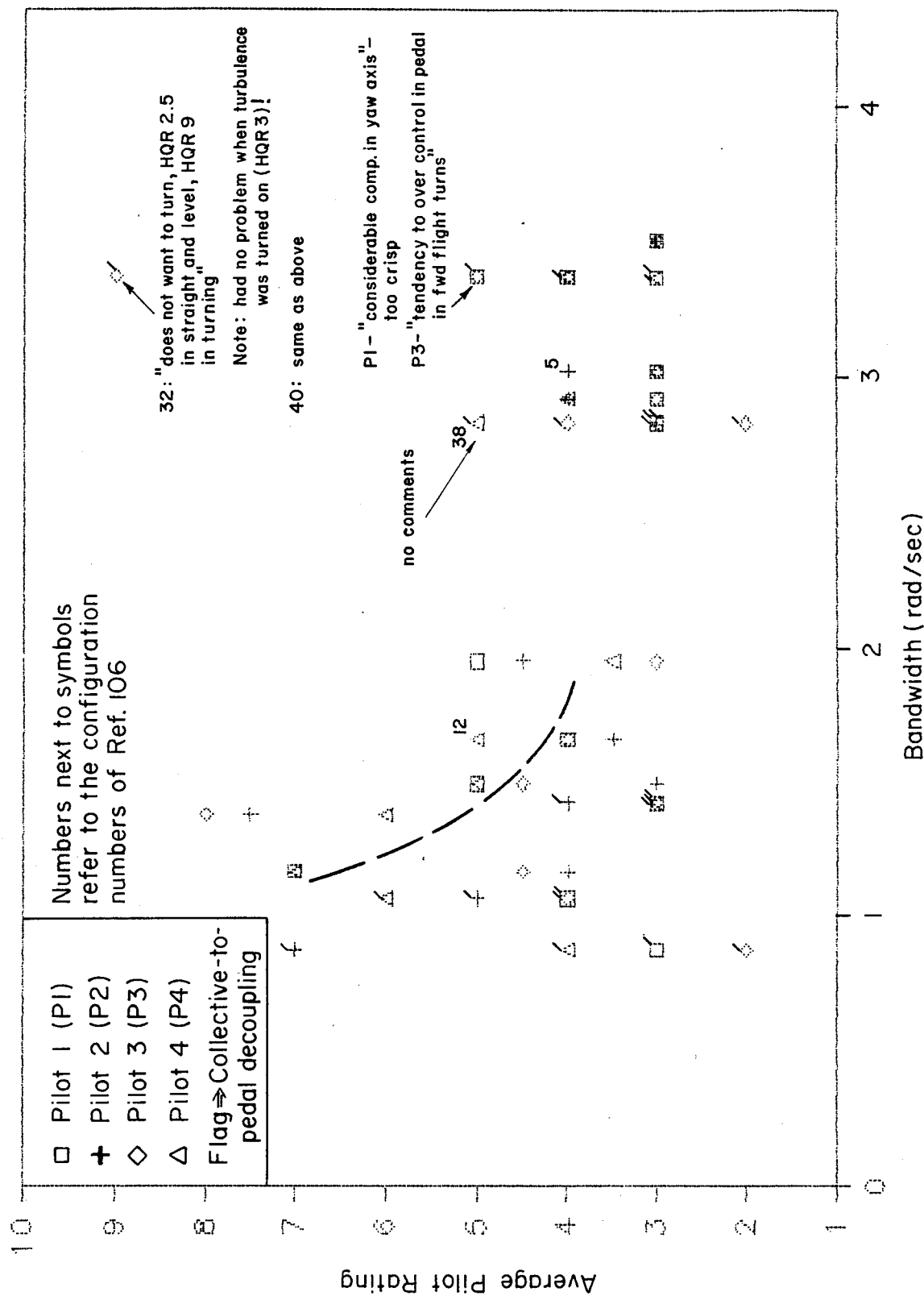


Figure 2(3.3.5.1). Pilot Ratings vs. Heading Bandwidth ( $\omega_{W_{H_0}}$ ) for NOE Flight from Simulation of Reference 106. No Turbulence

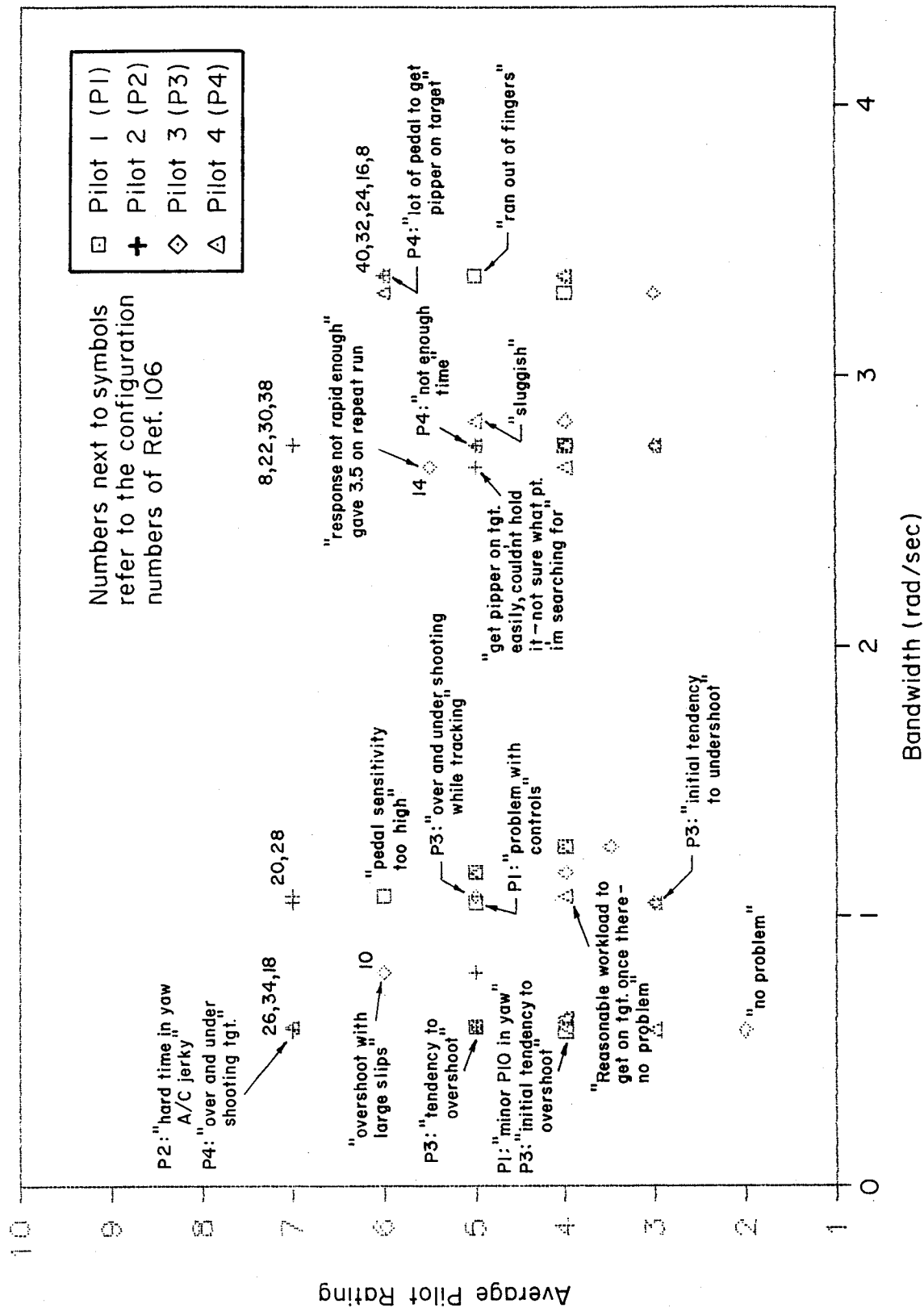


Figure 3(3.3.5.1). Pilot Ratings vs. Heading Bandwidth ( $\omega_{H_y}$ ) for Target Acquisition from Simulation of Reference 106. No Turbulence.

A matrix of damping and sensitivity values was evaluated on the NRC of Canada flight simulator in two separate experiments detailed in Reference 113 and 114.

Pilot rating data for a hover turn task from these two reports are included in Figure 1. The hover turn task was used in both these experiments as well as in the Reference 106 experiment. The data from References 113 and 114 include the effect of turbulence. It should also be noted that the pilot rating scale used in References 113 and 114 was the Cooper rating scale, which is somewhat different from the Cooper-Harper scale used in Reference 106. The Cooper scale is somewhat more harsh for the higher (poorer) ratings, and it is customary (e.g., References 1, 30, and 82) to interpret a Cooper rating of 5.5 (vs. 6.5 on the Cooper-Harper scale) as the Level 2 limit.

Control sensitivity was varied in the Reference 106 VMS experiment in an attempt to keep the steady-state yaw rate approximately constant. Thus, the evaluation pilots had no control over sensitivity. This is the most likely cause of some bad ratings for relatively "good" configurations, as is discussed later in this section. Pilots were advised to give a rating of 6 or worse if the configuration could be turned only  $\pm 40$  deg or less out of the wind. Sensitivity problems may also have affected the ratings of Reference 114.

Individual pilot ratings for hover turns accomplished with a UH-60A (Reference 5) have also been included in Figure 1. The ratings for the low and high bandwidths correspond to the VMS simulation model and actual flight test aircraft, respectively. Comments by the pilots for the flight test phase refer mainly to yaw dynamics, while those for the simulation phase allude to problems other than yaw dynamics since the primary emphasis of the experiment was a comparison of simulator vs. flight, i.e., motion and visual cues, altitude hold problems, and in one case, pedal sensitivity. Due to these differences, the simulator data from Reference 5 were not considered when defining the Level 1 and 2 bandwidth boundaries in Figure 1.

The correlation of pilot rating with bandwidth for NOE flight from the Reference 106 simulation (Figure 2) is very similar to that for the hover tasks. Pilot comments for Pilots 1 and 3, who gave Level 2 ratings for an apparently Level 1 configuration, indicate a problem with pedal sensitivity rather than the dynamics of the configuration. Pilot 1 commented that the response was "too crisp," while Pilot 3 commented that there was a "tendency to overcontrol in pedal." Pilot comments did not help in determining the cause of Pilot 3's rating of 9 for a high bandwidth configuration. However, his assigning of a much more lenient rating (Level 1) with turbulence and a Level 3 rating without turbulence for the same configuration, points to the existence of some problem other than the dynamics of the configuration. This point has therefore been eliminated from consideration when defining the Level 1 boundary.

Several configurations [corresponding to the XV-15 Tilt Rotor and the McDonnell Douglas No Tail Rotor (NOTAR)] were flown with the collective-to-pedal response uncoupled. This would cause a reduction in pilot workload for the NOE and deceleration tasks; these data, shown as flagged points on Figures 1 and 2, indeed account for most of the Level 1 ratings in the low (Level 2) bandwidth range. However, these cases also result in enough poor ratings for the NOE task (Figure 2) to prohibit a decrease in the bandwidth limit when collective-to-pedal decoupling is employed.

With all of the above factors in mind, the Level 1 and 2 boundaries in Figure 5b(3.3) were obtained from Figures 1 and 2. The Level 1 limit ( $\omega_{BW_\psi} \geq 2$  rad/sec) is based on the fact that it represents a knee in the data in Figures 1 and 2. That is, for values of  $\omega_{BW_\psi} > 2$  rad/sec, the ratings are essentially constant, albeit not consistently Level 1. The inability to achieve consistent Level 1 ratings may be attributable to the simulator fidelity issues discussed at length in the discussion for Paragraph 3.2.2, Section 5.

The Level 2 limit ( $\omega_{BW_\psi} \geq 0.50$  rad/sec) is based on an approximate fairing of the pilot rating data (Figures 1 and 2). The NOE data (Figure 2) suggest a slightly higher bandwidth than the hover results (Figure 1).

The target acquisition and tracking task required the pilot to place a pipper on a moving target and attempt a shoot down by firing a simulated missile within an allotted time of 15 seconds. The data for this task, presented in Figure 3, show considerably greater scatter than for the other tasks, and consistent Level 1 ratings were never attained. Pilot comments indicate that the complexity of the task itself tended to preclude Level 1 ratings, i.e., more than minimal compensation was required just to perform the task. Comparison of these data with the Helicopter Air Combat (HAC 3) simulation results presented in the supporting data for Paragraph 3.4.7.1 suggest that the heading bandwidth was not increased enough to provide Level 1 handling qualities. The data of Figure 3 thus (in general) support the limits of Figure 5a(3.3).

Unfortunately, the pilot rating trends that form the basis for this requirement are weak and require considerable interpretation. Attempts to plot this data against other flying qualities metrics such as rise time, natural frequency and damping, etc., were made in an effort to find a better correlation. However, much of the spread is due to vastly different ratings for the same configuration, and no metric will correlate such data. Bandwidth proved to be as good as, if not better than, any other flying qualities parameter.

In the absence of data for determining delay requirements, the shapes of the boundaries in Figure 5(3.3) have been drawn to be consistent with the roll limits of Figures 1b(3.3) and 1e(3.3).

d. Guidance for Application

Since the criterion has been developed in the frequency domain, demonstration of compliance requires determination of the bandwidth frequency. A method for determining bandwidth from flight test data is described in Appendix A.

e. Related Previous Requirements

The directional response requirement of MIL-H-8501A, Paragraph 3.3.19, essentially limits yaw damping ( $N_r$ ) as a function of the moment of inertia about the vertical axis of the helicopter ( $I_z$ ), i.e.,

$$N_r > 27(I_z)^{0.7} \text{ ft-lb/rad/sec}$$

For a conventional helicopter at hover and constrained in pitch, roll, and heave, the approximate yaw-rate-to-pedal transfer function is

$$\frac{r}{\delta_p} = \frac{N_{\delta_p} \cdot s}{s^2 - N_r' s + U_0 N_v' \cos \psi_0}$$

Thus, in the absence of any higher-order dynamics,  $-N_r'$  and  $U_0 N_v'$  essentially determine bandwidth frequency, according to the following relationship:

$$\omega_{BW} = -\frac{N_r'}{2} + \sqrt{\frac{N_r'^2}{4} + U_0 N_v'}$$

In light of the current mission-oriented requirements, bandwidth, which is a measure of the pilot's ability to perform a task, is considered to be a more suitable criterion than moment of inertia, which is a function of size. In addition, bandwidth explicitly accounts for any rotor lags, SCAS dynamics, etc., which can be significant for helicopters.



a. Statement of Requirements

3.3.5.2 Mid-Term Response. The mid-term response characteristics apply at all frequencies below the bandwidth frequency obtained in Paragraph 3.3.5.1. Use of a Heading Hold Response-Type (Paragraph 3.2.6) constitutes compliance with this paragraph, as long as any oscillatory modes following a pulse controller input have an effective damping ratio of at least 0.35. If heading hold is not available, the applicable criterion (Paragraph 3.3.5.2.1 or 3.3.5.2.2) depends on the degree of divided attention necessary to complete the Mission-Task-Elements according to Paragraph 3.1.1.

3.3.5.2.1 Fully Attended Operations. For those Mission-Task-Elements specified in Paragraph 3.1.1 as requiring fully attended operation (Paragraph 2.5.1), the mid-term response shall meet the requirements of Figure 3(3.3), except that the Level 1 limit on effective damping ratio for any oscillation is relaxed from 0.35 to 0.19.

3.3.5.2.2 Divided Attention Operations. For those Mission-Task-Elements specified in Paragraph 3.1.1 as requiring divided attention operations (Paragraph 2.5.2), the limits of Figure 3(3.3) shall be met, except that the Level 1 damping shall not be less than 0.19 at any frequency.

b. Rationale for Requirements

This set of requirements is aimed at assuring a satisfactory heading response at frequencies below the bandwidth frequency covered by Paragraph 3.3.5.1. It is identical to the mid-term pitch and roll requirement in Paragraph 3.3.2.2 except that the  $\zeta = 0.35$  boundary has been replaced with a  $\zeta = 0.19$  boundary in Figure 3(3.3). The primary intent is to place limits on the excursions in heading that the pilot can encounter during normal operations. If Heading Hold is provided, Paragraph 3.2.6 specifies how well it must function. In the absence of Heading Hold, the applicable requirement is determined based on whether the pilot is expected to operate in conditions of divided attention (Paragraph 3.3.5.2.2), or whether the pilot will be expected to devote full attention to flying the helicopter (Paragraph 3.3.5.2.1). The fundamental philosophy behind this division of requirements is described in the discussion for Paragraph 3.3.2.2.

For fully attended operations, Figure 3(3.3) of the specification is applied, with a relaxation on the minimum damping ratio from 0.35 to 0.19 at frequencies  $\geq 0.50$  rad/sec. At  $\omega \leq 0.50$  rad/sec, the requirement is identical to Figure 3(3.3), i.e.,  $\zeta \geq -0.20$ . The limits of Figure 3(3.3) were developed for pitch and roll oscillations (see Supporting Data for Paragraph 3.3.2.2). At low speeds, the oscillations encountered in flight are typically more yaw than roll, i.e., the dutch roll mode occurs mostly in yaw with  $\phi/\beta$  ratios typical of rotorcraft. Since there is no data for dutch roll damping in Low Speed and Hover, the Forward Flight value is used on the format in Figure 3(3.3), for  $\omega \geq 0.50$  rad/sec. At forward flight speeds, the data suggests that Level 1

flying qualities are obtained for a dutch roll damping ratio  $\geq 0.19$ , for all but the most aggressive tasks (see Supporting Data for Paragraph 3.4.8.1).

For divided attention operations, the 0.19 damping ratio requirement is extended to the origin. This eliminates the region of instability at frequencies below 0.50 rad/sec which is allowed for fully attended operations.

c. Supporting Data

There are no heading control data to directly support these requirements. The Supporting Data discussion for roll control (Paragraph 3.3.2.2) presents support for Figure 3(3.3).

d. Guidance for Application

None.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.6 Moderate-Amplitude Heading Changes (Attitude Quickness).

The ratio of peak yaw rate to change in heading,  $r_{pk}/\Delta\psi_{pk}$ , shall exceed the limit specified in Figure 6(3.3). The required heading changes shall be made as rapidly as possible from one steady heading to another and without significant reversals in the sign of the cockpit control input relative to the trim position. It is not necessary to meet this requirement for Response-Types which are designated as applicable only to UCE=2 or 3.

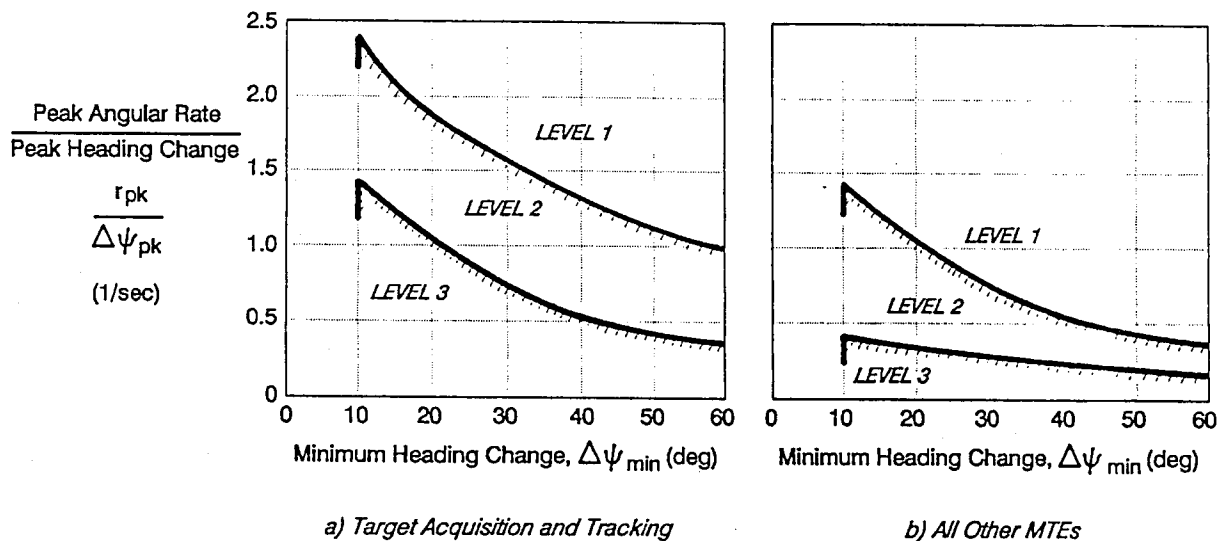


Figure 6(3.3). Requirements for Moderate-Amplitude Heading Changes -- Hover and Low-Speed

b. Rationale for Requirement

There is considerable evidence that a pilot will accept a reduction in response bandwidth for large control inputs, and the moderate-amplitude response requirements of the specification are based on this premise. Supporting Data for Paragraph 3.3.3 show such a reduction for roll, and the yaw requirement (Paragraph 3.3.6) is included on the assumption that this is also valid for yaw control.

Figure 1 illustrates the interrelationship between the requirements of this paragraph and those of Paragraphs 3.3.5.1 (small-amplitude control) and 3.3.8 (large-amplitude control). In the absence of supporting data, the Level 1 limit for Target Acquisition and Tracking (Figure 6a(3.3) -- see Figure 1a) is similar to the roll limit in Figure 4b(3.3) for small heading changes, and blends into the yaw-rate

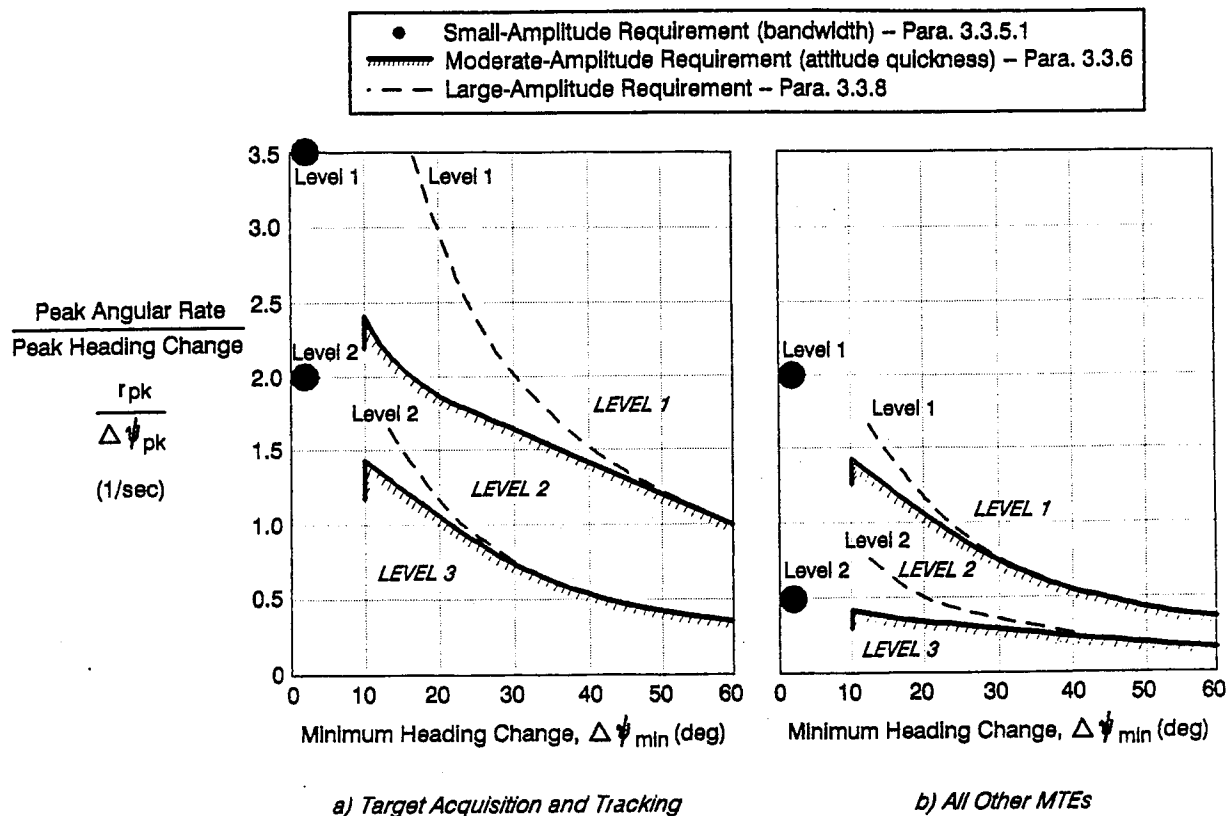


Figure 1(3.3.6). Comparison of Small-, Moderate-, and Large-Amplitude Yaw Requirements

requirement of Table 1(3.3) ( $r > 60$  deg/sec) for large heading changes. The Level 2 limit in Figure 6a(3.3) is almost entirely based on the Level 2 yaw rate limit of  $r > 22$  deg/sec in Table 1(3.3).

For small-amplitude yaw responses, Paragraph 3.3.5.1, the Level 1 bandwidth limit for All Other MTEs is identical to the Level 2 limit for Target Acquisition and Tracking, Figure 5(3.3); and for large-amplitude responses, Paragraph 3.3.8, the Level 1 yaw rate limit for Moderate Maneuvering is the same as the Level 2 limit for Aggressive Maneuvering, Table 1(3.3). On this basis, the Level 1 moderate-amplitude limit in Figure 6b(3.3) is identical to the Level 2 limit of Figure 6a(3.3). The Level 2 limit in Figure 6b(3.3) is approximately equal to bandwidth at the low end, and merges with the yaw rate requirement of  $r > 9.5$  deg/sec for larger heading changes.

#### c. Supporting Data

No data are available at this time. Maneuver performance plots, similar to those presented in the supporting data for Paragraph 3.3.3, are desirable to refine the Figure 6(3.3) boundaries.

d. Guidance for Application

The guidance for applying Paragraph 3.3.3 should be consulted.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.7 Short-Term Yaw Rate Response to Disturbance Inputs. Yaw response to inputs directly into the control surface actuator shall meet the bandwidth limits of Paragraph 3.3.5.1. If the bandwidth and phase delay parameters based on inputs to the control surface actuator can be shown by analysis to meet the cockpit control input bandwidth requirements, no testing is required. This requirement shall be met for Level 1 only.

b. Rationale for Requirement

It is possible, through command shaping, to achieve bandwidths in response to disturbances (i.e., turbulence) that are different from those for control inputs. This paragraph requires that, for Level 1, the disturbance-input bandwidths meet the control-input limits. It allows a relaxation for degraded operations (Levels 2 and 3), where command shaping may be employed to improve the control-response bandwidth with no corresponding improvement in the disturbance-rejection bandwidth.

c. Supporting Data

See the discussion for Paragraph 3.2.12.

d. Guidance for Application

See the discussion for Paragraph 3.2.12.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.7.1 Yaw Rate Response to Lateral Gusts. The yaw rate response following a step lateral gust input shall not exceed the limits of Table 2(3.3). The parameter  $r_{pk}$  in Table 2(3.3) is defined as the peak yaw rate within the first three seconds following the input.

TABLE 2(3.3). MAXIMUM ALLOWABLE YAW RATE  
RESPONSE TO LATERAL GUSTS

MISSION-TASK-ELEMENT	$r_{pk}/V_g$ $\left\{ \begin{array}{l} \text{deg/s} \\ \text{m/s} \\ \text{deg/sec} \\ \text{ft/sec} \end{array} \right\}$	
	LEVEL 1	LEVEL 2
Target acquisition and tracking	0.66 [0.20]	2.95 [0.90]
All MTEs not otherwise specified	0.98 [0.30]	3.3 [1.0]

Demonstration of compliance may be shown by analysis or simulation. The magnitude of the lateral gust input shall be varied from 10 to 25 knots (5.1 to 13 m/s) in the presence of a steady wind of up to 25 knots (13 m/s) from the most critical direction, except that the total wind velocity need not exceed 35 knots (18 m/s). The largest value of  $r_{pk}/V_g$  shall be used for comparison with Table 2(3.3).

b. Rationale for Requirement

This requirement is designed to limit heading deviations due to lateral gust disturbances.

Conventional helicopters are inherently responsive to gusts in the directional axis in hover and in forward flight. Hence, the pilot work load is very high in the directional axis in a gusty environment. A comparison of pilot ratings from the simulation of Reference 106 (discussed in Supporting Data for Paragraph 3.3.5.1) with and without turbulence shows a significant degradation in pilot rating due to turbulence. The average ratings for NOE flight are shown in Figure 1. When operating in a tailwind, the helicopter is directionally unstable to gust inputs ( $C_{ng}$  is negative) and hence  $r_{pk}/V_g$ , in the presence of steady winds from the most critical direction, translates to a steady tailwind for configurations with tail rotors or aft-mounted vertical fins. In that flight condition, the only way to keep  $r_{pk}/V_g$  low is with yaw axis stability augmentation, and the tendency for the yaw axis SAS

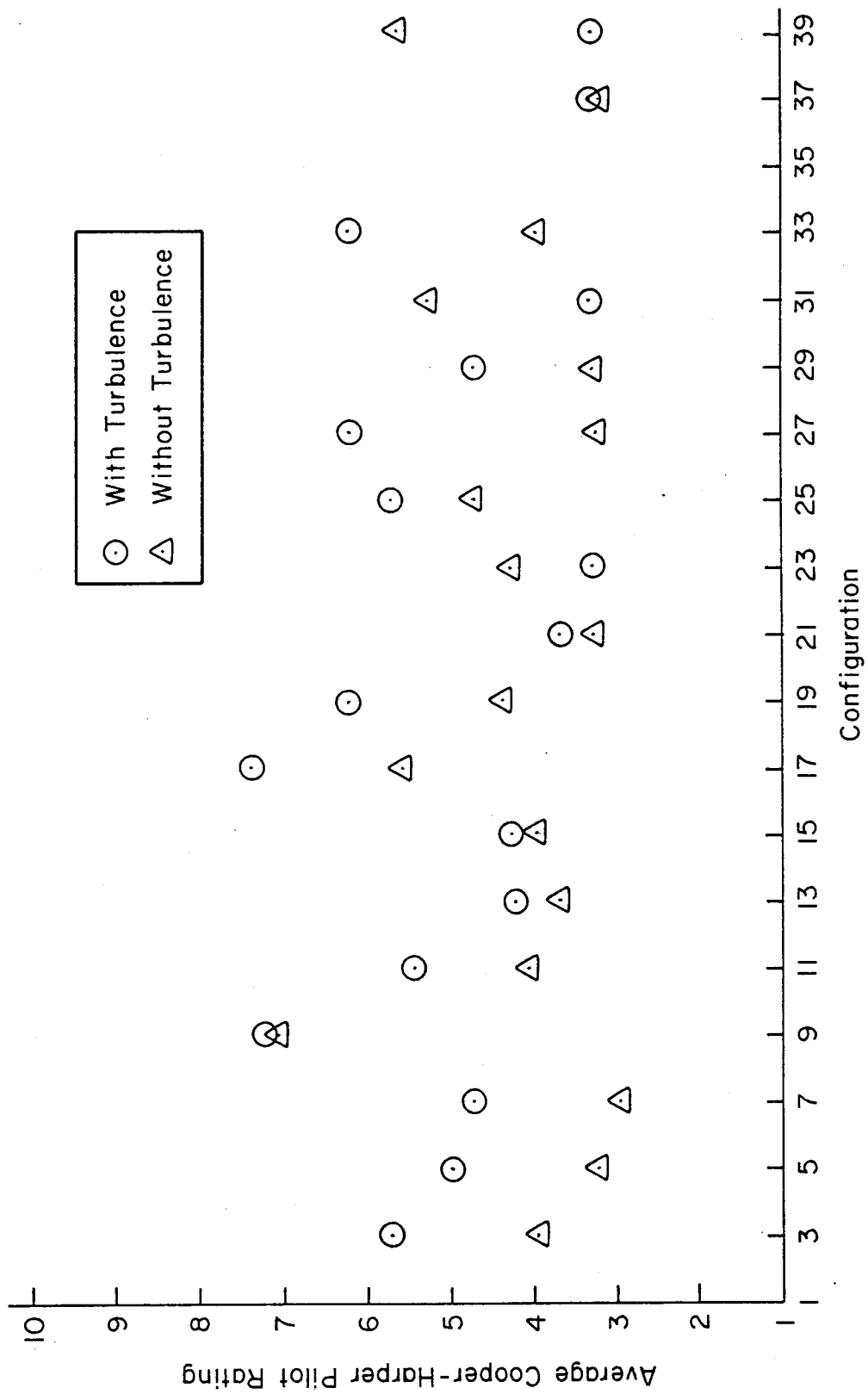


Figure 1(3.3.7.1). Effect of Turbulence on Pilot Ratings for NOE Task from Simulation of Reference 106



to saturate will be greatest in a tailwind. It is felt that the most stringent Mission-Task-Element demands low gust response to keep pilot workload at an acceptable level.

c. Supporting Data

The primary source of data is Reference 106 (see Supporting Data for Paragraph 3.3.5.1). Only data with a heading bandwidth greater than 2 rad/sec were used for pilot rating correlations with  $r_{pk}/V_g$  so the effects of the command/response are minimized. The averaged ratings for the three basic Mission-Task-Elements considered are shown in Figure 2. The maximum and minimum Cooper-Harper pilot ratings given for each configuration are also shown. The figures show a weak but observable trend.

The ratings for the NOE task (Figure 2c) show approximately the same sensitivity to  $r_{pk}/V_g$  as for the hover tasks (Figure 2a). The hover data in Figure 2a were used to set a Level 1 limit of 0.3 deg/sec/ft/sec, based on an apparent knee in the data at that value.

If the Mission-Task-Element requires the use of heading for weapon aiming a more stringent Level 1 limit of 0.20 deg/sec/ft/sec is required, based on the target acquisition and tracking ratings of Figure 2b.

The Level 2 limits are based on extrapolating the pilot rating correlations to a value of 6-1/2.

d. Guidance for Application

Demonstration of compliance is only possible through analysis or simulation.

e. Related Previous Requirements

A requirement on directional response to a disturbance input is stated in MIL-H-8501A (Paragraph 3.6.1.2) in terms of the damping of the oscillatory response. This is essentially a requirement on yaw damping ( $N_r$ ), covered by (Paragraph 3.3.5.1) in this specification.

There are no previous requirements on the magnitude of the response to a disturbance input.

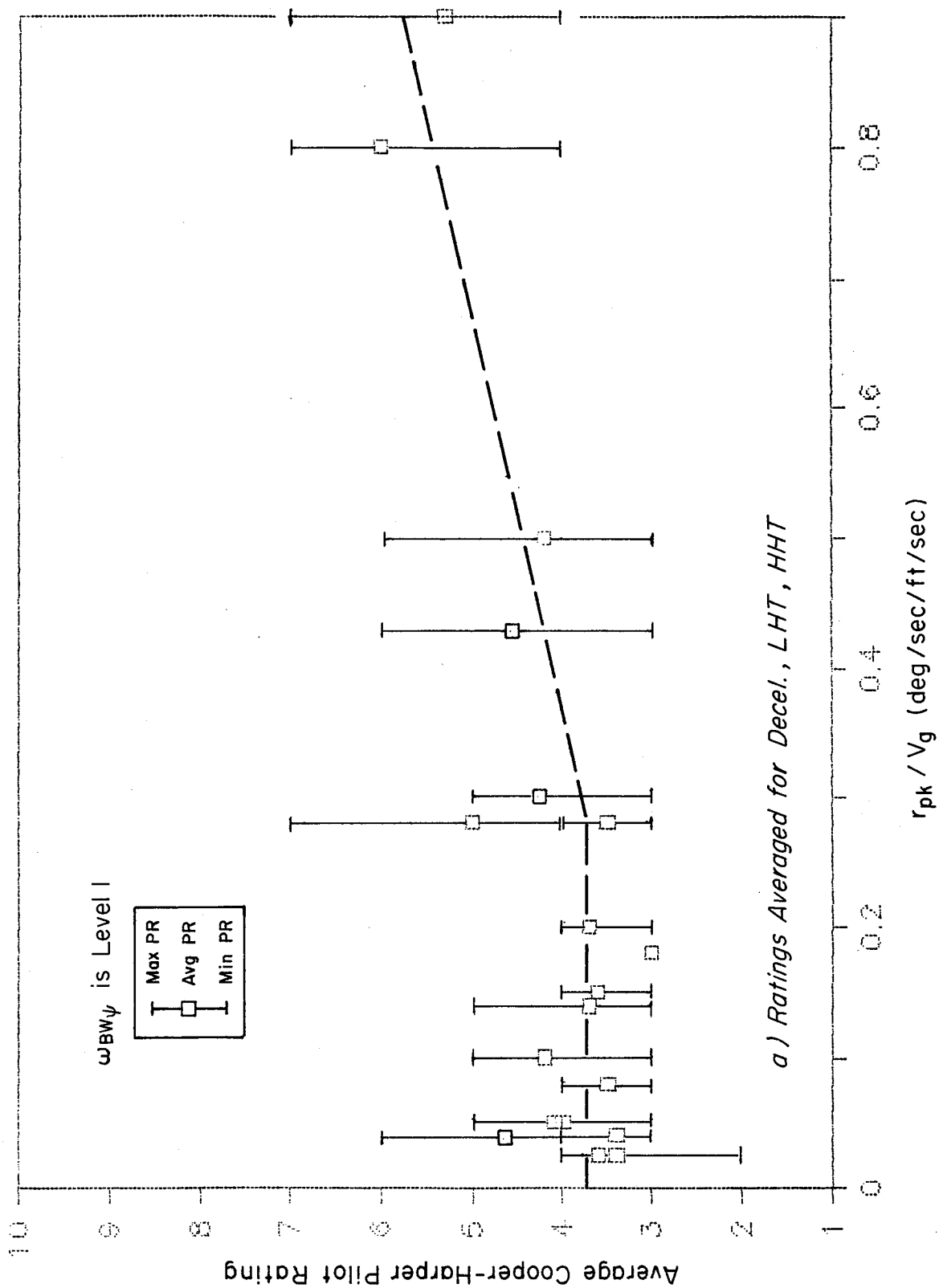


Figure 2(3.3.7.1).  $r_{peak}/V_g$  vs. Average Pilot Rating  
from Simulation of Reference 106

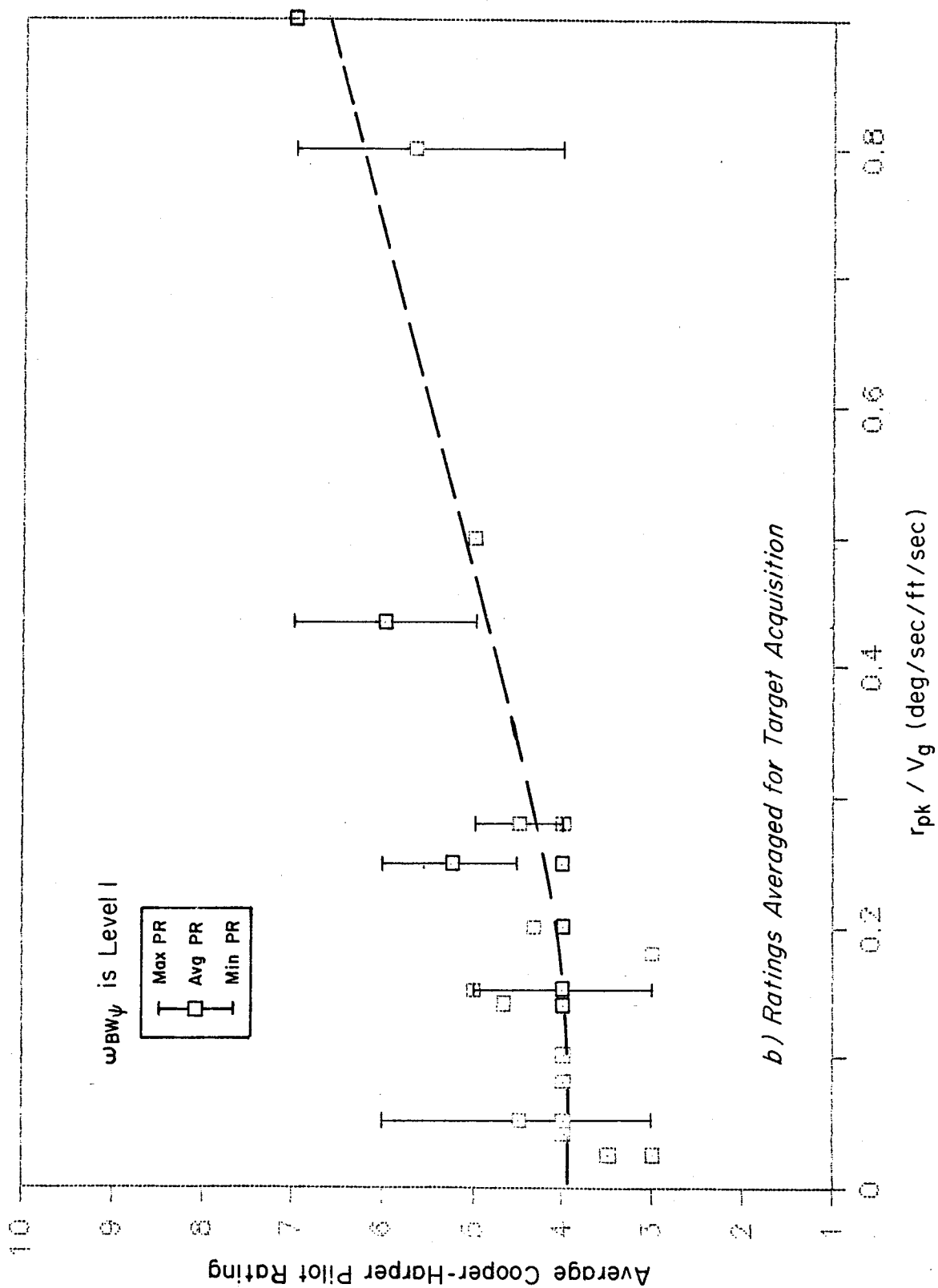


Figure 2(3.3.7.1). (Continued)

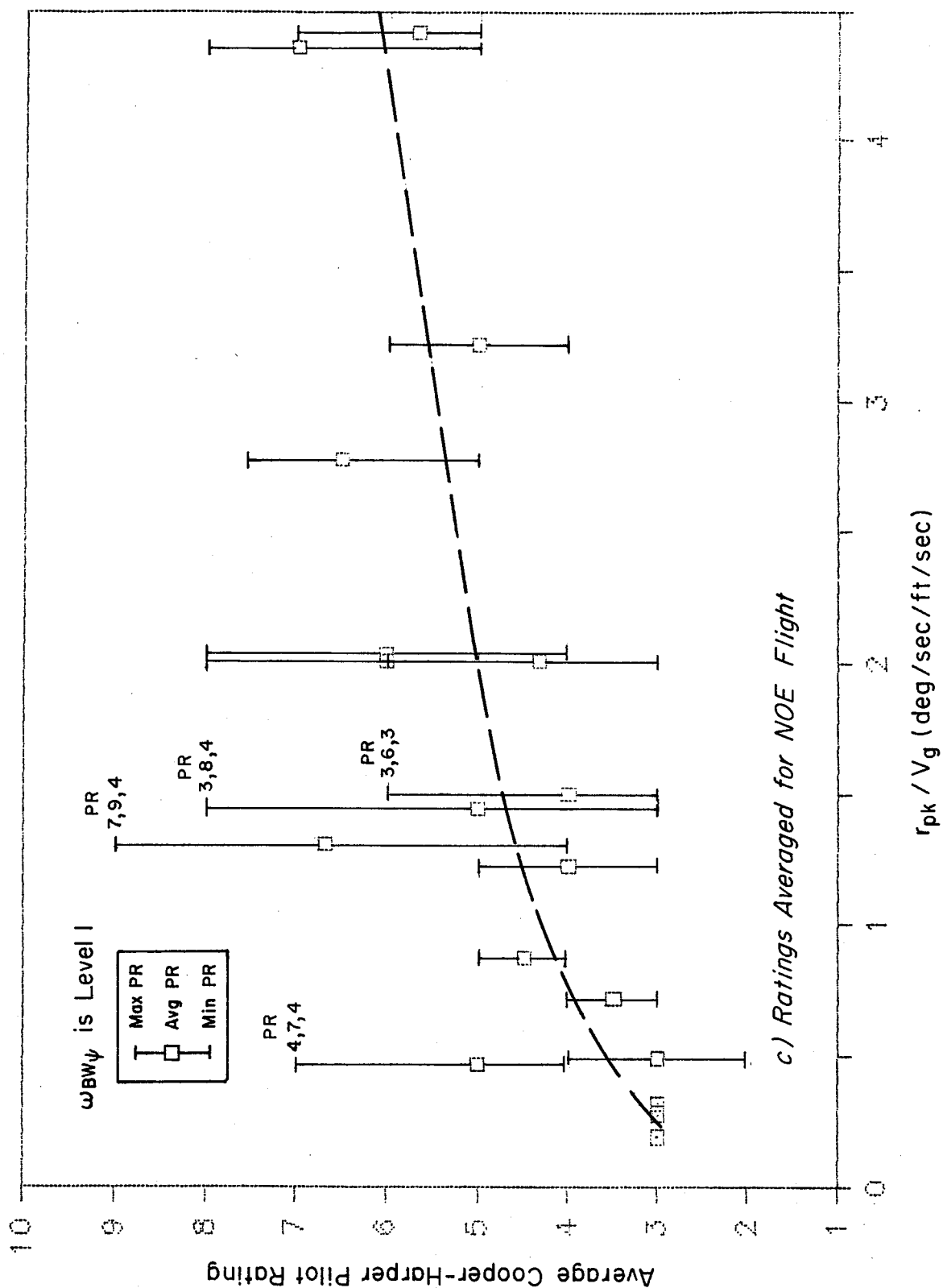


Figure 2(3.3.7.1). (Concluded)

a. Statement of Requirement

3.3.8 Large-Amplitude Heading Changes. The achievable yaw rate in Hover shall be no less than the values specified in Table 1(3.3). The specified angular rates must be achieved about the yaw axis while limiting excursions in the other axes with the appropriate control inputs, and with main rotor RPM at the lower sustained operating limit. Response-Types which are designated as applicable only to UCE = 2 or 3 must meet only the Limited Maneuvering requirements.

b. Rationale for Requirement

For large heading changes, the pilot is not as concerned about the bandwidth of the response as he is about the maximum rate of the response. In the specification, this is addressed for all axes by specifying limits on minimum achievable angular rates.

The yaw rate requirements of Table 1(3.3) are divided into three levels of aggressiveness, based upon the anticipated Mission-Task-Elements. The philosophy behind this division is described in Rationale for Requirement for Paragraph 3.3.4.

c. Supporting Data

The limits on minimum rate for limited maneuvering in Table 1(3.3), 9.5 deg/sec for Level 1 and 5 deg/sec for Level 2, are based on the MIL-F-83300 VSTOL requirements. MIL-F-83300 specifies control power in terms of time to achieve an attitude in 1 second, which has been converted to an equivalent steady angular rate at full controller deflection. Assuming a minimum (conservative in terms of control power) level of damping of 2.5 1/sec,

$$\frac{\dot{\psi}(ss)}{\psi(1)} = \frac{-1}{T \left( 1 - e^{-1/T} - \frac{1}{T} \right)}$$

where  $\frac{1}{T} \Rightarrow N_r$  for conventional helicopters

$$\dot{\psi}(ss) = 1.58 \psi(1)$$

The MIL-F-83300 requirements for heading control power are 6 deg in 1 second for Level 1, and 3 deg in 1 second for Level 2. These requirements convert to steady-state rates of 9.5 deg/sec and 4.7 deg/sec, respectively. These steady-state values are used in Table 1(3.3), rounded to the nearest half-degree-per-second.

The 22 deg/sec yaw rate limit for Level 1 for moderate maneuvering in Table 1(3.3) is based on the ability to do a quickstop and turn into the wind. The required yaw rates for such a maneuver were found in the Reference 85 flight tests, discussed in the Supporting Data for

Paragraph 3.3.4 [see Table 1b(3.3.4)]. This maneuver is an element of an assault landing such as would be required of a utility rotorcraft in combat.

The Level 2 requirement for moderate maneuvering MTEs is taken as the Level 1 requirement for the limited maneuvering MTEs, on the basis that the mission could be continued, albeit with degraded effectiveness.

The minimum yaw rate for aggressive maneuvering for Level 1 [ $\pm 60$  deg/sec in Table 1(3.3)] is based on a Letter of Agreement (LOA) maneuver for a proposed advanced attack/scout helicopter that requires a 180 deg yaw in 5 sec. A key aspect of this maneuver is that it requires the pilot to not only yaw 180 deg, but to stabilize within 2 deg without overshoot. The peak yaw rate required depends on the yaw axis time constant,  $T$  ( $T = 1/N_r$  for conventional unaugmented helicopters) according to the following simple relationship which assumes a first-order response (usually valid for yaw control):

$$\dot{\psi}_{ss} = \frac{\Delta\psi}{\Delta t - 2T} = \frac{180}{5-2} = 60 \text{ deg/sec}$$

where  $\Delta\psi$  is the total heading change, and  $\Delta t$  is the total allotted time to accomplish the maneuver. This assumes a step yaw controller input at both ends of the maneuver, and a value of  $T = 0.5$  sec corresponding to  $\omega_{BW\psi} = 2$  rad/sec as required for Level 1 by Paragraph 3.3.5.1.

The Level 2 limit for the aggressive maneuvering MTEs is simply the Level 1 limit for moderate maneuvering. This would allow sufficient agility for the mission to be continued, albeit with degraded performance.

d. Guidance for Application

None.

e. Related Previous Requirements

MIL-F-83300 (Reference 29), Paragraph 3.2.3.1, specifies control power in terms of attitude change in one second or less. The current approach more directly addresses control power from the pilot's view point: the maximum achievable rate or attitude.

a. Statement of Requirement

3.3.9 Interaxis Coupling. Control inputs to achieve a response in one axis shall not result in objectionable responses in one or more of the other axes. This shall hold for control inputs up to and including those required to achieve the moderate amplitude responses in Paragraphs 3.3.3 and 3.3.6. Specific limits on interaxis coupling are given in Paragraphs 3.3.9.1, 3.3.9.2, and 3.3.9.3.

b. Rationale for Requirement

This paragraph introduces the topic, and sets a qualitative limit on the degree, of allowable interaxis coupling. As noted, the following subtopics are more specific as to the currently recognized quantitative level of "objectionable" responses. Table 1 summarizes the types of coupling that exist for single-rotor helicopters. The intent of this requirement is to cover any coupling not specifically addressed in 3.3.9.1, 3.3.9.2, or 3.3.9.3.

c. Supporting Data

None.

d. Guidance for Application

None.

e. Related Previous Requirements

None.

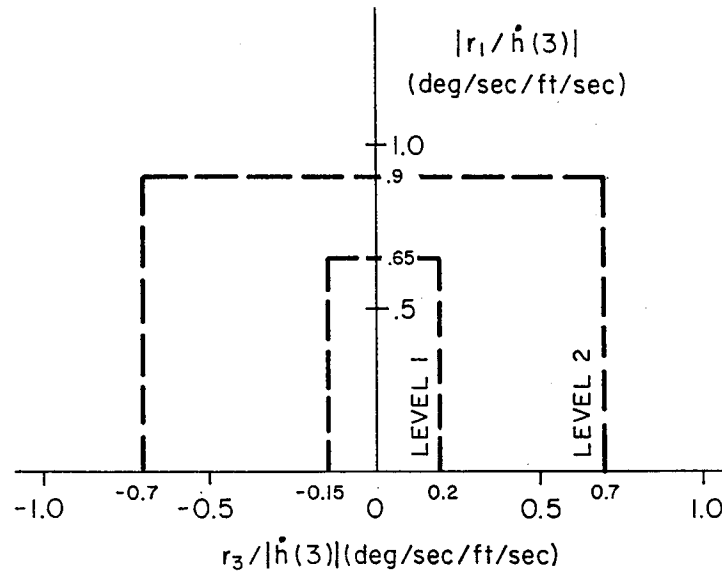
TABLE 1(3.3.9). SINGLE-ROTOR HELICOPTER COUPLING SOURCES (REFERENCE 105)

RESPONSE INPUT	PITCH	ROLL	YAW	CLIMB OR DESCENT
Longitudinal Stick	Pure (Prime)	1) Lateral flapping due to longitudinal stick 2) Lateral flapping due to pitch rate 3) Lateral flapping due to load factor	Negligible	Desired for verti- cal flight path control in forward flight
Lateral Stick	1) Longitudinal flapping due to lateral stick 2) Longitudinal flapping due to roll rate	Pure (Prime)	1) Undesired in hover, caused by directional stability 2) Desired for turn coordination and heading control in forward flight	Descent with bank angle at fixed power
Pedals (Rudder)	Negligible	1) Roll due to tail rotor thrust 2) Roll due to side- slip	Pure (Prime)	Undesired due to power changes in hover
Collective	1) Transient longi- tudinal flapping with load factor 2) Steady longitudinal flapping due to climb and descent in forward flight caused by rotor flapping 3) Pitch due to change in horizontal tail lift	1) Transient lateral flapping with load factor 2) Steady lateral flapping with climb and descent 3) Sideslip induced by power change causes roll due to dihedral	Power change varies requirement for tail rotor thrust	Pure (Prime)



a. Statement of Requirement

3.3.9.1 Yaw Due to Collective. The yaw rate response to abrupt collective inputs with the directional controller free shall not exceed the boundaries specified in Figure 7(3.3). In addition, there shall be no objectionable yaw oscillations following step or ramp collective changes in the positive and negative directions. Oscillations involving yaw rates greater than 5 deg/sec shall be deemed objectionable.



where:

$r_1$  = first peak (before 3 seconds) or  $r(1)$  if no peak occurs before 3 seconds

$$r_3 = \begin{cases} (r(3) - r_1) & \text{for } r_1 > 0 \\ (r_1 - r(3)) & \text{for } r_1 < 0 \end{cases}$$

$r(1)$  and  $r(3)$  are yaw rate responses measured at 1 and 3 seconds, respectively, and  $h(3)$  is altitude rate response measured at 3 seconds, following a step collective input at  $t = 0$

In the unlikely event of more than one peak before 3 seconds, the largest peak (by magnitude) should be designated as  $r_1$ .

Figure 7(3.3). Collective-to-Yaw Coupling Requirements

b. Rationale for Requirement

Collective-to-yaw coupling is a fundamental characteristic of all single rotor helicopters. The rotorcraft Mission-Task-Elements (Paragraph 3.2.2) include some collective-intensive tasks such as the pull-up/push-over, rapid accel/decel, and bob-up/down maneuvers. A heavy directional-control workload due to collective applications would adversely affect the pilot's ability to complete these tasks, and heading excursions due to collective applications should therefore be minimized.

The present limits are based on data where the yaw axis Response-Type was Rate with no heading hold. The requirement is valid for a helicopter with a conventional side-mounted collective stick and pedal-type directional controller. Further investigation is necessary to determine the effect of 3- or 4-axis sidestick controllers on these boundaries.

c. Supporting Data

The bulk of the supporting data for Figure 7(3.3) was obtained from an experiment conducted by the Army Aeroflightdynamics Directorate (AFD) and Systems Technology, Inc., on the Vertical Motion Simulator at NASA Ames Research Center. Results of this experiment have not been published elsewhere. The baseline helicopter model was a UH-60A Blackhawk. Turbulence was not included in the experiment. Different levels of collective-to-yaw crosscoupling were obtained by inserting collective-to-pedal crossfeeds which varied the magnitude of coupling as a function of frequency. The experiment was configured to investigate exclusively the effect of collective-to-yaw coupling on two collective-intensive tasks: an NOE course with two berms, and a hover bob-up/bob-down sequence at the end of the course. The pilots were instructed to fly the NOE course both as aggressively as possible, maintaining an average speed of 40 kts, and more moderately at a speed of approximately 20 kts. Pilot ratings were obtained for the aggressive and nonaggressive NOE tasks, and for the bob-up/bob-down. Collective-to-yaw coupling was not a factor for the nonaggressive tasks (i.e., all the ratings were Level 1). Problems due to excessive coupling were exposed dramatically during the aggressive tasks.

This criterion is a measure of not only the magnitude, but also the shape of the yaw rate response to a step collective stick input. The importance of shape can be seen by examining the responses evaluated in the simulation, presented in Figure 1. Included in this figure are the average pilot ratings for the bob-up task. Cases with a reversal in yaw acceleration received poorer ratings than others with similar peak yaw rates but no reversal in yaw acceleration. The parameters defined in Figure 7(3.3) account for these characteristics by measuring the peak yaw rate and the value of yaw rate after 3 seconds.

The average pilot ratings for the aggressive NOE and bob-up tasks are shown plotted against the proposed criterion parameters in Figure 2. Additional data from a directional control experiment (Reference 106)

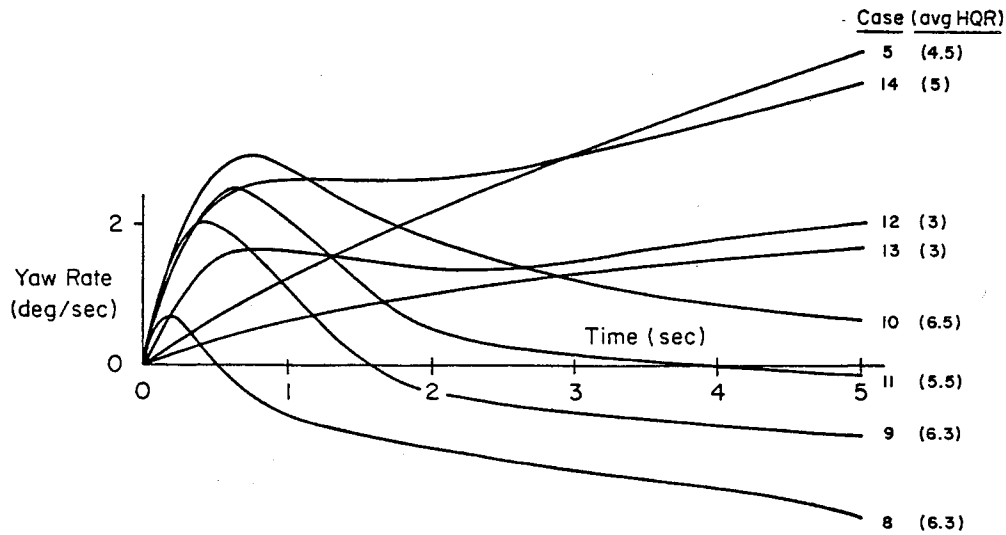


Figure 1(3.3.9.1). Yaw Rate Responses to Step Collective Input at  $t=0$  for Evaluation Cases from VMS Simulation (Hover)

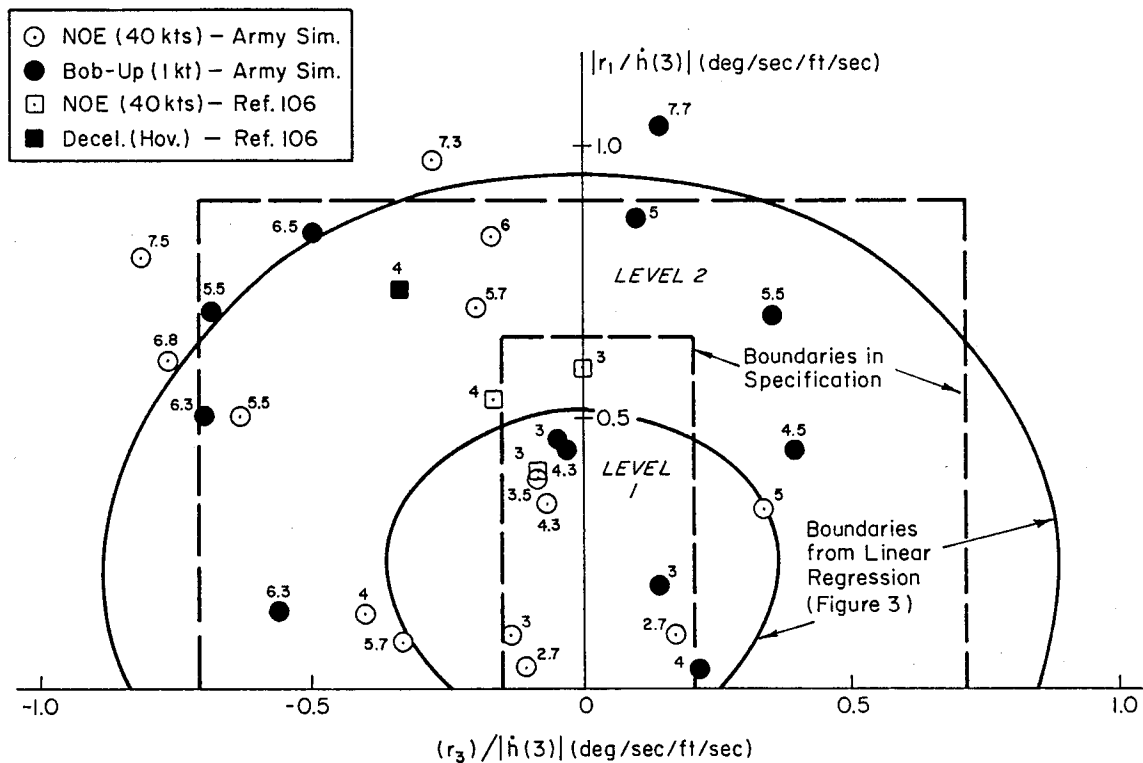


Figure 2(3.3.9.1). Comparison of Pilot Ratings from Simulator with Collective-to-Yaw Coupling Performance Boundaries

have been included in Figure 2. Since the experiment described in Reference 106 was not intended to specifically investigate collective-to-yaw coupling, only those ratings for which the pilots specifically commented on collective-to-yaw coupling have been included.

In order to further investigate the influence of each of the criterion parameters on pilot rating, a multiple linear regression fit to the data was performed using an elliptical equation (Figure 3). This resulted in the curved boundaries shown in Figure 2. An elliptical equation was used since the spread of the data seemed to indicate the existence of elliptical boundaries. Moreover, an alternate linear equation would result in a "pyramid" shaped boundary which, for data in the proximity of its apex, would show an unrealistic sensitivity to variations in  $r_3/|\dot{h}(3)|$ . The curved boundaries of Figure 2 result in regions (particularly for Level 1) with no supporting data. The square boundaries represent a more conservative criterion to reflect the existing data.

In addition to meeting the boundaries of Figure 7(3.3), it is also required that there be no objectionable yaw oscillations following a step or ramp collective change. This part of the requirement was motivated by experience gained during the development of the AH-64 attack helicopter where a yaw oscillation that occurred during quick-stops was deemed to be objectionable by the pilots. This heading oscillation was induced by oscillations in the engine-rotor RPM governor

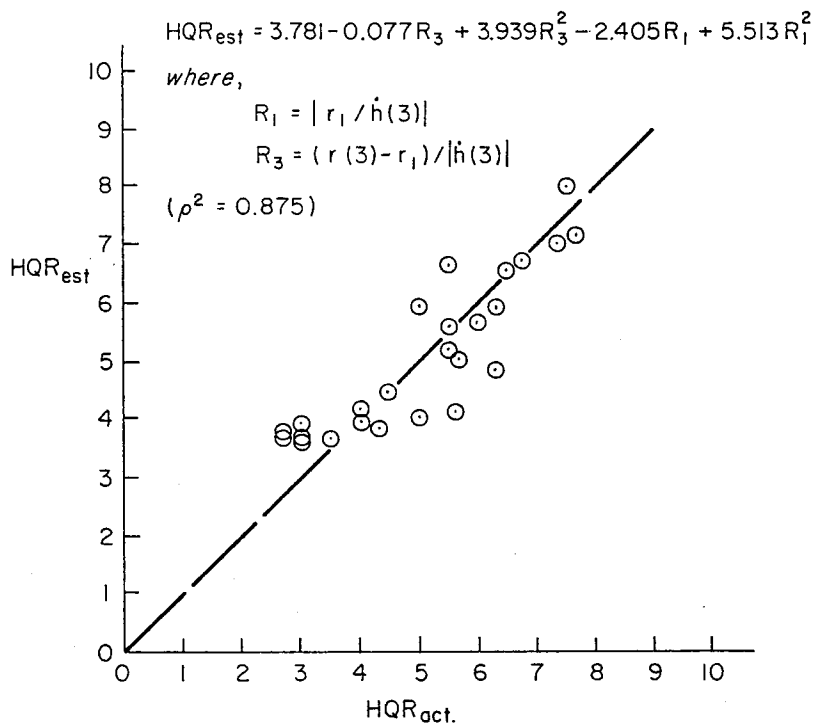


Figure 3(3.3.9.1). Linear Regression Fit to Data  
From Coupling Experiment

as shown in Figure 4. The lack of handling qualities data based on systematic variations in the amplitude and frequency of governor-induced oscillations required the use of "no objectionable" as a pass-fail measure. This is augmented by specifying that yaw rates of greater than 5 deg/sec would be deemed as objectionable. Five deg/sec was picked on the basis that it is half of the peak yaw rate in Figure 4 -- an oscillation that was definitely judged as objectionable, and worthy of a control system modification. That modification consisted of feedforward collective shaping, as well as an increase in bandwidth of the heading SAS from 2.5 to 4 rad/sec.

d. Guidance for Application

None.

e. Related Previous Requirements

None.

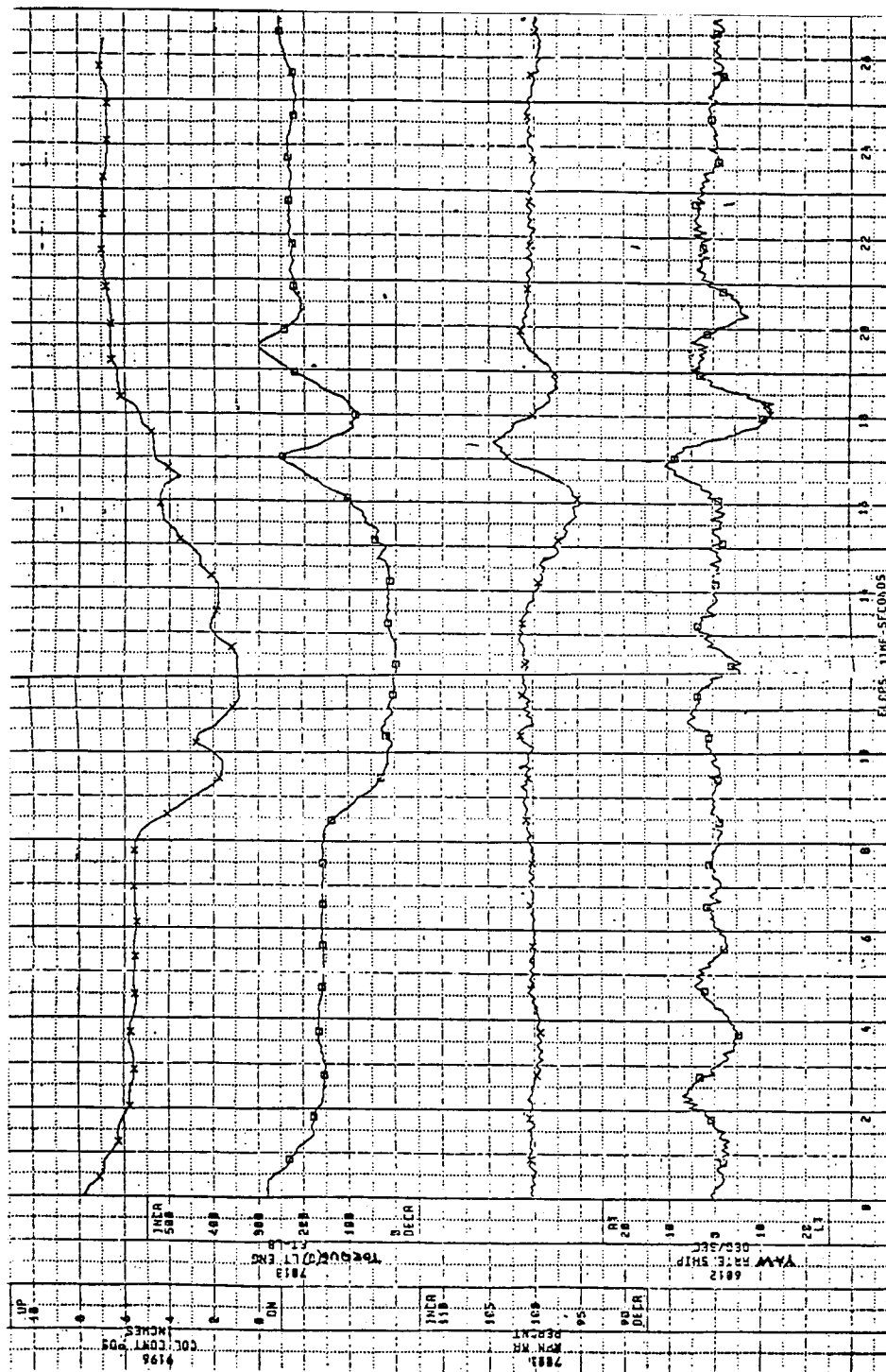


Figure 4(3.3.9.1). Vertical-Directional-Engine Coupling During Quickstop  
(Unpublished; Provided by Mr. R. Prouty of McDonnell-Douglas Helicopters)

a. Statement of Requirement

3.3.9.2 Pitch-to-Roll and Roll-to-Pitch Coupling During Aggressive Maneuvering. The following requirement applies for the Aggressive Maneuvering Mission-Task-Elements in Table 1(3.3). The ratio of peak off-axis response to desired response,  $\theta_{pk}/\phi$  ( $\phi_{pk}/\theta$ ), following an abrupt lateral (longitudinal) cyclic step input, shall not exceed the limits specified in Table 3(3.3) for at least 4 seconds after the input is initiated. Heading excursions shall be limited by use of the heading controller. This shall hold for control inputs up to and including those required to achieve the moderate-amplitude responses in Paragraph 3.3.3.

TABLE 3(3.3). MAXIMUM VALUES FOR PITCH-TO-ROLL AND ROLL-TO-PITCH COUPLING

PARAMETER	LEVEL 1	LEVEL 2
$(\theta_{pk}/\phi) \delta_A$ or $(\phi_{pk}/\theta) \delta_B$	$\pm 0.25$	$\pm 0.60$

b. Rationale for Requirement

Cross-coupling between pitch and roll is common, and in fact is almost unavoidable, for single-rotor helicopters. Reference 163, for example, discusses the level of coupling encountered on three helicopters (the YUH-61A, BO-105C, and CH-47). It is possible for the ratio of unwanted response (e.g., pitch attitude for a lateral cyclic input) to desired response to have values in excess of unity, especially for hingeless-rotor designs (Figure 1). In terms of handling qualities, there is an upper limit on the magnitude of such coupling, beyond which the pilot's ability to perform the specified tasks is compromised.

Numerous flight and simulation studies (e.g., References 83, 87, 108, 161, 162, 172, and 196) have investigated the effects of pitch/roll coupling for both low-speed and forward-flight tasks. The primary observation from these studies is that the impact of cross-coupling on handling qualities is highly task-dependent: for the more benign tasks, such as ILS approaches and hovering, very high levels of coupling are not objectionable. For aggressive maneuvering, however, such as roll reversals, vertical landings, dolphins and slaloms, relatively low levels of coupling result in degraded handling qualities.

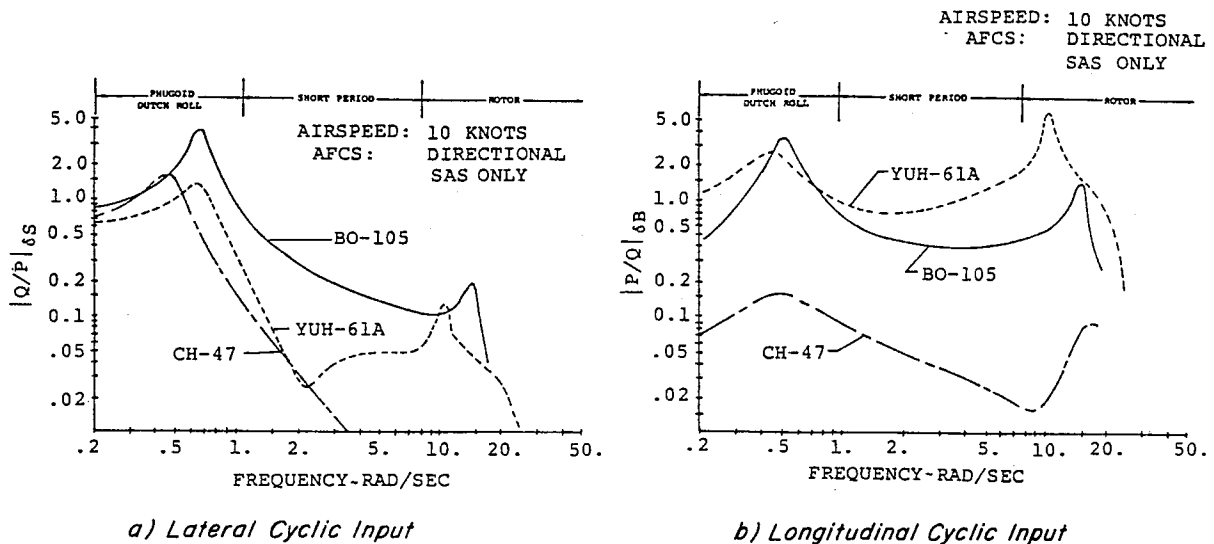


Figure 1(3.3.9.2). Coupling Response Ratios (Unwanted Response/Desired Response) for Three Helicopters (from Reference 163)

Because of the task dependency, this paragraph places limits on the allowable pitch-to-roll and roll-to-pitch coupling for the more aggressive MTEs corresponding to the "Aggressive Maneuvering" category in Table 1(3.3). For all other maneuvers, the more general wording of Paragraph 3.3.9 applies: "Control inputs to achieve a response in one axis shall not result in objectionable responses in one or more of the other axes."

There is an obvious frequency-varying nature to coupling, as illustrated in Figure 1. Ideally, limits on coupling would be functions of frequency as well, since the pilot is likely to be more tolerant of coupling at some frequencies (e.g., a high level of coupling at very low frequencies may not interfere with aggressive maneuvering, but might be undesirable for a more benign task such as instrument approach). There is very little data for setting such limits, however, and it is felt that this paragraph, which is intended specifically for aggressive maneuvering MTEs, is appropriate in the short term only. A time limit of five seconds was chosen to reflect this.

#### c. Supporting Data

Supporting data for this paragraph come from References 108, 161, 162, 172, and 196. Before reviewing these reports, it will be helpful to look at some of the fundamental concepts governing pitch/roll coupling.



## 1. Sources of Coupling

There are two primary sources of pitch/roll coupling for single-rotor helicopters: out-of-axis moments resulting from gyroscopic coupling, and out-of-axis moments due to cyclic stick inputs. The details of the physical characteristics that cause these moments are not of interest here (Reference 83 contains a discussion of these characteristics); instead, we wish to understand their effect on the dynamics of the helicopter.

For simplicity, consider the helicopter to be a single rotating disk; then both pitch rate and roll rate damping are provided entirely by the in-axis gyroscopic damping moment,  $M$  (in units of ft-lb/rad/sec). This gyroscopic moment, divided by the moment of inertia, corresponds to the classical rate damping derivative in units of rad/sec; since helicopters typically have higher moments of inertia in pitch than in roll, pitch damping will be smaller than roll damping. We define

$$M_q = -M/I_y \text{ and}$$

$$L_p = -M/I_x$$

Hence for most single-rotor helicopters,  $M_q/L_p = I_x/I_y$ .

Gyroscopic coupling is produced by an out-of-axis moment,  $H$  (units of ft-lb/rad/sec), and the coupling derivatives are

$$M_p = -H/I_y \text{ and}$$

$$L_q = H/I_x$$

Thus, if we ratio the coupling and damping derivatives,

$$M_p/M_q = H/M \text{ and}$$

$$L_q/L_p = -H/M, \text{ or}$$

$$M_p/M_q = -L_q/L_p$$

Other contributions, such as tail rotor or horizontal tail effects, are quite minor for most helicopters; for multi-rotor helicopters, such as the XV-15 and the CH-47, these relationships do not hold at all.

The second form of coupling, due to out-of-axis control responses, is influenced by such configurational details as canted tail rotors or presence of a horizontal tail. The contribution of pitching moment due to lateral cyclic is designated  $M_{\delta_A}$ , and of rolling moment due to longitudinal cyclic as  $L_{\delta_B}$ , both with dimensions of rad/sec/(unit of control deflection).

The equations of motion for pitch and roll response, ignoring all other axes (and assuming that the product of inertia,  $I_{xz}$ , is zero), are:

$$\begin{aligned}\dot{p} - L_p p - L_q q &= L_{\delta_A} \delta_A + L_{\delta_B} \delta_B \\ \dot{q} - M_q q - M_p p &= M_{\delta_B} \delta_B + M_{\delta_A} \delta_A\end{aligned}\tag{1}$$

Written in terms of Laplace transforms, these equations, in matrix form, are:

$$\begin{bmatrix} s - L_p & -L_q \\ -M_p & s - M_q \end{bmatrix} \begin{bmatrix} p \\ q \end{bmatrix} = \begin{bmatrix} L_{\delta_A} & L_{\delta_B} \\ M_{\delta_A} & M_{\delta_B} \end{bmatrix} \begin{bmatrix} \delta_A \\ \delta_B \end{bmatrix}\tag{2}$$

The characteristic equation for this system is the determinant of the left-hand matrix,

$$\Delta = (s - L_p)(s - M_q) - L_q M_p = s^2 - (L_p + M_q)s + L_p M_q - L_q M_p\tag{3}$$

[Note that if there is no coupling, this reduces to  $(s - L_p)(s - M_q)$ ]. The transfer functions for the roll attitude and pitch attitude responses to a lateral cyclic input can be derived as:

$$\begin{aligned}\frac{\phi}{\delta_A} &= \frac{1}{s} \frac{p}{\delta_A} = \frac{L_{\delta_A}(s - M_q + L_q M_{\delta_A}/L_{\delta_A})}{s\Delta} \\ \frac{\theta}{\delta_A} &= \frac{1}{s} \frac{q}{\delta_A} = \frac{M_{\delta_A}(s - L_p + M_p L_{\delta_A}/M_{\delta_A})}{s\Delta}\end{aligned}\tag{4}$$

These equations describe the effect of the gyroscopic and control coupling derivatives on the helicopter's pitch and roll responses for a lateral cyclic input. The latter equation, especially, is of interest since it corresponds to the unwanted response. The magnitude of this unwanted response is relevant only if evaluated in relation to the

magnitude of the desired response -- i.e., the ratio of  $\theta/\phi$  due to lateral cyclic. This is given by:

$$\left(\frac{\theta}{\phi}\right)_{\delta_A} = \frac{M_{\delta_A} \left( s - L_p + M_p L_{\delta_A} / M_{\delta_A} \right)}{L_{\delta_A} \left( s - M_q + L_q M_{\delta_A} / L_{\delta_A} \right)} \quad (5)$$

The value of the term  $(-L_p + M_p L_{\delta_A} / M_{\delta_A})$  is almost always larger than  $(-M_q + L_q M_{\delta_A} / L_{\delta_A})$ , so the maximum value of this ratio is at the steady state:

$$\left(\frac{\theta_{pk}}{\phi}\right)_{\delta_A} = \left(\frac{\theta}{\phi}\right)_{\delta_A} (s.s.) = \frac{L_{\delta_A} M_p - M_{\delta_A} L_p}{M_{\delta_A} L_q - L_{\delta_A} M_q} \quad (6)$$

In addition, except for extremely high levels of coupling, the term  $|L_{\delta_A} M_q| \gg |M_{\delta_A} L_q|$ , so

$$\left(\frac{\theta_{pk}}{\phi}\right)_{\delta_A} \approx \frac{M_{\delta_A} L_p}{L_{\delta_A} M_q} - \frac{M_p}{M_q} \quad (7)$$

If the control coupling is negligible, this reduces to  $-M_p/M_q$ , which is simply the ratio of damping moment to coupling moment,  $H/M$ , due to the main rotor.

This procedure can be followed to derive the equations describing the responses to longitudinal cyclic, for which

$$\left(\frac{\phi_{pk}}{\theta}\right)_{\delta_B} = \left(\frac{\phi}{\theta}\right)_{\delta_B} (s.s.) = \frac{M_{\delta_B} L_q - L_{\delta_B} M_q}{L_{\delta_B} M_p - M_{\delta_B} L_p} = \frac{L_{\delta_B} M_q}{M_{\delta_B} L_p} - \frac{L_q}{L_p} \quad (8)$$

Again, if there is negligible control coupling, this reduces to  $-L_q/L_p = H/M$ .

For most of the references used as support for this paragraph and the forward-flight counterpart, gyroscopic coupling is described in terms of either  $M_p/M_q$  ( $= -L_q/L_p$ ) or  $H/M$ . As Equations 6 and 8 indicate, these ratios account for the rate coupling due to gyroscopic moments, but they do not account for the total coupling, including control cross-coupling.

In developing requirements for this paragraph, the derivative ratios were considered and abandoned, recognizing that they are both difficult to estimate and incomplete in describing the actual level of pitch/roll coupling. Instead, the parameters chosen are intended to account for all forms of coupling, and are amenable to measurement from flight data. These parameters are also still quite closely related to the derivative ratios. Analysis of the supporting data involved calculating the parameters required using whatever information was

available: either the fully coupled responses if all stability and control derivatives were provided, actual ratios if time histories were provided, or, as a last resort, by using the approximations given by Equations 6 and 8.

## 2. Analysis of the Data

In the flight experiment of Reference 161, the effects of gyroscopic coupling were evaluated by varying the coupling moment  $H$  using a variable-stability helicopter. The basic damping and control response characteristics of this helicopter were marginal to begin with (corroborated by comments in Reference 108), and this probably had some effects on the pilots' opinions. No quantitative data (including handling qualities ratings) are given; instead, an assessment of the pilots' evaluations for various tasks is provided. Variations in roll-due-to-pitch coupling, corresponding to the ratio  $L_q/L_p$ , from 0 to 0.84 were evaluated, though the pilots considered this form of coupling to be relatively unimportant. The highest level of pitch-due-to-roll coupling evaluated varied with task. All tasks were either at hover ("holding the aircraft absolutely motionless") or at a speed of 30-35 kts. The results are summarized as:

- Hovering: Pitch-due-to-roll coupling was varied from 0 to 3.9 in combination with the roll-due-to-pitch coupling given above; Reference 161 reports that "even at maximum level of coupling, the pilot reported that no coupled response was apparent." In the absence of external disturbance (assuming the tests were conducted on calm days), the hovering task probably did not require large or aggressive inputs.
- Low-speed ILS approaches: The pitch-due-to-roll was varied from 0 to 1.7 with no objections by the pilots. Again, this maneuver probably did not require large or rapid inputs.
- Steep turns and roll reversals: For these maneuvers,  $M_p/M_q$  varied between 0 and 1.24. Qualitative assessments of controllability in the presence of cross-coupling were given by three pilots, as follows:  $M_p/M_q = 0.31$  -- Acceptable; 0.62 -- Marginal; 0.93 -- Poor; 1.24 -- Unacceptable.

The pilots considered pitch-due-to-roll coupling to be much more severe than roll-due-to-pitch, reflecting the fact that all of the maneuvers (with the exception of hovering) were primarily lateral in nature. Had the tasks included primarily longitudinal maneuvers -- such as a dash/quickstop or dolphin -- the roll coupling would probably have been exposed.

The results of the Reference 161 experiment support the separation of coupling requirements into nonaggressive and aggressive MTEs. In

addition, the qualitative assessments in the roll-reversal evaluations suggest that controllability is compromised when the ratio  $M_p/M_q$  has a value between 0.31 and 0.62.

The flight experiment of Reference 108 investigated the effects of coupling due to the control cross-coupling derivatives,  $M_{\delta_A}$  and  $L_{\delta_B}$ . The variable-stability helicopter used in the Reference 161 experiment was used here as well, with the basic level of pitch and roll rate damping and control power (producing a response that was, according to Reference 108, "marginally satisfactory to unsatisfactory"), and with control power and damping at three times the basic level. Control cross-coupling was produced by reorienting the cyclic controller in the cockpit at different angles from zero to 55 degrees. A lateral cyclic stick input would thus produce a pure roll response at a control orientation (phase angle) of zero degrees and a pure pitch response at 90 degrees. Hence the effective lateral control derivative,  $L_{\delta_A}$ , would decrease from its nominal value to zero as phase angle is increased, and the coupling derivative,  $M_{\delta_A}$ , would increase from zero to a value equal to  $M_{\delta_B}$  at 90 degrees. The control and coupling derivatives are given by the following expressions:

$$\begin{aligned} L_{\delta_A}(\text{effective}) &= L_{\delta_A} \cos \epsilon \\ M_{\delta_B}(\text{effective}) &= M_{\delta_B} \cos \epsilon \\ L_{\delta_B} &= L_{\delta_A} \sin \epsilon \\ M_{\delta_A} &= M_{\delta_B} \sin \epsilon \\ \epsilon &= \text{Control Phase Angle} \end{aligned} \quad (9)$$

Assuming all coupling is due to the control phase angle, the coupling parameters required for this paragraph are given by:

$$\begin{aligned} \left( \frac{\theta_{pk}}{\phi} \right)_{\delta_A} &= \frac{M_{\delta_A}}{L_{\delta_A}(\text{effective})} = \frac{M_{\delta_B} \sin \epsilon}{L_{\delta_A} \cos \epsilon} = \frac{M_{\delta_B}}{L_{\delta_A}} \tan \epsilon \\ \left( \frac{\phi_{pk}}{\theta} \right)_{\delta_B} &= \frac{L_{\delta_B}}{M_{\delta_B}(\text{effective})} = \frac{L_{\delta_A} \sin \epsilon}{M_{\delta_B} \cos \epsilon} = \frac{L_{\delta_A}}{M_{\delta_B}} \tan \epsilon \end{aligned} \quad (10)$$

The tasks flown in the Reference 108 experiment consisted of low-speed ILS approaches, circling patterns (takeoff, hovering, transition to and from forward flight, and vertical landing), square patterns at constant heading, and roll reversals (at 45 kts). As with the Reference 161 tasks, these are primarily lateral in nature. Handling qualities ratings (using the Cooper pilot rating scale) are shown in Figure 2 as a function of pitch-due-to-roll coupling. All ratings are in the Level 2

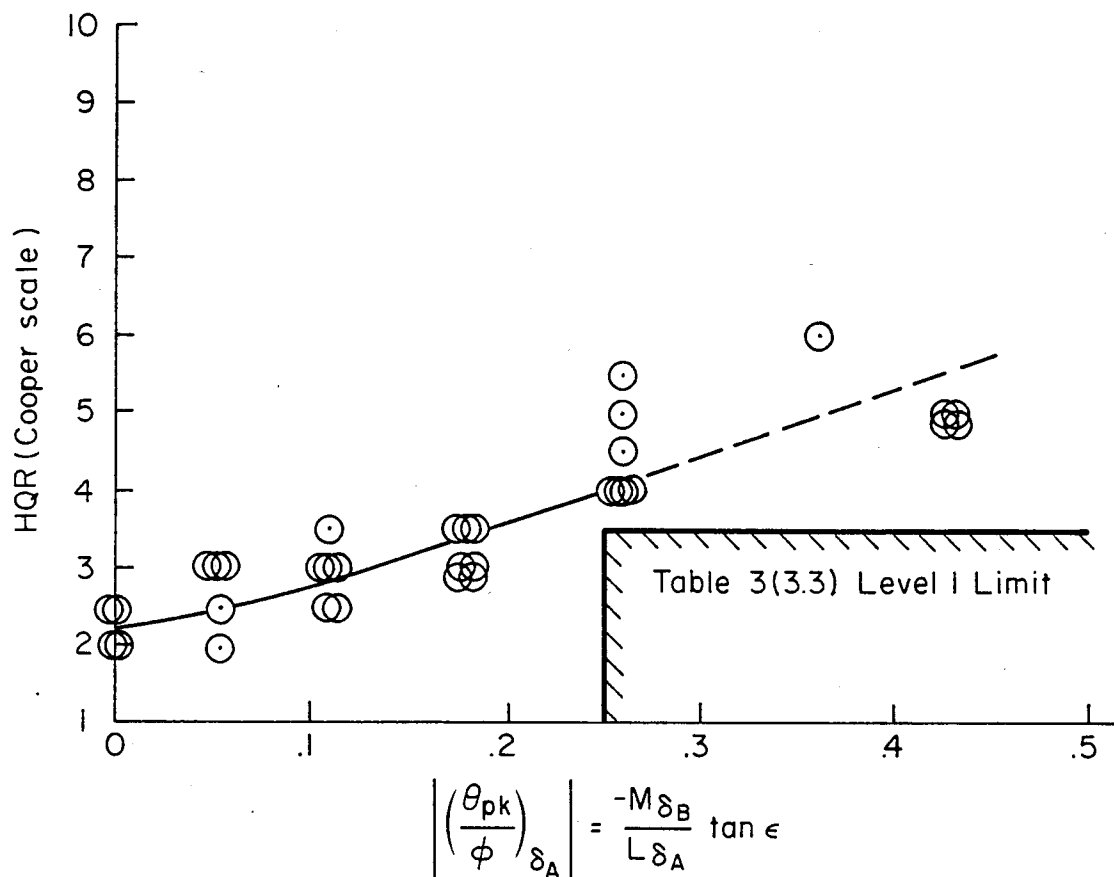
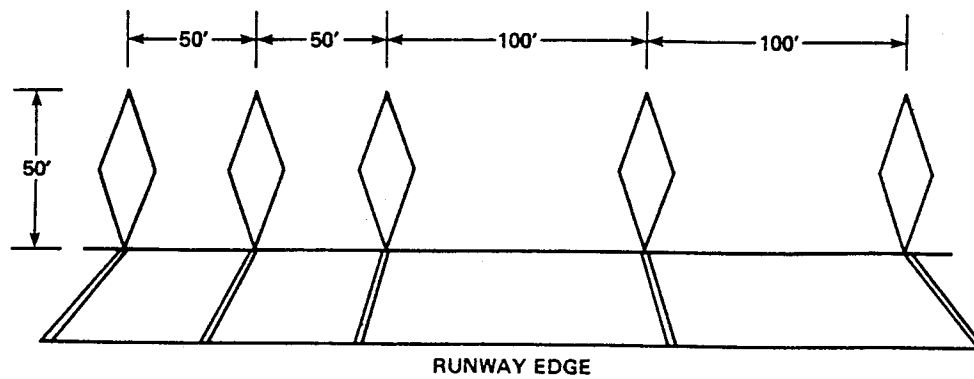


Figure 2(3.3.9.2). Handling Qualities Ratings from Low-Speed Flight Experiment of Reference 108

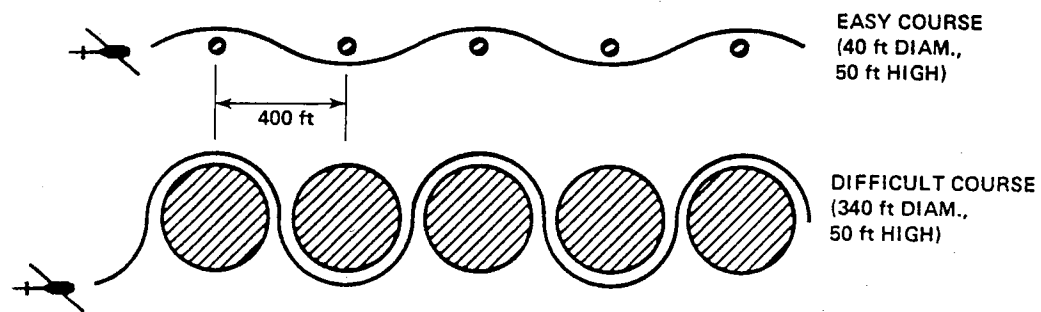
region for coupling values above 0.26, supporting a Level 1 limit at a value below 0.26. There are not enough data to determine a Level 2 limit.

The simulation of Reference 172 investigated a wide range of pitch-to-roll and roll-to-pitch coupling values. Helicopter models representative of hingeless and articulated rotor configurations were modified through the rotor parameters to exhibit a range of both control-type coupling and angular-rate-damping coupling. Tasks (Figure 3) consisted of 100-ft lateral sidesteps from a hover, stabilizing at each step, and 30-kt constant-speed slaloms around cylindrical obstacles.

For the sidesteps, task aggressiveness was varied by changing the time increment required between sidesteps. This was accomplished by sounding a tone at either 8- or 10-sec intervals, and the task required the pilots to stabilize before the next tone sounded. Reference 172 reports that there was little difference in the results for these two times, so the sidestep data include both the 8- and 10-sec tasks.



*a) The 100ft Lateral Sidestep Course, Three Right, Three Left*



*b) Overhead View of Slalom Courses*

Figure 3(3.3.9.2). Low Speed Tasks Evaluated in VMS Simulation of Reference 172

Two slalom courses were flown, as Figure 3b indicates: an easy course around 40-ft diameter pylons, and a more difficult course around 340-ft diameter tanks. A line was marked on the ground for both courses, indicating the desired ground track, and the pilots were required to keep this line in the forward window for desired performance.

Variation of the dynamics of the two helicopter models resulted in a corresponding variation in control power and bandwidth. For the cases evaluated, however, these effects were secondary to the effects of coupling, and most of the cases met the Level 1 bandwidth requirements of the specification. Pitch bandwidths varied between approximately 0.9-2 rad/sec, and roll bandwidths between 2-4 rad/sec for all cases. The bandwidths of the hingeless-rotor configurations were about twice those of the articulated-rotor cases.

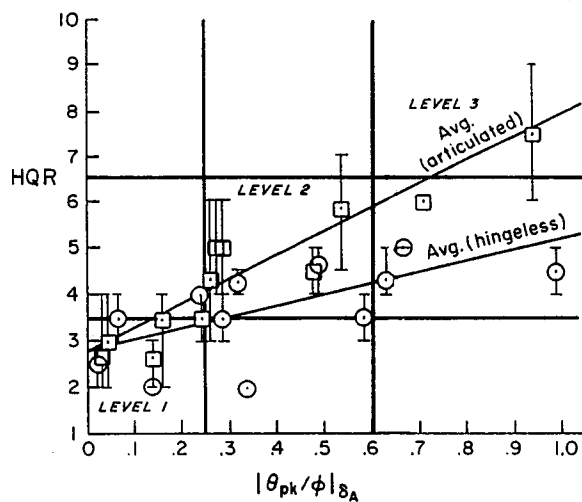
Handling qualities ratings are summarized in Figure 4. For the 30-kt slalom tasks, Figures 4a and 4b, there is a difference in the trend of pilot rating with coupling for the two rotor types. This is probably due to the different basic vehicle response bandwidths; the articulated-rotor configurations, with lower bandwidths in pitch and roll, were probably more sensitive to problems with cross-coupling. As Reference 172 points out, this result suggests that the criterion on cross-coupling should directly address basic vehicle bandwidth as well. There is, however, too little data to determine how this should be done, and the data presented here, along with those in the supporting data for the forward-flight requirement, indicate that the current limits are adequate until more research is performed.

For the easy slalom, Figure 4a, there is very little variation in handling qualities rating with coupling for the hingeless-rotor configuration. The results for the articulated-rotor case are somewhat stronger. By contrast, the difficult slalom revealed strong effects of coupling for both rotor types, Figure 4b. The variation in rating with coupling is similar for the two rotor types as well. It is important to note, however, that the task difficulty for this slalom precluded Level 1 handling qualities for even the low-coupling cases. If the average-rating lines in Figure 4b were shifted downward to start at some reasonable rating (for example, an HQR of 2 with zero coupling), the articulated-rotor results would again support the current limits, while the hingeless-rotor line would indicate less sensitivity to coupling.

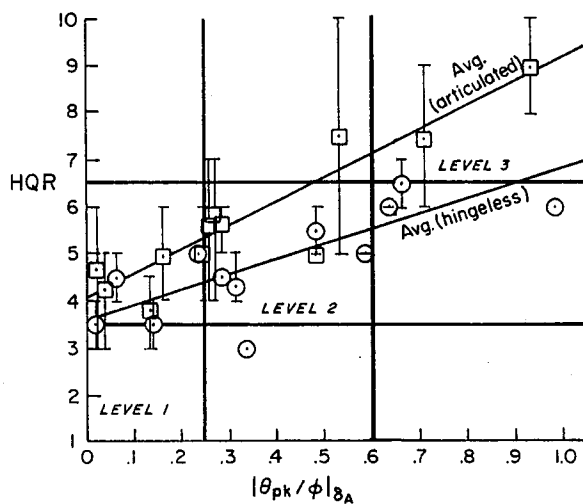
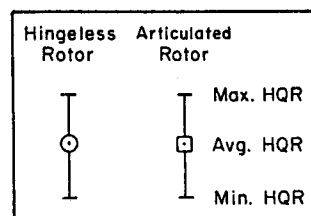
For the hovering sidestep task, Figure 4c, the data for the two rotor types overlay. Again, neither helicopter model was considered to be Level 1 for this task even with very low coupling. The trends, however, do not conflict with the current limits, though these data suggest that the Level 2 limit may be somewhat stringent.

The Army/NASA CH-47 variable stability helicopter was used to repeat a portion of the Reference 172 VMS simulation in actual flight conditions (Reference 196). Some slight differences in dynamics resulted due to the inertial characteristics of the CH-47, and only the sidestep maneuver was performed. The average HQRs show strong agreement with the Figure 4c data, as Figure 5 shows.

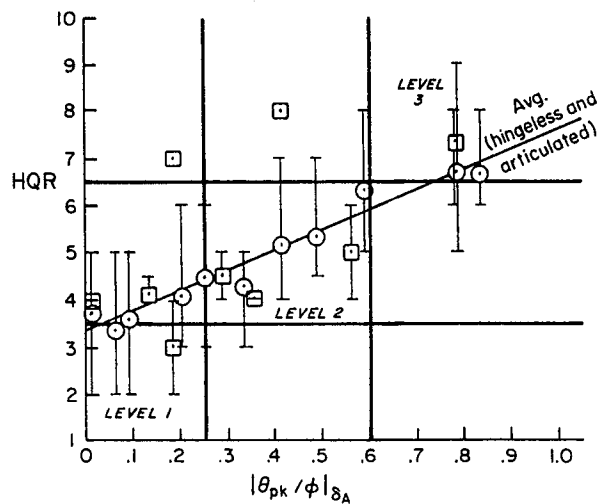




a) Easy Slalom

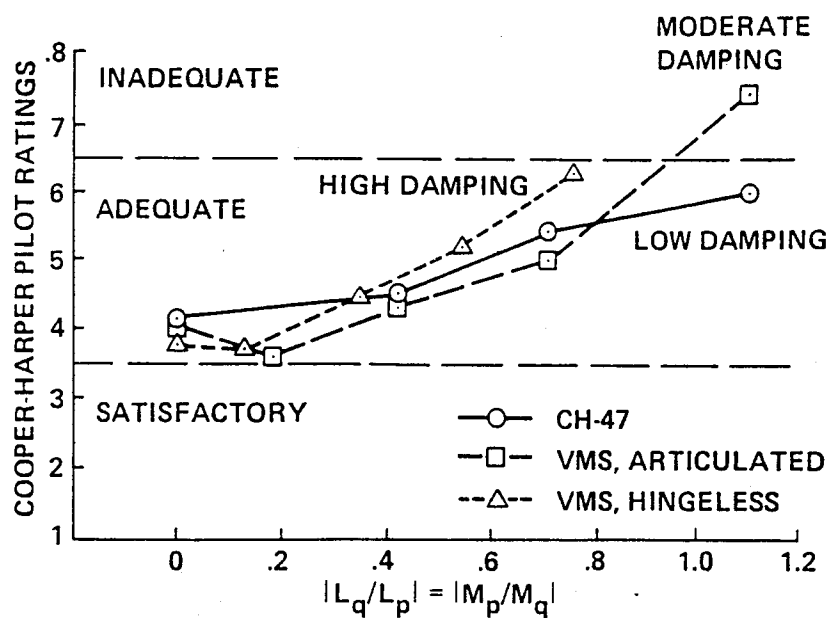


b) Difficult Slalom

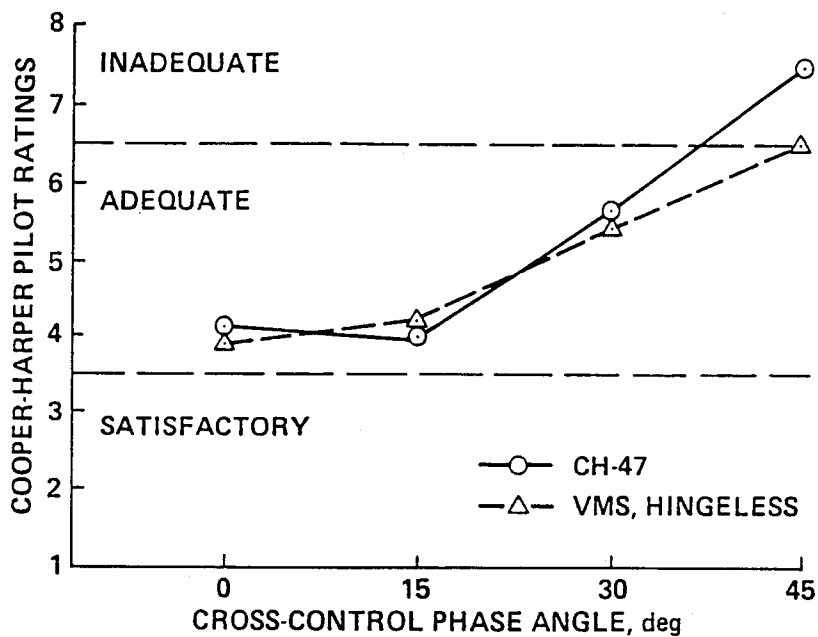


c) Sidestep

Figure 4(3.3.9.2). Handling Quality Ratings from Simulation of Reference 172



a) Angular-Rate Coupling



b) Control Coupling

Figure 5(3.3.9.2). Comparison of Average Handling Quality Ratings from CH-47 Flight Tests (Reference 196) and VMS Simulation (Reference 172) for Sidestep Task

d. Guidance for Application

The requirements of this Paragraph are written in a form intended to be amenable to flight testing. Cyclic inputs are to be applied to the helicopter throughout the low-speed flight regime, and the pitch and roll attitude responses obtained. The most straightforward method for analysis is to plot the response ratio ( $\phi/\theta$  for a longitudinal cyclic input,  $\theta/\phi$  for a lateral cyclic input) for five seconds and simply measure the peak value of the ratio.

e. Related Previous Requirements

There are no requirements in the V/STOL or helicopter specifications concerning coupling. The limits adopted here are close to those recommended by other investigators (e.g., References 83 and 87) for the derivative ratios  $L_q/L_p$  and  $M_p/M_q$ . The data from these references are reviewed in the supporting data for the forward-flight requirement.

a. Statement of Requirement

3.3.9.3 Height Due to Yaw Control. The height response due to yaw control inputs shall not be objectionable.

b. Rationale for Requirement

This requirement was included based on review comments received on the draft version of the specification. These comments indicated that there have been rotorcraft which had an unacceptable sink-rate due to pedal inputs in hover.

c. Supporting Data

None are available, hence the requirement is subjective.

d. Guidance for Application

Since there are no supporting data, this requirement must be complied with via subjective testing. This testing should be conducted in hover, and should consist of rapid pedal inputs consistent with the proposed mission tasks in Paragraph 3.1.1.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.10 Response to Collective Controller

3.3.10.1 Height Response Characteristics. The vertical rate response shall have a qualitative first-order appearance for at least 5 seconds following a step collective input. The limits on the parameters defined by the following equivalent first-order vertical-rate-to-collective transfer function are given in Table 4(3.3).

$$\frac{\dot{h}}{\delta_c} = \frac{K e^{-\tau_{h_{eq}}^* s}}{T_{h_{eq}}^* s + 1}$$

TABLE 4(3.3). MAXIMUM VALUES FOR HEIGHT RESPONSE TO COLLECTIVE CONTROLLER

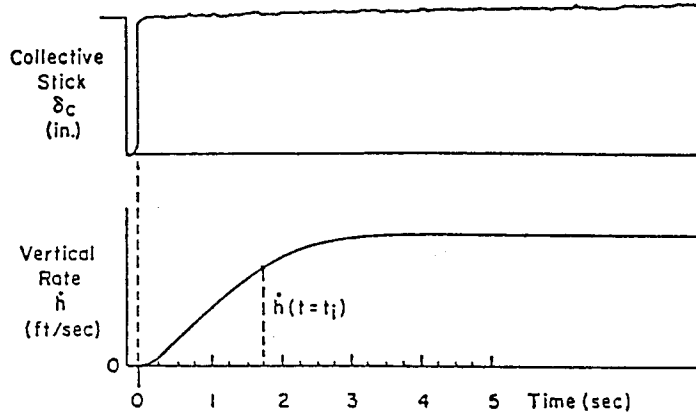
LEVEL	$T_{h_{eq}}^*$ (sec)	$\tau_{h_{eq}}^*$ (sec)
1	5.0	0.20
2	$\infty$	0.30

The equivalent system parameters are to be obtained using the time domain fitting method defined in Figure 8(3.3). The coefficient of determination,  $r^2$ , shall be greater than 0.97 and less than 1.03 for compliance with this requirement.

b. Rationale for Requirement

Factors which must be considered to assure good vertical axis handling qualities are:

- The dynamics of the vertical response (Paragraph 3.3.10.1). This consists of the basic heave damping plus any initial lags in the response.
- Torque limits (Paragraph 3.3.10.2). Torque limits are nearly always set by the maximum stress that can be placed on the transmission to achieve a reasonable service life, and are not usually associated with handling qualities. However, when operating near the maximum rotorcraft performance limits (hot, high, and heavy), the pilot is faced with a divided attention task consisting of height control and avoiding exceedance of the maximum allowable torque. Paragraph 3.3.10.2 is included to insure that setting power does not require excessive pilot attention.



- Obtain  $\dot{h}$  readings (ft/s or m/s) from  $\dot{h}$  response to step collective input at .05 sec intervals for a time span of 0-5 sec (see sketch above) - a total of 101 data points.
- Use a three variable nonlinear least squares algorithm to obtain a best fit curve to this data in the time domain using the following form for the estimated  $\dot{h}$  ( $\dot{h}_{est}$ ),

$$\dot{h}_{est}(t) = K \left[ 1 - e^{-\frac{1}{T_{heq}} (t - \tau_{heq})} \right] \text{ for } t > 0$$

where  $t$  is time (sec) and  $K$ ,  $\frac{1}{T_{heq}}$  and  $\tau_{heq}$  are the variables.

- The function to be minimized is the sum of squares of the error ( $e$ ), defined as,

$$e^2 = \sum_{i=1}^{101} [\dot{h}(t = t_i) - \dot{h}_{est}(t = t_i)]^2$$

where  $t_i$  is the time (sec) at the  $i^{th}$  observed data point (see sketch)

- The goodness of fit of the estimated curve is determined by the coefficient of determination ( $r^2$ ) which is defined as,

$$r^2 = \frac{\sum_{i=1}^{101} [\dot{h}_{est}(t = t_i) - \bar{\dot{h}}]^2}{\sum_{i=1}^{101} [\dot{h}(t = t_i) - \bar{\dot{h}}]^2}$$

where  $\bar{\dot{h}}$  is the mean of the observed  $\dot{h}$ ,

$$\bar{\dot{h}} = \frac{\sum_{i=1}^{101} \dot{h}(t = t_i)}{101}$$

Figure 8(3.3). Procedure for Obtaining Equivalent Time Domain Parameters for the Height Response to Collective Controller

- Vertical axis control power (Paragraph 3.3.10.3). This is generally treated as a performance issue, but is also an important contributor to the vertical axis handling qualities. The achievable steady rate-of-climb is the proper metric as a measure of performance, whereas for handling qualities, it is the initial vertical velocity to a pilot commanded input that matters most. The vertical translational rate at 1.5 seconds after initiating the input has been selected as representative of the short term vertical rate capability (see Paragraph 3.3.10.3). The vertical rate at 1.5 seconds was considered to be long enough after the input that it is not overly sensitive to the inability to achieve a perfect step collective change.

If the torque response has an overshoot characteristic, it will be necessary to ramp in the collective to avoid exceeding the torque limit. This will reduce the achievable vertical rate in 1.5 seconds and is a realistic limitation which effectively requires more vertical axis control power (T/W) if significant torque overshoots exist.

- Rotor RPM governing (Paragraph 3.3.10.4). Poor governing can affect the basic vertical response characteristics and/or cause oscillations or lack of authority in the heading axis. The first of these manifests itself as a reduction in the effective heave damping, and is covered in Paragraph 3.3.10.1, while the second is covered by Paragraph 3.3.9.1 ("Yaw Due to Collective"). Paragraphs 3.3.10.1 and 3.3.9.1 effectively prohibit the undesirable effects of poor RPM governing. In addition, Paragraph 3.3.10.4 ("Rotor RPM Governing") requires that the rotor RPM stay within the specified limits during the performance of all required Mission-Task-Elements.

The requirement in Paragraph 3.3.10.1 of the specification has been formulated for a vertical rate Response-Type, typical of most helicopters. A time domain equivalent systems approach was found to be the best compromise for describing and specifying the vertical rate response in the presence of possible higher order engine/governor dynamics. This will be discussed further in "Supporting Data." The requirement places limits on the equivalent time constant and time delay in the first five seconds of the vertical rate response following a step collective input. The inverse equivalent time constant and equivalent time delay are measures of the helicopter heave damping and initial lags over that five-second time period. The requirement is applicable for only five seconds following the input since, for most operations, this is a sufficiently long time to expect a first-order response. In addition, five seconds corresponds to the Level 1 limit on  $T_{h_{eq}}$ , so it effectively requires a first-order-appearing response for one time constant.

This requirement was selected in lieu of a rise time and response shaping criterion, recommended in earlier drafts of the specification.

It is an improvement over rise time since it considers the response over a given period of time. A single point method such as time to 50 percent of peak amplitude was found to be overly sensitive to uncertainties in the responses typically obtained in a flight environment.

### c. Supporting Data

This criterion is based primarily on flight test data generated through experiments performed by the National Research Council (NRC) of Canada and by the Army at Ames Research Center.

#### 1. Definitions of the Parameters

The transfer function of vertical rate to collective for typical helicopters is well approximated by:

$$\frac{\dot{h}}{\delta_c} = \frac{Z_{\delta_c}}{(s + 1/T_{heq})} e^{-r_{heq}s}$$

For example, see the step response of the CH-47B shown in Figure 1. A simple rise time parameter such as time to 50 percent or 63.2 percent of peak vertical rate is sufficient to adequately describe such a response. There can be exceptions to this first-order form, however. Several possible deviations from the first-order form are shown in Figure 1. These deviations, which take the form of an added first (or higher) order lag, are caused by the engine/governor, actuator or mechanical linkages, or a combination of these factors. The unusual nature of the XV-15 governor (Reference 175) also gives rise to a non-first-order type of response as shown in Figure 1.

Experiments have shown that pilots are not overly sensitive to such deviations from the first-order response shape as long as the curve is at least concave down and the initial lags are not excessive. On this basis, it was decided to characterize the height response to collective inputs as an equivalent first order response and a pure time delay. Using this approach, higher order dynamics would be reflected in the equivalent parameters,  $T_{heq}$  and  $r_{heq}$ . A five second time period from the onset of the collective step was chosen for the equivalent systems matching procedure based on the duration required to expose the modes of interest to a pilot involved in closed loop operations in the height axis.

As specified in the requirement, a nonlinear least squares method was selected to match the first order time domain model to available flight and simulator step responses. A modified Gauss-Newton (Reference 177) iterative algorithm was used to minimize the square of the error. The coefficient of determination ( $r^2$ ) is a measure of the goodness of fit of the model to the data and is also specified in the requirement in order to ensure a reasonable match. A coefficient of determination that lies outside the limits specified in the requirement would indicate an excessive deviation of the response from the first order form.



## 2. Review of Flight Test Data

Pilot rating data from the experiments outlined in References 173 and 174 are shown plotted against inverse equivalent time constant and equivalent time delay in Figure 2. The  $r^2$  values in all cases fell within the limits specified in the requirement. The boundaries are also shown in Figure 2.

Two separate height control experiments are described in Reference 174. In the first experiment, heave damping was varied in conjunction with two different pitch/roll control systems (RCAH and ACAH). The pitch/roll characteristics were rated Level 1 when evaluated separately with known good heave dynamics. The configurations were evaluated over a hover course consisting of precision hover, landing and bob-up/down tasks. The average pilot ratings for the bob-up/down and landing tasks are shown in Figure 2 (square symbols) for the two pitch/roll control systems. The ratings show a general degradation with decreasing  $1/T_{heq}^*$  with only a minor difference between the two pitch/roll control

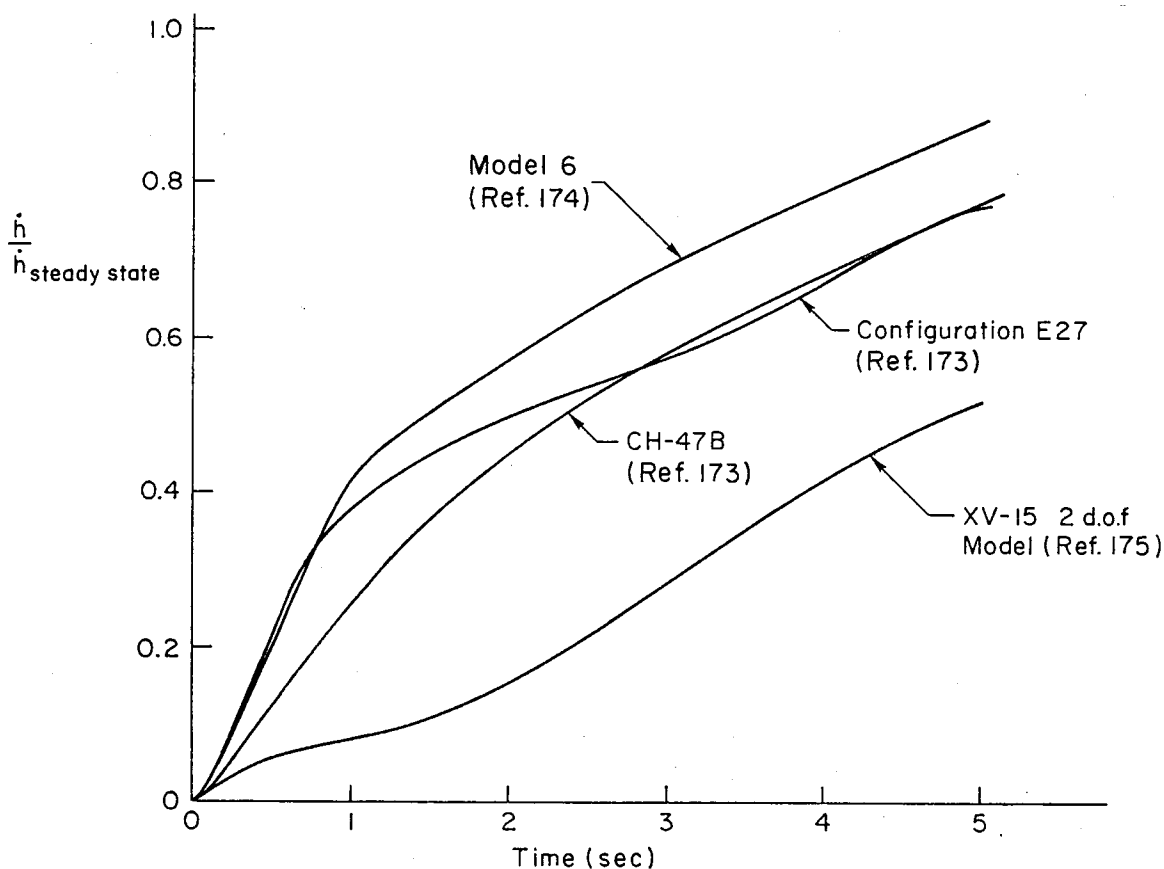


Figure 1(3.3.10.1). Experimental and Actual Vertical Rate Responses to a Step Collective at  $T = 0$

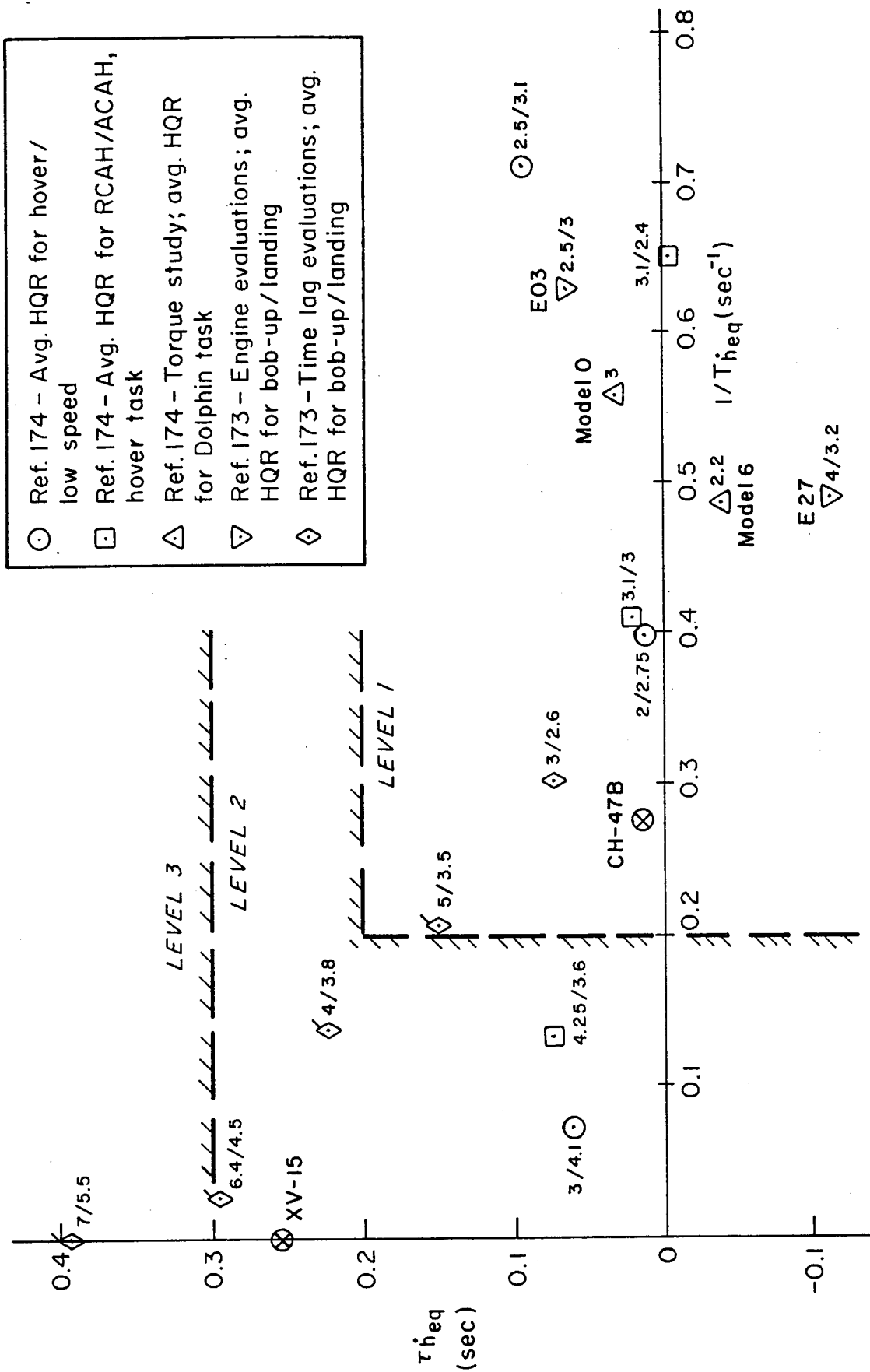


Figure 2(3.3.10.1). Supporting Data for Height Control Requirements (Flagged Symbols Indicate Matches to Theoretical Models; All Others are Matches to Actual Flight Data)

systems. Control sensitivity was not separately optimized by the pilots in the experiment.

Control sensitivity was pilot optimized in the second experiment reported in Reference 174, where the effects of decreasing heave damping and thrust-to-weight ratio (T/W) on pilot performance were evaluated over five tasks: hover, landing, bob-up, quickstop and dolphin. The ratings for configurations with the best T/W ratio of 1.10 were averaged over pilots and tasks to represent hover (hover, landing and bob-up) and low speed (quickstop and dolphin) tasks. These are shown in Figure 2 (circles). The averaged ratings for the hover tasks show a relative insensitivity to the equivalent time constant of the response (i.e., heave damping). The averaged ratings for the low speed tasks, however, show some degradation with decreasing  $1/T_{heq}$ . Time delay was not a factor in these evaluations. The pitch/roll Response-Type was Rate (no Attitude Hold) and was Level 1 when evaluated separately.

The effect of system lags on pilot opinion was investigated on a motion base simulator in Reference 176 (NASA Ames VMS) and in flight in References 111 and 173 (NASA Ames CH-47). System lags were simulated in each case by cascading a first-order lag with the first-order heave dynamics of the helicopter. The results of these three experiments are shown in Figure 3 (reproduced from Reference 173). It is important to note the differences in task and pilot rating scale when comparing the results of References 111 and 173. The difference in task derives mainly from the relatively moderate degree of aggressiveness demanded by the circuit flying task of Reference 111 compared with that required for the more aggressive bob-up task of Reference 173. In addition, pilot ratings from the Cooper scale of Reference 111 could be at least one rating point higher (for Cooper ratings of 3 and above) when re-interpreted on the Cooper-Harper scale used in Reference 173.

The average pilot ratings for each of the two tasks (bob-up and landing) evaluated in the more recent Reference 173 experiment are also included in Figure 2, where it can be seen that as  $\tau_{heq}$  increases,  $1/T_{heq}$  decreases. The adverse ratings given to these configurations could therefore be due to high time delay, sluggish rise time, or both.

The decrease of  $1/T_{heq}$  with increasing  $\tau_{heq}$  for these configurations is due to, and is therefore an indication of, the increasing dominance of the lag as the lag time constant is increased, in the first five seconds of the step response. A low value of  $1/T_{heq}$  by itself, in light of the Reference 174 data showing pilot tolerance of low  $1/T_{heq}$ , does not explain the Level 2 and 3 ratings given to the high time delay configurations of Reference 173. It is therefore more likely that the pilot ratings are degraded as a result of the adverse "short-term" response described by  $\tau_{heq}$ .

In addition to the effects of system lags, the flight experiment of Reference 173 investigated the effect of rpm cueing on pilot rating. The configurations evaluated were comprised of three different engine/governor combinations with response characteristics ranging from good (fast reacting - configuration E03) to bad (slow reacting - configuration E27). The higher order dynamics introduced by these engine/

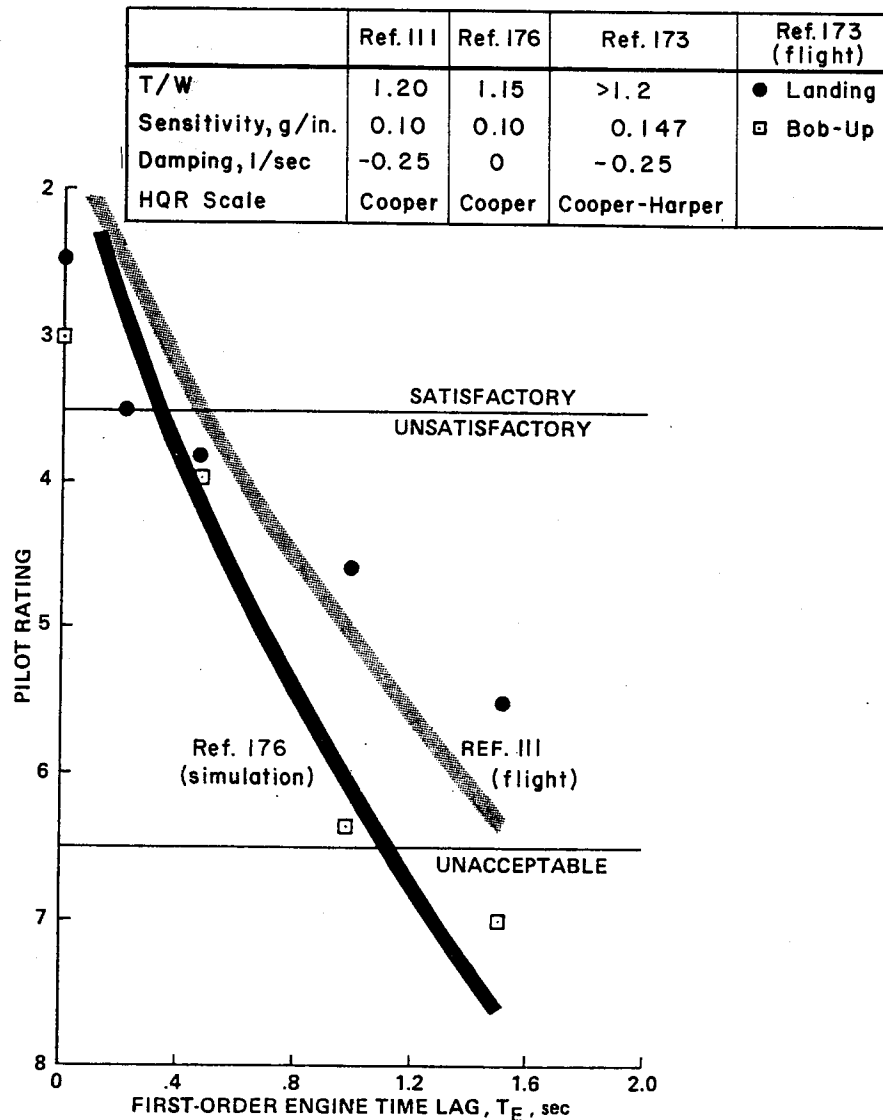


Figure 3(3.3.10.1). Comparison of Pilot Rating Results of References 111, 173, and 176 (Actual Data for Reference 173; Results from References 111 and 176 are Shown as Faired Lines)

governor combinations are exhibited by the step response of configuration E27 in Figure 1. To isolate and evaluate the effect of the vertical rate response dynamics on pilot opinion, the two extreme (fast and slow) engine/governor combinations were evaluated with no rpm cueing. The average pilot ratings for these two configurations, shown on Figure 2, agree with the Level 1 boundary. Pilot comments for the slow engine/governor configuration (E27) indicate that the higher order dynamics, while being noticeable, did not (with one possible exception) adversely affect their performance.

The final flight test data to be considered are the engine/governor evaluations performed by the NRC, Canada and detailed in Reference 174. In this experiment the simulated engine torque and rotor rpm were displayed to the pilot and reflected the various engine/governor dynamics under evaluation. Adverse torque dynamics (reflected in the torque gauge), combined with marginal values of T/W, proved to be the primary cause of Level 2 and 3 pilot ratings in this experiment. Two configurations (models 0 and 6, Figure 2) with significantly different response shapes were given Level 1 ratings by the pilots. The response of model 0 represented a "perfect" engine/governor and was hence first-order in nature. Model 6, however, represented an engine/governor combination which produced a distinctly nonlinear vertical rate response (Figure 1).

These results suggest that the pilot is not sensitive to the detailed shape of the vertical rate response, and lend support to the use of a lower order equivalent system criterion.

The distinctly different response shapes caused by the slow reacting engine/governor configurations in Reference 173 (Configuration E27) and Reference 174 (Model 6), resulted in negative values of  $\tau_{heq}^*$  (Figure 2). This is an artifact of the higher order dynamics and is therefore an indication of the presence of such dynamics in the response (see Figure 4). Negative values of  $\tau_{heq}^*$  tend to arise from an initial rapid response which subsequently "bleeds off."\* Since there is no evidence that pilots object to this, there are no restrictions on negative values of  $\tau_{heq}^*$ .

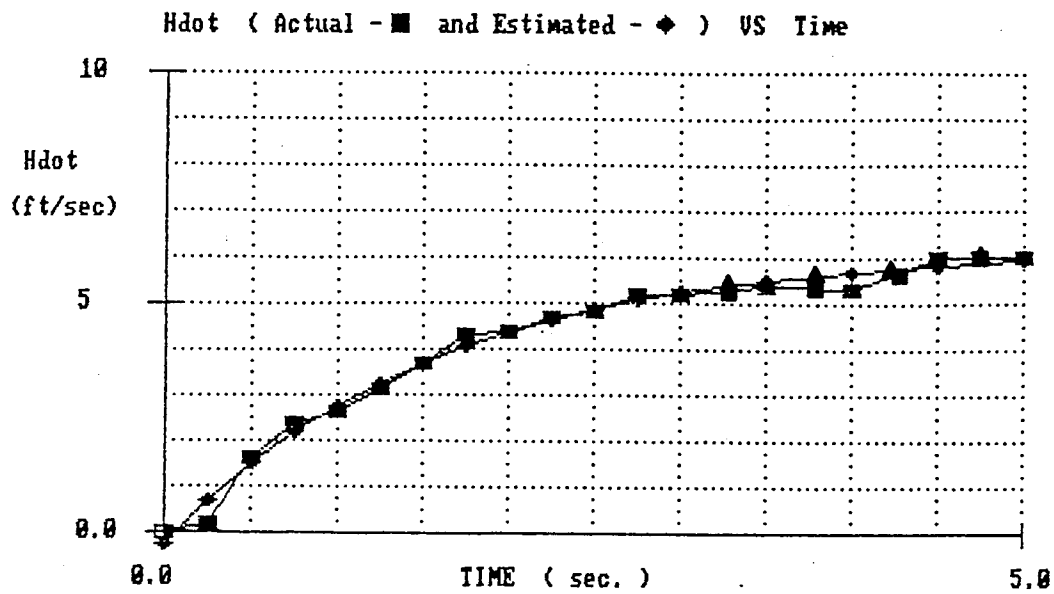
### 3. Definition of Limits

A Level 1 limit of  $0.2 \text{ sec}^{-1}$  was set for  $1/T_{heq}^*$  ( $T_{heq}^* = 5 \text{ sec}$ ) on the strength of the data in Figure 2. This limit may be conservative for relatively non-aggressive tasks such as the hover tasks in Reference 174, where a greater tolerance was shown by the pilots to configurations with low values of  $1/T_{heq}^*$  (circles in Figure 2). This result is similar to that obtained from the flight tests reported in Reference 111 where, given a configuration with a reasonable steady state climb rate capability, a vertical acceleration-type system ( $Z_w = 0$  or  $1/T_{heq}^* = 0$ ) was deemed Level 1 (on average) by three pilots. The task in Reference 111 was a relatively non-aggressive acceleration and climbout to a race-track pattern at 45 kts.

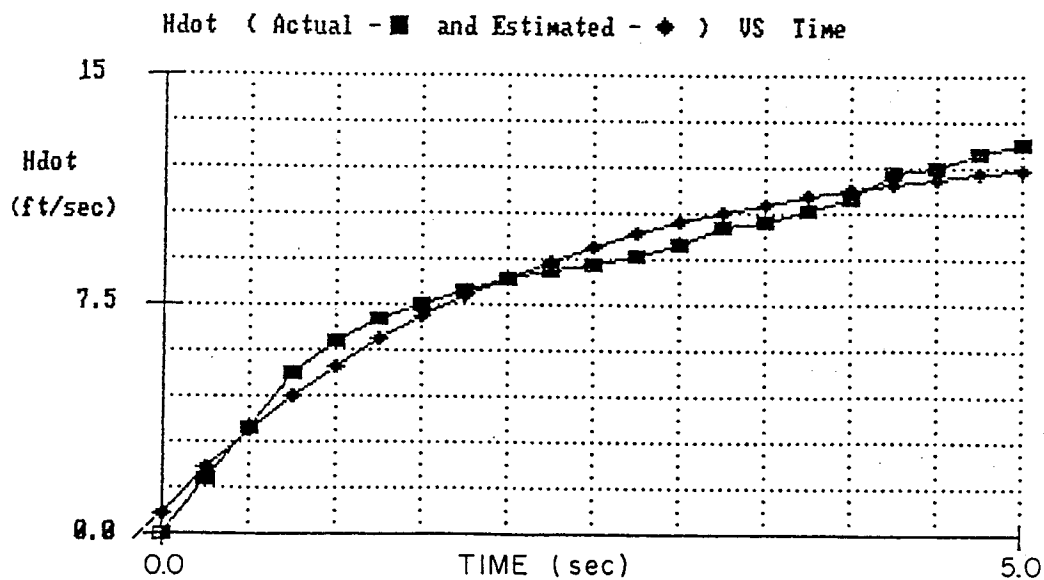
Most present day helicopters should have no difficulty meeting the Level 1 requirement on  $1/T_{heq}^*$ . The values of  $Z_w$  for conventional helicopters (Reference 28) support this: in the presence of reasonably good rotor RPM governing, negligible system lags or time delay,  $1/T_{heq}^*$  is well approximated as  $-Z_w$ .

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\*Negative values of  $\tau_{heq}^*$  result only when the initial rapid response has a reasonably long duration, such as in Figure 1, Model 6, as opposed to small initial response, as for the XV-15 two degrees-of-freedom model in Figure 1.



a) Configuration E03:  $1/T_{\dot{h}_{eq}} = 0.634 \text{ sec}^{-1}$ ;  $\tau_{\dot{h}_{eq}} = 0.065 \text{ sec}$ ;  
 $r^2 = 0.988$



b) Configuration E27:  $1/T_{\dot{h}_{eq}} = 0.492 \text{ sec}^{-1}$ ;  $\tau_{\dot{h}_{eq}} = -0.111 \text{ sec}$ ;  
 $r^2 = 0.974$

Figure 4(3.3.10.1). Effect of Higher Order Dynamics on the LOES Parameters  
 (Configurations from Reference 173)

The Level 2 limit of zero on  $1/T_{h_{eq}}^*$  (i.e.,  $1/T_{h_{eq}}^* = \infty$ , a vertical acceleration type response) reflects the lowest possible value for that parameter.

The data plotted on Figure 2 form the basis for the Level 1 limit of 0.20 sec and the Level 2 limit of 0.3 sec on equivalent time delay.

#### 4. Review of Simulation Data

A review of available ground-based simulator data is useful for providing further insight, although there are sufficient questions regarding absolute validity to eliminate such data from consideration for developing the boundaries.

The engine/governor configurations evaluated in the Reference 110 simulation were the same as those evaluated in the flight tests of Reference 173 and, therefore, provide the basis for an interesting flight/simulation result comparison. The equivalent model parameters for the Reference 110 configurations were calculated from computer-generated time responses. A pure time delay of 185 msec was added to the computer-generated responses to represent simulator motion and visual delays. The step responses used represented the largest collective inputs allowed by the torque limits set in the engine model. The average pilot ratings for evaluations with no rpm cueing from the simulation and flight tests are shown in Figure 5. The only average Level 2 ratings in Figure 5 were those given to the slow engine/governor configurations (as evidenced by the negative value of  $r_{h_{eq}}^*$ ) in the simulation (Reference 110). Individual ratings (also shown in Figure 5) indicate considerable disagreement between the evaluation pilots about these configurations. These disagreements are probably a result of different pilot control strategies, i.e., the higher-order effect would only be noticeable to a pilot using a relatively open-loop control strategy consisting of long-duration collective step inputs. These effects were noticed but not rated as harshly in flight (Reference 173). The pilot ratings in Figure 5 show the familiar trend towards more degraded ratings from ground-based simulation than flight for a given set of dynamics.

#### d. Guidance for Application

Demonstration of compliance requires the measurement of the vertical rate response at 0.05 sec intervals over the first 5 sec following a step collective input. This corresponds to a 20 Hz signal sample rate which is within the capabilities of the digital data acquisition equipment used today.

The procedure for calculating the equivalent time domain parameters is defined in the requirement. An iterative algorithm is required to perform the least-squares fit because of the non-linear function being used. A modified Gauss-Newton iterative algorithm (defined in Reference 177) was used to calculate the equivalent parameters shown in the "Supporting Data" section.

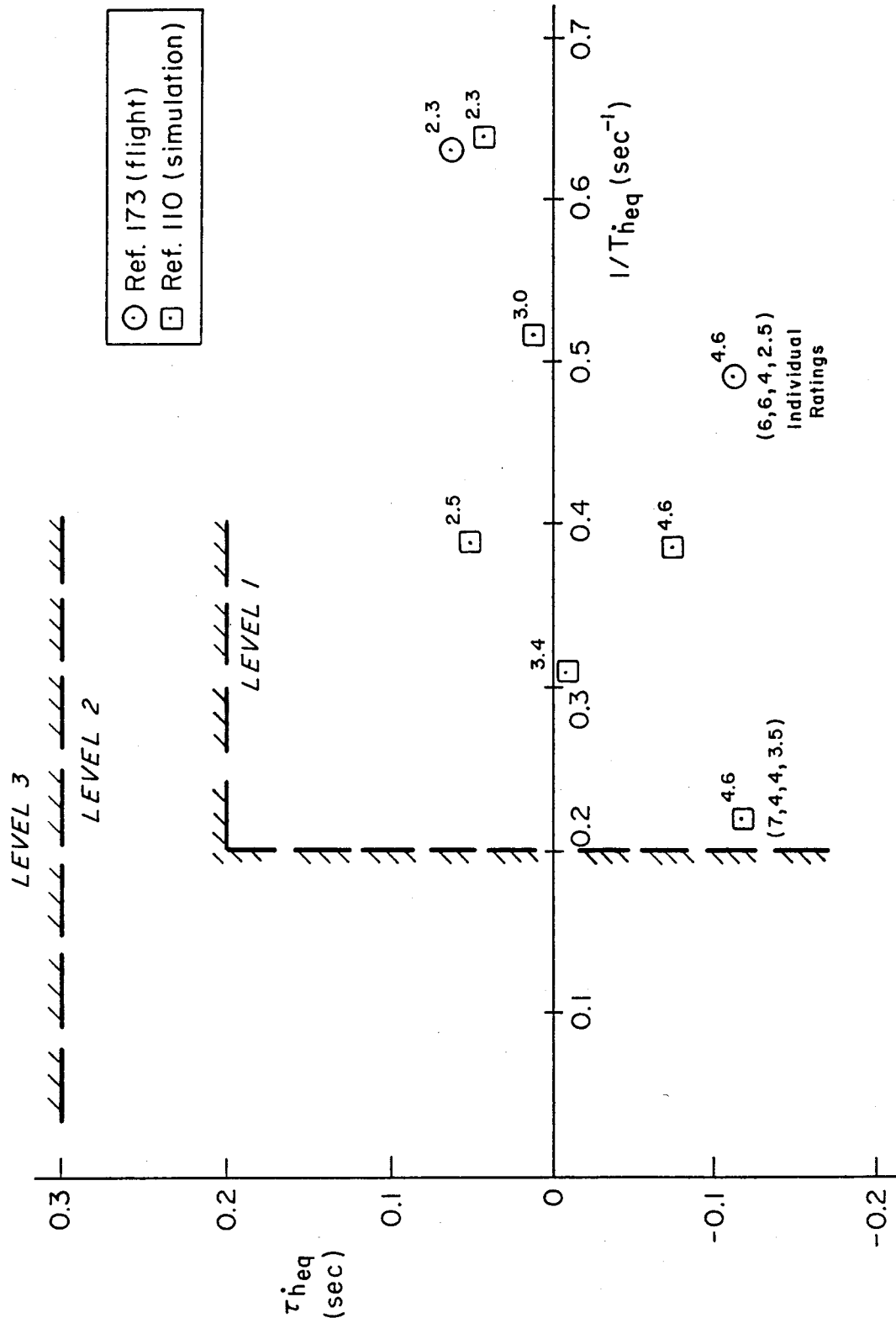


Figure 5(3.3.10.1). Comparison of Simulator and Flight Data on Engine/Governor Effects



The shape of the collective input must be as step-like as possible as any significant ramping of the input may result in an abnormally high equivalent time delay. The number of points used in the match is an important parameter. A coarser match (using fewer data points) may be performed on first-order-like responses with little effect on the results. Reducing the number of points in the matching procedure for a response showing any higher order effects can result in a significantly different equivalent time delay value.

e. Related Previous Requirements

There are no related previous requirements in MIL-H-8501A. MIL-F-83300 requires that heave damping be greater than zero. The current requirement sets limits on the equivalent time constant (approximating heave damping) as well as equivalent time delay of the vertical rate response. This criterion is an improvement on the previous requirement as equivalent time delay is an important parameter which determines pilot performance in closed-loop operations, and since the specified method does not require the identification of vehicle heave damping.

a. Statement of Requirement

3.3.10.2 Torque Response. Torque, or any other parameter displayed to the pilot as a measure of the maximum allowable power that can be commanded without exceeding engine or transmission limits, shall have dynamic response characteristics that fall within the limits specified in Figure 9(3.3). This requirement shall apply if the displayed parameter must be manually controlled by the pilot to avoid exceeding displayed limits.

b. Rationale for Requirement

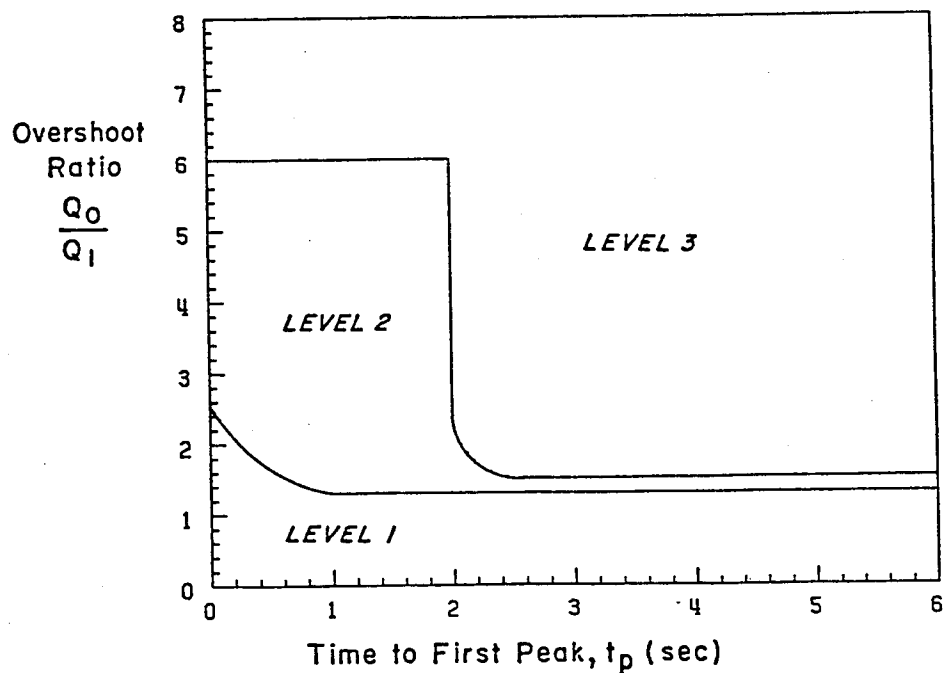
The need to constantly monitor a manually controlled critical parameter, such as torque, increases workload as it distracts the pilot from the primary task. Unpredictable behavior of this parameter in response to control inputs tends to cause the pilot to divide his attention away from the primary task resulting in reduced efficiency and aggressiveness. A requirement was therefore necessary to ensure that the response of this parameter is predictable, thereby allowing the pilot to manipulate the controller with minimum attention demands to achieving the limit value.

This requirement was based on data which focused on torque as a parameter of concern. It can also apply to another parameter if it is designated as being critical to engine/transmission limits and can be directly controlled by the pilot.

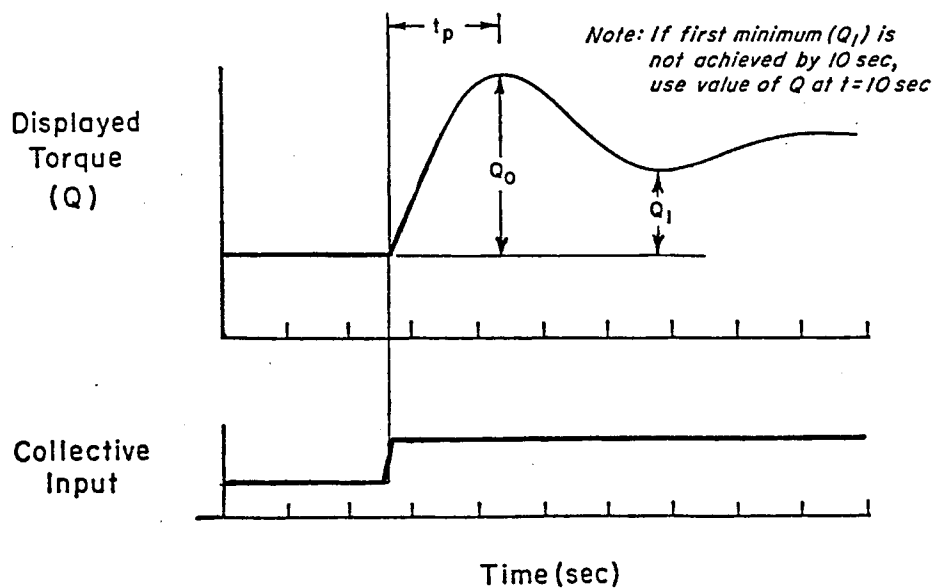
c. Supporting Data

The primary support for this requirement comes from the flight tests performed by the National Research Council (NRC) of Canada (reported in Reference 174). This data was augmented by data from a simple fixed-base simulation undertaken at STI, and a second fixed-base simulation conducted at NASA-Ames provided additional data. The results of the simulations have not been published previously.

A time history of the dynamics of the NRC flight test configurations evaluated in the engine/governor experiment is shown in Figure 1. The torque responses from this study provided the basis for this requirement and the basis for the range of configurations evaluated in the fixed-base simulations. The STI simulation consisted of a sidestick-type controller and two CRT scopes displaying altitude and torque. The heave dynamics were first-order with a  $1/T_{heq}$  of 0.3 rad/sec. The torque dynamics were based on the models used in the NRC experiment. The evaluation pilot for the STI simulation was an experienced helicopter pilot who was also an evaluation pilot for the NRC flight tests. The task was a simulated bob-up/down while maintaining torque under a prescribed limit. The results from the NRC and STI experiments identified the torque overshoot magnitude and the damping of the response as the parameters of concern to the pilot.



*a) Requirement on Dynamics of Displayed Torque Based on Step Collective Change*



*b) Definition of  $Q_0/Q_1$  and  $t_p$  for Displayed Torque Requirement*

Figure 9(3.3). Displayed Torque Response Requirement

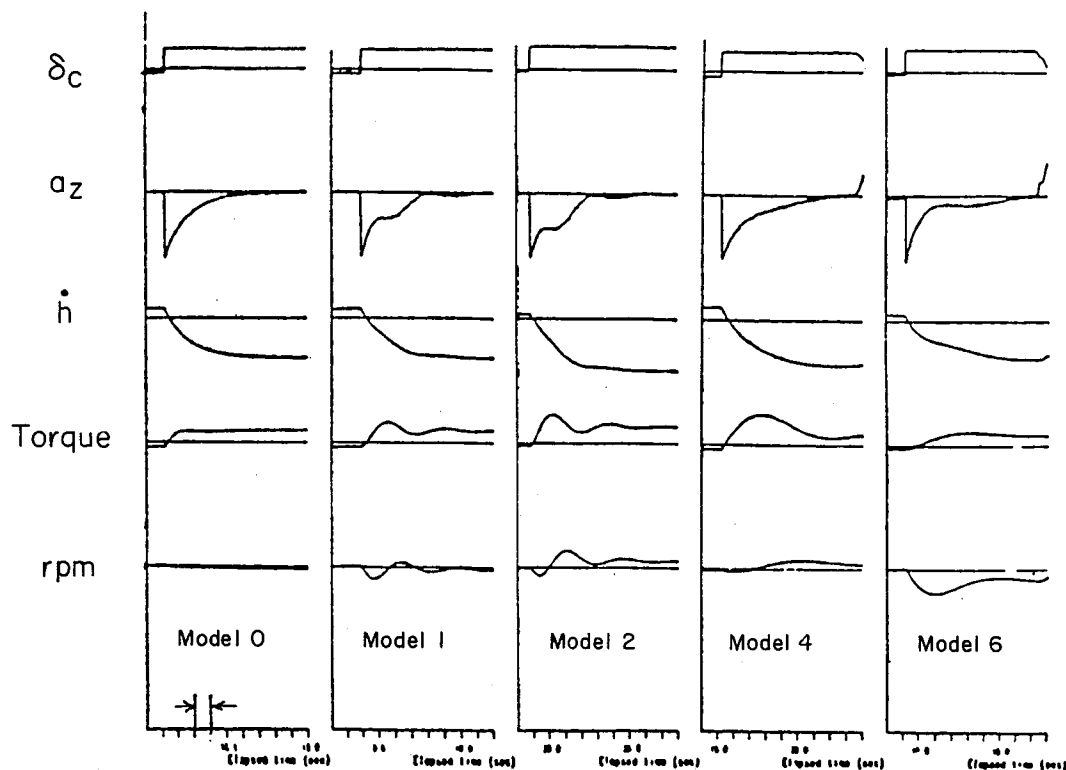


Figure 1(3.3.10.2). Engine Governor/Rotor Models (NRC)

The fixed-base simulation performed at NASA-Ames used these two earlier experiments as a basis for modelling and setting initial configurations. Four pilots performed the evaluations: three test pilots and one line pilot. Ninety percent of the evaluations were obtained from the two Army test pilots, both with extensive rotary wing experience. The tasks for performing the handling qualities evaluations were a timed 15 ft bob-up/bob-down maneuver and a maximum rate-of-climb maneuver.

The data from the Ames, NRC, and STI experiments are plotted against the Figure 9a(3.3) requirement in Figure 2. The pilot ratings and subsequent Level boundaries on Figure 2 imply that larger overshoot is acceptable if the initial peak occurs in a relatively short time. In contrast, if the initial peak occurs later an equivalent overshoot may become unsatisfactory because it is slow responding and unpredictable and therefore difficult to control.

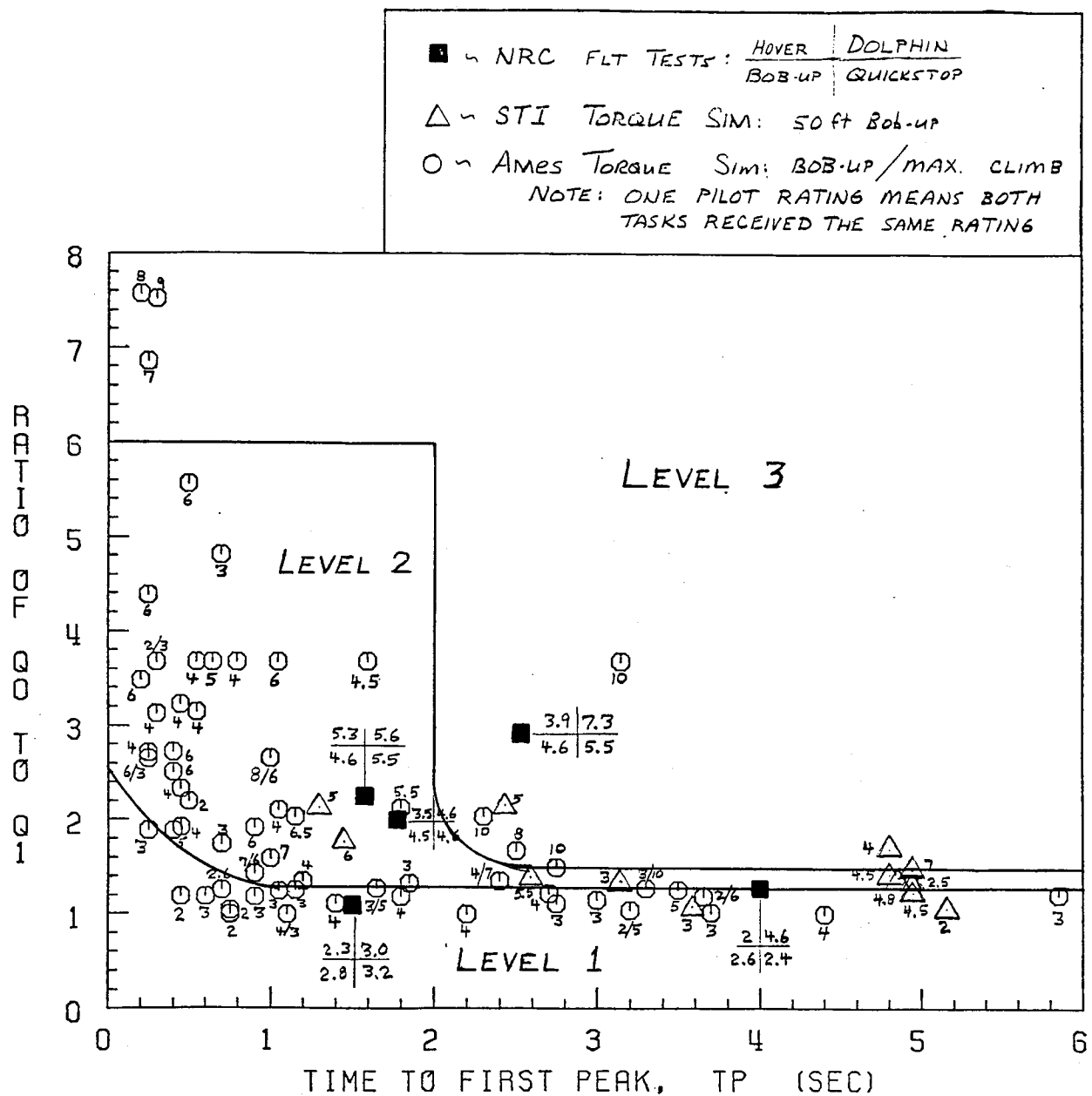


Figure 2(3.3.10.2). Supporting Data for Displayed Torque

d. Guidance for Application

Demonstration of compliance is to be accomplished by performing a rapid vertical controller input (usually collective, but it may be any other controller used by the pilot to control displayed torque), and obtaining a measurement of the appropriate dynamic response parameters. It is anticipated that these dynamics may be measured from the same maneuver used to demonstrate compliance for the vertical axis control power (3.3.10.3). The maneuver shall be initiated from a steady hover in the most critical conditions.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.3.10.3 Vertical Axis Control Power. While maintaining a spot hover with the wind from the most critical direction at a velocity of up to 35 knots (18 m/s), and with the most critical loading and altitude, it shall be possible to produce the vertical rates specified in Table 5(3.3), 1.5 seconds after initiation of a rapid displacement of the vertical axis controller from trim. Applicable engine and transmission limits shall not be exceeded.

TABLE 5(3.3). VERTICAL AXIS CONTROL POWER

LEVEL	ACHIEVABLE VERTICAL RATE IN 1.5 SECONDS -- m/s (ft/min)
1	0.81 (160)
2	0.28 (55)
3	0.20 (40)

b. Rationale for Requirement

Vertical axis control power is usually considered to be a performance parameter, and therefore outside the influence of a handling qualities specification. As a performance parameter, the vertical control power is usually specified as a rate-of-climb which must be achieved under the most adverse loading conditions, altitude, and temperature necessary to complete the mission. However, testing has shown that a sluggish vertical response has handling qualities implications. For example, the flight tests of References 111 and 174 indicated Level 2 pilot ratings due to an inadequate initial vertical response, even though sufficient steady rate-of-climb capability was available to accomplish the task. This deficiency manifests itself as a tendency to overtorque the engine/transmission with an attendant increase in pilot workload required to prevent such an occurrence. This phenomenon is aggravated by low heave damping where increased collective activity is required to provide the necessary equalization.

The requirement specifies the hover flight condition as it is most critical in terms of excess available power. The implied Mission-Task-Elements are therefore the bob-up and rapid acceleration from hover.

Several parameters were considered as representative of the initial vertical response following a vertical axis controller input: the initial vertical acceleration, the thrust-to-weight ratio (T/W), and the vertical rate some short time following the initiation of a maximum controller input. The initial normal acceleration was rejected as being too difficult to measure due to accelerometer noise, confusion over the effect of dynamic inflow which causes a short  $n_z$  pulse, and an

inability to account for the effect of systems with torque overshoot following a step collective input.  $T/W$  can be measured from the achievable steady state climb rate, but this requires a considerable altitude range which may not be available in the down direction. More significantly, the steady climb method does not account for the effects of additional lags that may show up in the vertical rate response. For example, if the helicopter altitude-rate response is characterized by

$$\dot{h}_{\max} = \frac{g(T/W - 1)}{(s - Z_w)(T_L s + 1)}$$

it can be seen that variations in the first order lag,  $T_L$ , will have no effect on the steady height response, but will strongly impact the initial response, which is of primary interest. Therefore, it was decided to use the achievable vertical-rate response shortly following the initiation of the collective input. The choice of 1.5 seconds as the reference time for the requirement was based on the generic characteristics of the height response following a pure unit step collective input. These are shown in Figure 1 for the configurations tested in Reference 174. It would have been theoretically desirable to eliminate the effect of heave damping\* on  $\dot{h}(1.5)$  by selecting a reference time less than 1 second. However, it was judged that this would have made the requirement overly sensitive to the ability to produce a nearly ideal step collective input. The use of 1.5 seconds tends to minimize that problem

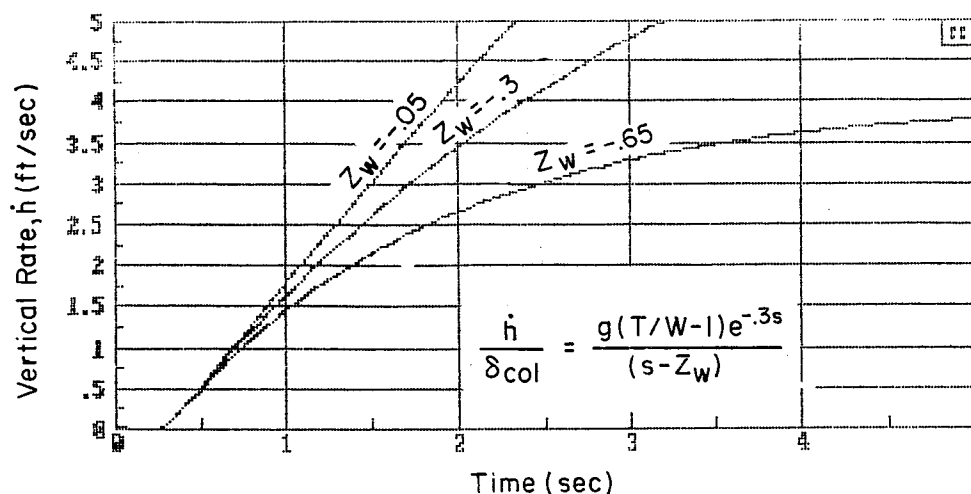


Figure 1(3.3.10.3). Generic Effects of  $Z_w$  on Vertical Rate

\*Heave damping is separately covered by specifying a limit on  $T_{heq}$  in Paragraph 3.3.10.1.



without making the requirement overly dependent on the value of heave damping. In practice, the flight data of Reference 174 show that for extremely low values of T/W, some reduction in heave damping is desirable (discussed further below), indicating that some sensitivity of the parameter to  $Z_w$  is justified.

c. Supporting Data

The data used to support this requirement are the flight tests performed by the NRC in direct support of this specification (Reference 174), and earlier tests performed at NASA Langley using the variable stability CH-47 (Reference 111).

The handling quality rating (HQR) data for the bob-up task in the Reference 174 flight tests is shown as a function of  $Z_w$  and  $T/W^*$  in Figure 2. The rating spread at the lowest T/W is interpreted as an overriding pilot desire for initial response rather than due to insufficient heave damping (i.e., better ratings for lower  $Z_w$ ). These pilot rating data are plotted vs. the maximum achievable vertical-rate 1.5 seconds following initiation of the collective input ( $\dot{h}(1.5)$ ) in Figure 3, where the correlation is excellent.

The task in the NASA Langley CH-47 experiment (Reference 111) was a rapid transition from hover to forward flight followed by a climb to 400 ft. The first part of this task represents the rapid acceleration Mission-Task-Element. However, the second part is strongly dependent on the climb performance, and we might expect the data to correlate better with rate-of-climb. These data, as well as the Reference 174 NRC data, have been plotted on a grid of  $\dot{h}(1.5)$  vs. the maximum achievable steady rate-of-climb with the results shown in Figure 4. This format illustrates a distinct requirement for an  $\dot{h}(1.5)$  of at least 160 ft/min for Level 1 regardless of the steady rate-of-climb capability. The Level 2 limit is less well defined, but a value of 55 ft/min seems reasonably well justified (see also Figure 3). The Level 3 limit is based on an extension of the faired Reference 174 data in Figure 3.

The steady rate-of-climb required for Level 1 for the CH-47 acceleration/climb task was about 500 ft/min, while 300 ft/min was adequate for the NRC bob-up/down task (see Figure 4). Limits on steady rate of climb are not included in this handling qualities specification because of strong industry and government opinion that it might conflict with performance requirements specified elsewhere.

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\*T/W was obtained using the steady rate-of-climb method based on the following expression:

$$T/W = \frac{\dot{h}_{ss} Z_w}{g} + 1$$

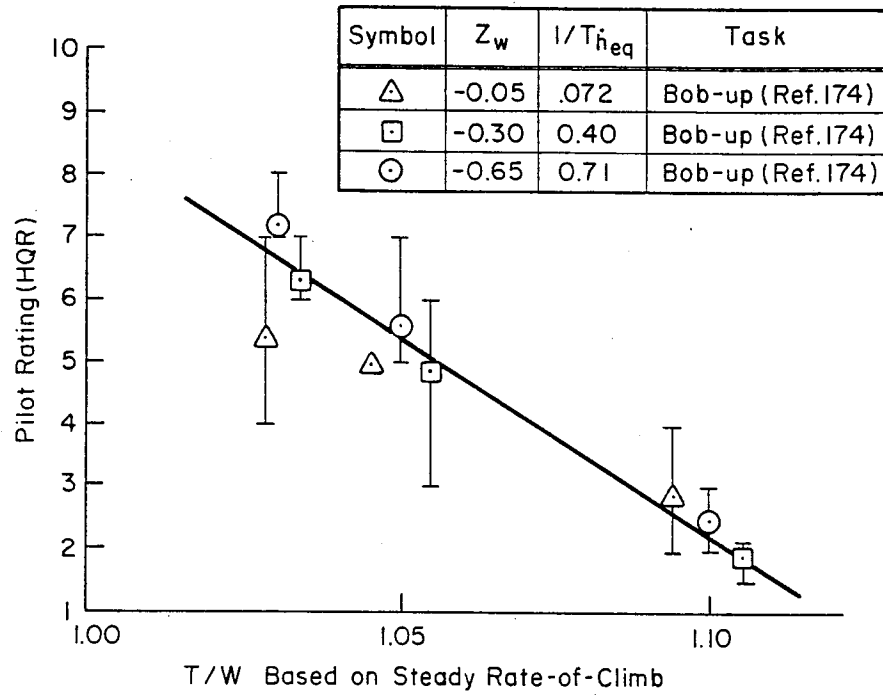


Figure 2(3.3.10.3). Effect of T/W on Pilot Ratings

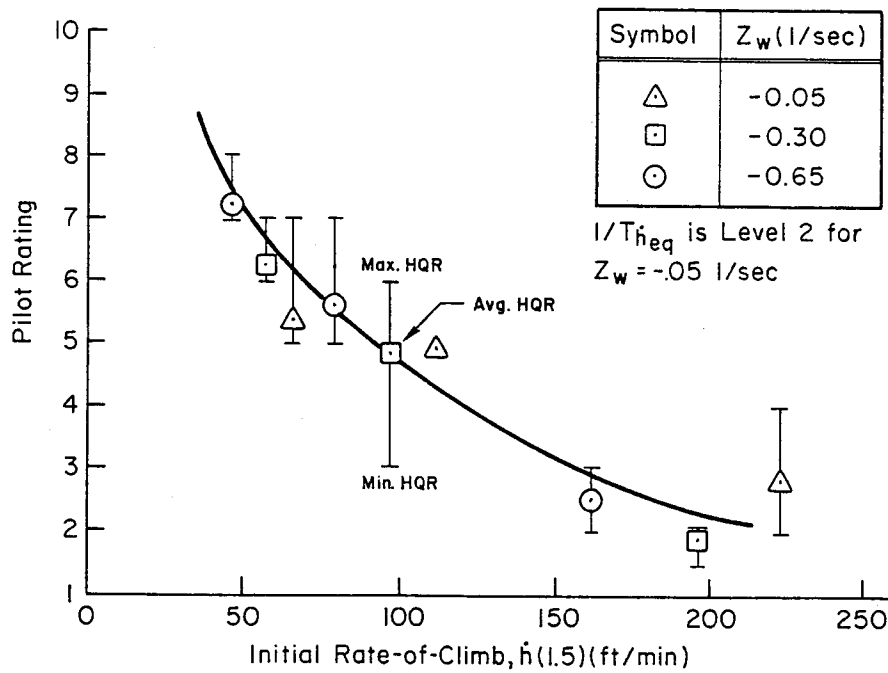


Figure 3(3.3.10.3). Reference 174 Flight Test Data for Bob-Up Maneuver

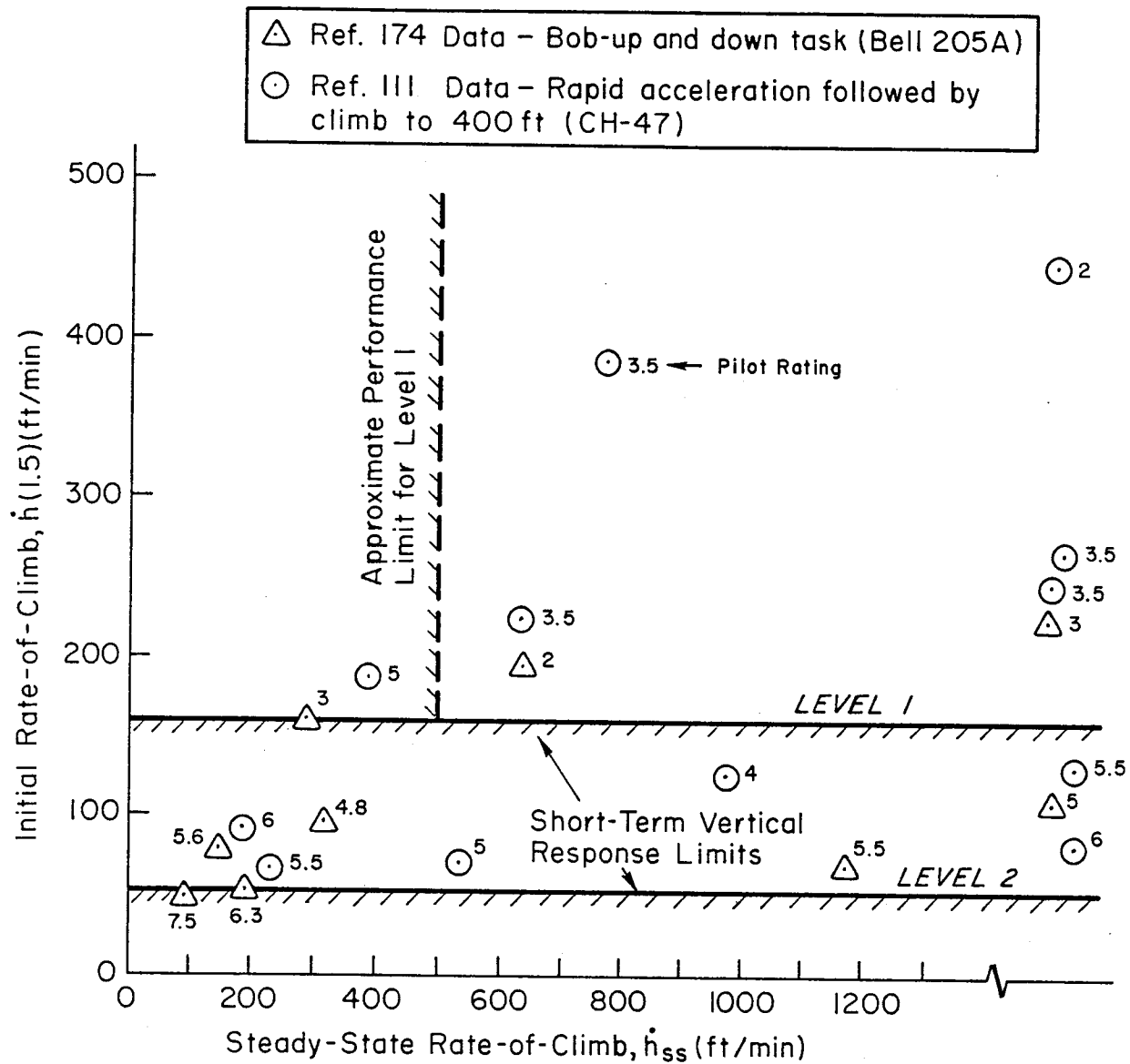


Figure 4(3.3.10.3). Initial Response vs. Steady Rate-of-Climb

d. Demonstration of Compliance

Demonstration of compliance is to be accomplished by performing a rapid vertical controller (usually collective) input to the torque red-line or power (RPM) limit and obtaining a measurement of the rate-of-climb 1.5 seconds after initiation of the input. The maneuver is to be initiated from a steady hover. Clearly, the most critical flight condition will be at maximum gross weight, and at the density altitude which defines the upper limit of an Operational Flight Envelope. Since finding such a density altitude environment to conduct these tests could be impractical, it is expected that the manufacturer will calculate an equivalent reduction in red-line torque, or an equivalent increase in aircraft weight which will simulate the required condition.

The use of  $\dot{h}(T)$  as a criterion for vertical axis control power inherently accounts for any tendency for the RPM to droop unacceptably or for torque to overshoot its steady value for a step increase in collective. Such droops or overshoots require the pilot to lag or "ramp in" the power while performing compliance testing to avoid exceeding the power train limits. This has a considerable negative effect on  $\dot{h}(1.5)$  which is intuitively correct, and is consistent with the Reference 174 flight results where pilots gave poor ratings to configurations with low T/W combined with a tendency for torque overshoot.

e. Related Previous Requirement

MIL-F-83300 specifies control power in terms of incremental vertical acceleration capability and T/W. The current method is both more straightforward to test in flight and more directly applicable to a handling qualities specification.

a. Statement of Requirement

3.3.10.4 Rotor RPM Governing. The rotor RPM shall remain within the limits set by the Service Flight Envelopes during the execution of all specified Mission-Task-Elements conducted within the Operational Flight Envelopes. It shall be possible to meet all directional control requirements at the lowest sustained operating RPM limit.

b. Rationale for Requirement

Poor rotor RPM governing can affect the basic vertical response characteristics and cause oscillations or lack of authority in the heading axis. While these characteristics are specified elsewhere, it seems necessary to explicitly require that the rotor RPM remain within the specified limits for all of the required Mission-Task-Elements.

c. Supporting Data

No explicit supporting data is available for this requirement, although there is some limited information on the effects of RPM control in Reference 110. This data is summarized in Figure 1. The solid circles in Figure 1 represent averaged pilot rating data for the configurations with full RPM cueing, consisting of a gauge, overspeed/underspeed warning light, and rotor RPM sound system. The open circles are the averaged data with the above cues removed. The no-cueing cases were accomplished by presenting the pilot with a constant or ideal RPM display and sound system while allowing normal RPM to couple into the engine and vehicle models. The datum point for the ideal governor in Figure 1 indicates the best rating possible for this particular vehicle and task and forms a basis for comparing all other data on the figure. Note that with no cueing, the slow-governor-response case couples sufficiently with the vehicle to provide pilot rating degradation of about 1-1/2, and an additional degradation of about two ratings occurs with full RPM cueing. For the intermediate- and fast-responding governors, however, the degradation in pilot ratings appears to be a result only of RPM control.

d. Demonstration of Compliance

This requirement will be considered met if the rotor RPM does not exceed the specified limits during the conduct of all of the required Mission-Task-Elements.

e. Related Previous Requirements

None.

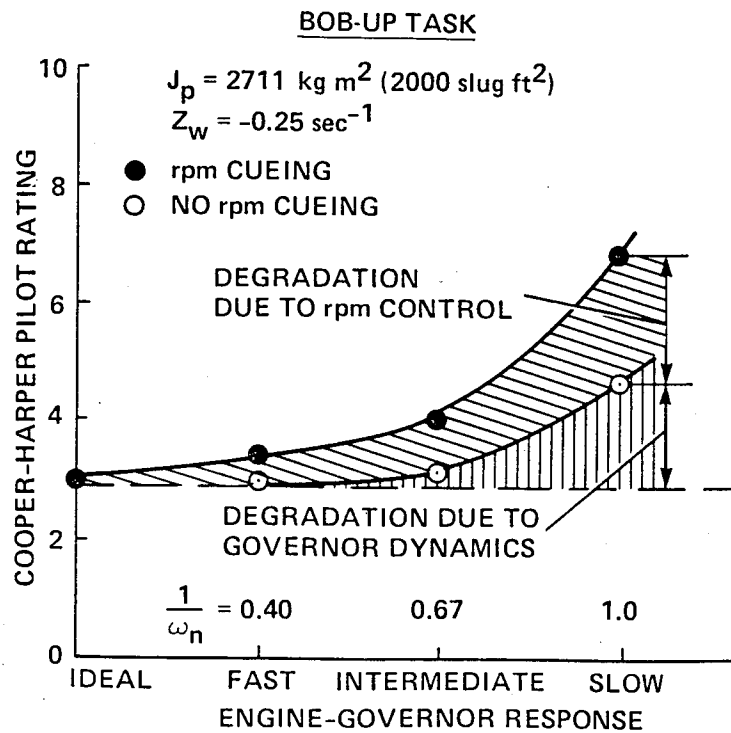


Figure 1(3.3.10.4). Effects of rpm Control  
(from Reference 110)

a. Statement of Requirement

3.3.11 Position Hold. When Position Hold is required by Table 1(3.2), the rotorcraft shall be capable of automatically holding its position with respect to a ground fixed or shipboard hover reference. The rotorcraft shall maintain its position within a 3 m (10 ft) diameter circle during a 360 degree turn in a steady wind of up to 35 knots (18 m/s). The 360 degree turn shall be accomplished by the use of the directional controller with the other controls free, and shall be completed in less than 10 seconds if "aggressive" maneuvering is required in Table 1(3.3), 30 seconds if only "moderate" maneuvering is required and 45 seconds for "limited" maneuvering. The pitch and roll attitude shall not exceed  $\pm 18$  degrees at any point in the 360 degree turn. The pitch and roll attitude response to pitch and roll controller inputs shall meet the requirements of Paragraph 3.3.2 with the Position Hold system engaged. There shall be a clear annunciation to the pilot indicating status of the Position Hold function.

b. Rationale for Requirement

There is considerable evidence from helicopter experience to indicate that position hold is necessary for two basic purposes: 1) when the tasks and visual environments are so severe that the pilot cannot cope, even when fully attended, and 2) to allow the pilot to perform side tasks requiring him to remove his hands from the controls while in hover. The former was demonstrated in Reference 13, where the task was to get a V/STOL aboard a DD963 in Sea State 5, and in the UCE experiments discussed in support of Paragraph 3.2.2, and the latter was demonstrated in the Army combat simulations of Reference 155.

The required accuracy for the hover hold was derived from an LHX Letter of Agreement (LOA) maneuver. That maneuver consists of a 180 deg hover turn in less than 5 sec without deviating more than 5 ft in horizontal position with a wind of 35 kts. This has been extended to the current limits. Government and industry representatives concur that the 3 m diameter circle is adequate to perform the proposed missions and is achievable.

The requirement assures that the integrity of the stick-to-attitude response is maintained with the position hold system on. Such a requirement may seem like a contradiction in terms, since it is not possible to hold position and maintain the integrity of the stick-to-attitude response simultaneously. The intent is to have the position hold designed so that the pilot can override it. Such a feature would allow the pilot to make minor adjustments, and tends to remove the feeling that the system is "fighting" the pilot.

The last sentence of the requirement specifically states that some cockpit annunciation be provided to the pilot. It is imperative that the pilot be able to quickly tell when the system is engaged, and, even more importantly, if it has been inadvertently disengaged.

c. Supporting Data

The single-pilot combat simulation of Reference 155 (described in detail in the discussion for Paragraph 3.2.2) investigated the importance of position hold for single-pilot air combat, ground attack, and hovering tasks, all initiated from a hover. No real differences in pilot opinion were found for the air combat and ground attack tasks, but position hold was necessary for the hover tasks that required the pilot to operate aircraft and communications systems. As in the Reference 4 simulation, position hold was active only when stick forces were low (1/2 lb or less).

d. Guidance for Application

None.

e. Related Previous Requirements

None.



a. Statement of Requirement

3.3.12 Translational Rate Response-Type. The translational rate response to step cockpit pitch (roll) control position or force inputs shall have a qualitative first order appearance, and shall have an equivalent rise time,  $T_{x_{eq}}(\dot{T}_{y_{eq}})$ , no less than 2.5 sec and no greater than 5 sec. The parameter  $T_{x_{eq}}(\dot{T}_{y_{eq}})$  is defined in Figure 10a(3.3). For Level 1, the following requirements/recommendations apply:

- a. The pitch and roll attitudes shall not exhibit objectionable overshoots in response to a step cockpit controller input.
- b. Zero cockpit control force and deflection shall correspond to zero translational rate with respect to fixed objects, or to the landing point on a moving ship.
- c. There shall be no noticeable overshoots in the response of translational rate to control inputs. The gradient of translational rate with control input shall be smooth and continuous.

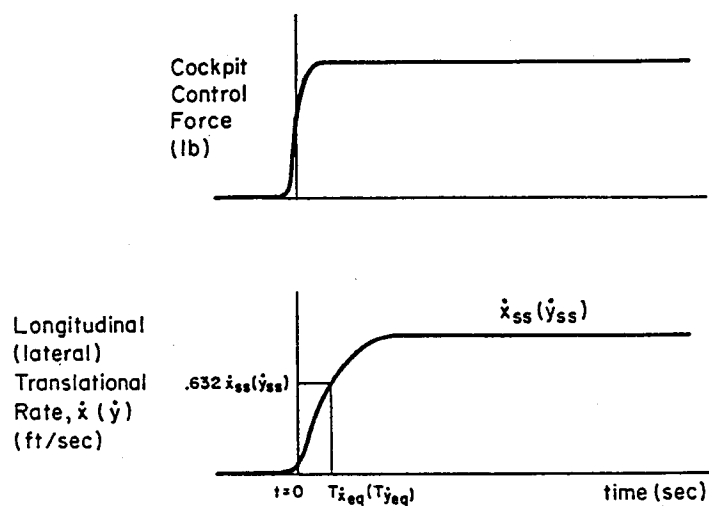
In addition, for centerstick controllers, it is recommended that the variation in translational rate with control deflection shall lie within the limits of Figure 10b(3.3). For sidestick controllers, it is recommended that the variation in translational rate with control force shall lie within the limits of Figure 10c(3.3).

b. Rationale for Requirement

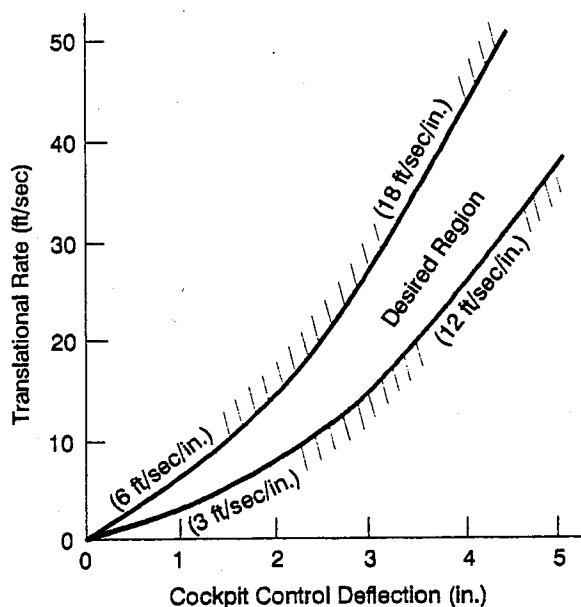
Translational Rate Command (TRC) is specified in Table 1(3.2) for certain Mission-Task-Elements in conditions of degraded Usable Cue Environment. The requirements in this paragraph set response limits on TRC systems.

There have been a number of flight and ground-based simulation studies of TRC Response-Types, many of which have resulted in conflicting opinion regarding what constitutes an acceptable response. The results of several such experiments are described in Supporting Data. The limits specified in this paragraph are those that are known to produce acceptable flying qualities.

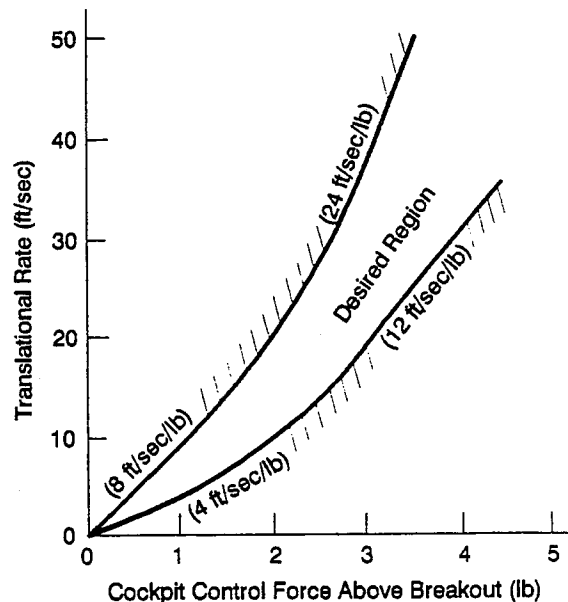
There are numerous competing criteria for specifying the short-term response of translational rate to control inputs. The simplest of these assumes a first-order-appearing response and sets limits on an equivalent rise time parameter. More exotic criteria, such as frequency-response envelopes (e.g., Reference 179) and more complex time response parameters (Reference 59), were briefly considered. For any rotorcraft for which translational rate is developed by changing aircraft pitch (or roll) attitude, analysis suggests that the simple rise-time approach



a) Definition of Equivalent Rise Time,  $T_{\dot{x}_{eq}} (T_{\dot{y}_{eq}})$



b) Recommended Control/Response for Centerstick Controllers



c) Recommended Control/Response for Sidestick Controllers

Figure 10(3.3). Requirements and Recommendations for Longitudinal (Lateral) Translational Rate Response-Types -- Hover and Low Speed

used here is sufficient, especially if the response of translational rate shows no dominant nonlinearities or higher order effects. A range on rise time is defined to regulate against overly abrupt attitude responses (for very short rise times), and overly sluggish translational rate responses (for very long rise times). An additional requirement that the attitude response does not exhibit excessive overshoots is included to assure that the integrating of the attitude response to cockpit control inputs has not been compromised; the results of Reference 4 clearly indicate a preference for TRC to look as Attitude-like as possible. The combined effect of these requirements is to assure that the attitude response of any TRC Response-Type looks very similar to an Attitude Response-Type in the short term.

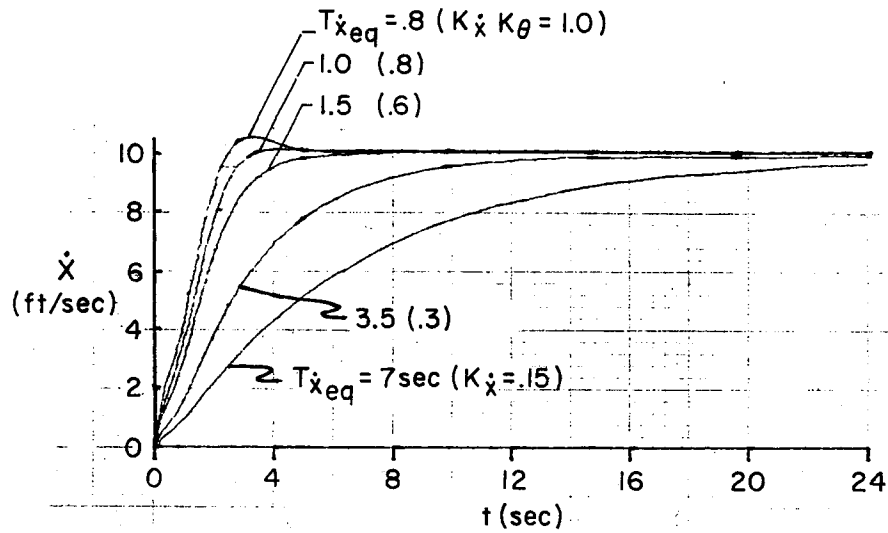
The plot in Figure 1 indicates the generic effect of increasing the outer loop translational rate gain on the attitude response. Figure 1 shows that as long as longitudinal acceleration is achieved through attitude changes, a decrease in the velocity time constant can only be achieved with rapid attitude overshoots. As will be shown below, such overshoots result in unacceptable attitude bobbling and inadequate consonance between the controller and attitude. Part "a" of the requirement is aimed at disallowing such unacceptable attitude overshoot characteristics. The lower limit on the velocity time constants is also driven by this factor.

A primary value of TRC is that it provides a tactile cue that relates cockpit control position and force to ground speed. This cue would not be available if zero control input did not correspond to zero speed, with respect to either the ground, or a reference point on a moving ship for sea-based operations. Part "b" of the requirement is intended to provide the necessary connection between linear rate and control input. It should be noted that a trim system could be provided as long as the pilot has a method to quickly find the null position. A more practical solution may be to switch to a more appropriate Response-Type for constant-velocity maneuvers.

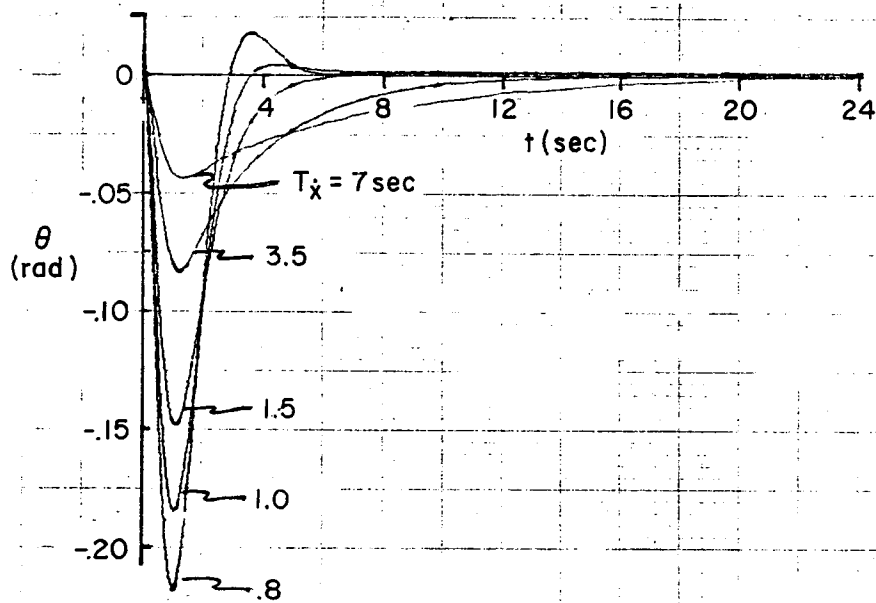
The translational rate overshoot requirement (Part "c") qualitatively disallows "noticeable" overshoots and does not provide any quantitative guidance as such data is not available. Since TRC represents a high level of augmentation, it is unlikely that a system with significant velocity overshoots would be proposed. The qualitative requirement is provided as a reminder that such characteristics would be unacceptable.

The requirement includes a recommendation that input shaping be employed, similar to most lateral augmentation systems on current fighter aircraft. Such input shaping is highly desirable to allow precision tracking around zero displacement without sacrificing the ability to command large rates. The shaping is effective only if precision tracking is accomplished about zero stick deflection, which is insured by Part "b". The shaping boundaries given in the specification are recommended only, since there is insufficient data to determine these limits. Presentation of recommended boundaries in the specification (as opposed

Note:  $T_{\dot{x}_{eq}}$  is obtained by assuming a first order form equation.  
 $T_{\dot{x}_{eq}} = t$  when  $X$  is 63% of steady state



a) Path Response



b) Pitch Attitude Response

Figure 1(3.3.12). Time Responses to a 1 in. Step Longitudinal Control Input

to the background document) is a break from tradition, and was done to emphasize the need for such input shaping with a TRC Response-Type.

c. Supporting Data

The primary source of data for this requirement is the moving-base simulation documented in Reference 4. Some of the results from that simulation are also presented in the Supporting Data for Paragraph 3.3.2.1. A portion of the simulation was devoted to parametrically varying the response characteristics of a TRC Response-Type to determine what were the fundamental requirements for shipboard landings. Additional data to be discussed here come from the X-22A flight experiments of Reference 59, and the moving-base simulations of References 14 and 48. Fixed-base simulation studies, such as those of Reference 179, were of minimal value because of the critical importance of the short-term attitude response that dominated pilot comments in the Reference 4 simulation. This concern for attitude abruptness is not evident in fixed base simulations because of the missing rate and acceleration cues.

The Reference 4 moving-base simulation used a model of a V/STOL to investigate handling qualities requirements for approach and landing on a moving ship. The NASA Ames Research Center's Vertical Motion Simulator (VMS) was used with computer-generated imagery of the ship in Sea State 3 conditions. Two pilots participated in the experiment, in which the short-term response dynamics and control/response input shaping were varied. The pilot comments reflected a strong objection to the short-term response of most of the configurations tested even though the performance was in the desirable range. This dilemma was most apparent for one case, when the pilot commented that "I definitely would not allow this in the spec" and assigned an HQR of 3. At this point, the pilots were instructed to pay careful attention to the decision tree in the Cooper-Harper scale (e.g., "Is it satisfactory without improvement?", etc., Figure 1(2.8)), and their ratings changed to Level 2 or 3 (HQRs of 5, 6, or 7). Because of these initial problems with the use of the Cooper-Harper scale, the most useful information comes from the pilot comments and not the pilot ratings.

Both a TRC and a TRC with position hold (TRCPH) were evaluated. The position hold was active only when stick forces were less than 1/2 lb, and since this was seldom true (i.e., the task required the pilots to continuously maneuver), there was no difference between the two Response-Types. Therefore, most evaluations were made with the TRC without position hold. Mechanization of the TRC Response-Type is sketched in Figure 2. Tight inner-loop dynamics were provided through pitch and roll attitude and rate feedbacks. The translational rate feedback gain was set at a fixed value that assured adequate gust rejection properties and a high attitude bandwidth. Variations in response were made by changing the prefilter time constant  $T_f$  and the nonlinear stick shaping (as shown in Figure 3).

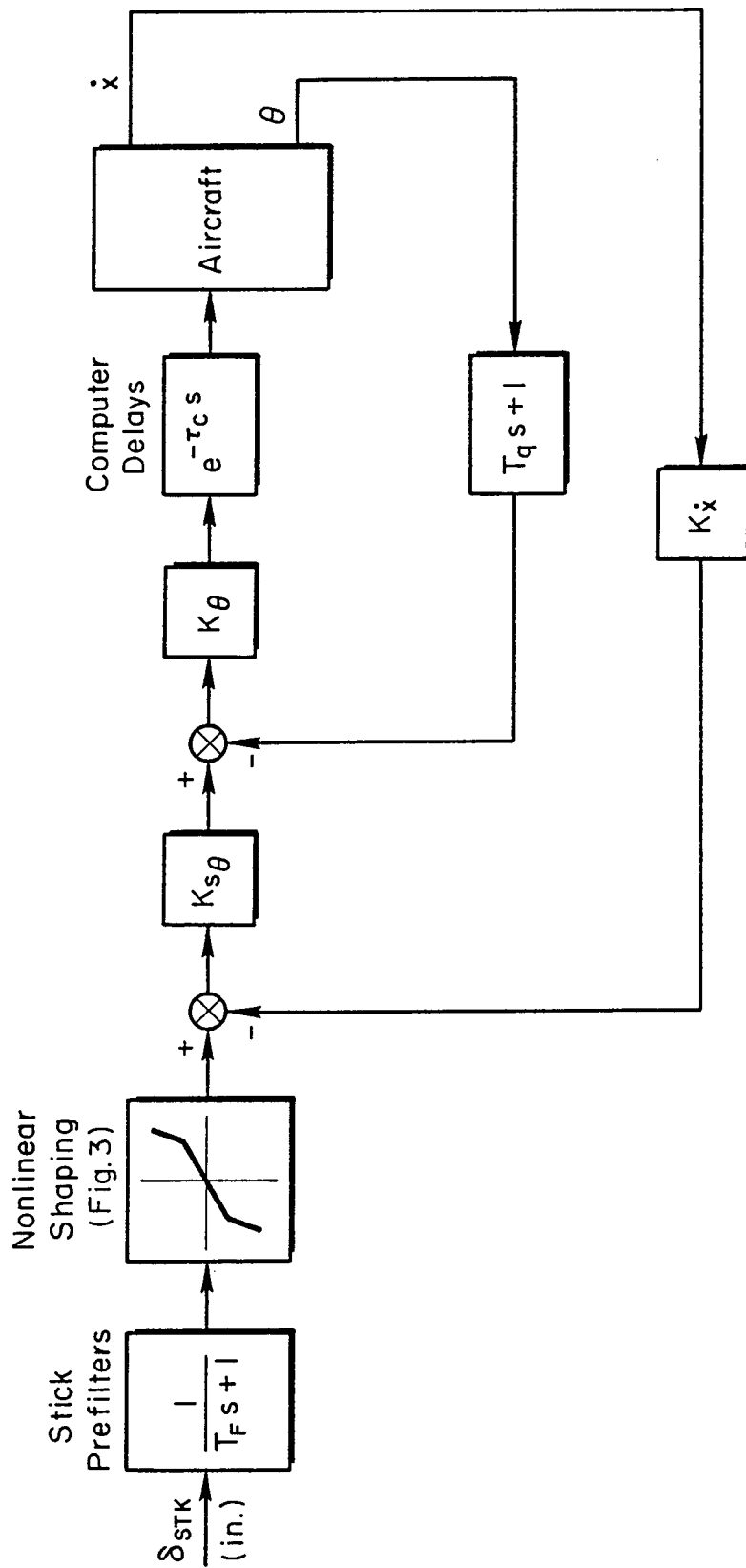


Figure 2(3.3.12). Mechanization of TRC Response-Type for V/STOL Simulation of Reference 4 (Roll Loop was Identical)

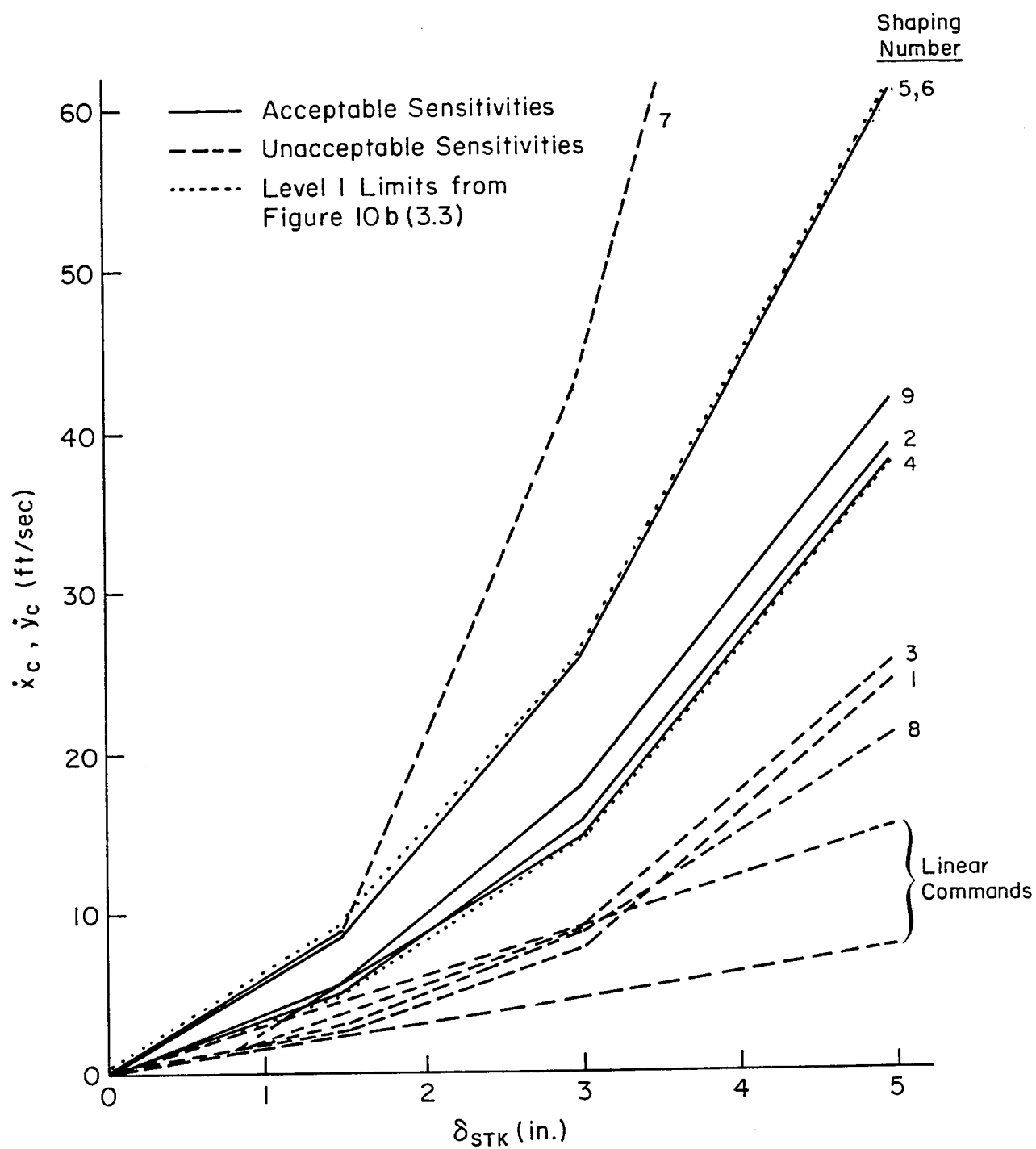


Figure 3(3.3.12). Translational Rate Command Input Shaping from Reference 4 Simulation

Results of the Reference 4 TRC simulation are given in Table 1. The baseline TRC case (Case 56) was evaluated with both linear command shaping and nonlinear shaping. The extensions on case number correspond to stick shaping curves in Figure 3; the initial command slopes are listed in Table 1, along with values of  $T_F$  and HQRs for the approach/stationkeeping/vertical landing tasks. Again, emphasis should be placed on the comments and not the ratings because of the rating problems mentioned above.

With no input prefilter ( $T_F = 0$ ), the comments in Table 1 reflect an excessive attitude abruptness for all of the tested stick shaping cases. This was a result of the very rapid inner-loop dynamics; the attitude and rate feedback gains chosen (Figure 2) were those used for the best ACAH Response-Type in an earlier experiment (see Supporting Data for Paragraph 3.3.2.1). The attitude bandwidth was 4 rad/sec after all computational and visual delays were accounted for. Translational rate feedback did not change the bandwidth appreciably, and the response dynamics should have been ideal according to all existing criteria for TRC (e.g., References 12 and 59). The pilots' objections to the excessive attitude abruptness and overshoots were surprising initially and led to the investigation of prefiltering and command shaping.

Addition of the stick prefilter had the effect of toning down the very abrupt attitude response without compromising the gust rejection properties of the feedback loops (i.e., the bandwidth for the response to turbulence was unchanged). It also rapidly increased the value of  $T_{\dot{x}_{eq}}$  (i.e., the response to stick inputs became increasingly sluggish as  $T_F$  was increased). This is demonstrated in Figure 4. Figure 4a shows the time responses of  $\theta$  and  $\dot{x}$  for a unit command input for the three prefilter values. With no prefilter, there is a very rapid response in translational rate ( $T_{\dot{x}_{eq}} = 1.7$  sec), but since the only way to achieve this rapid response is through an attitude change ( $\ddot{x} \triangleq g\theta$ ) there is a correspondingly rapid pitchdown and immediate washout of the pitch attitude resulting in an "abrupt" response. Increasing the stick prefilter tones down this abruptness, but increases  $T_{\dot{x}_{eq}}$  significantly, and results in a TRC response that looks very similar to an Attitude Response-Type (note especially the pitch attitude time history for  $T_F = 2$  sec in Figure 4).

The pilot comments of Table 1 show that the only favorable responses were those where the highest stick filter was used. This can be seen by summarizing some comments from Table 1, as follows:

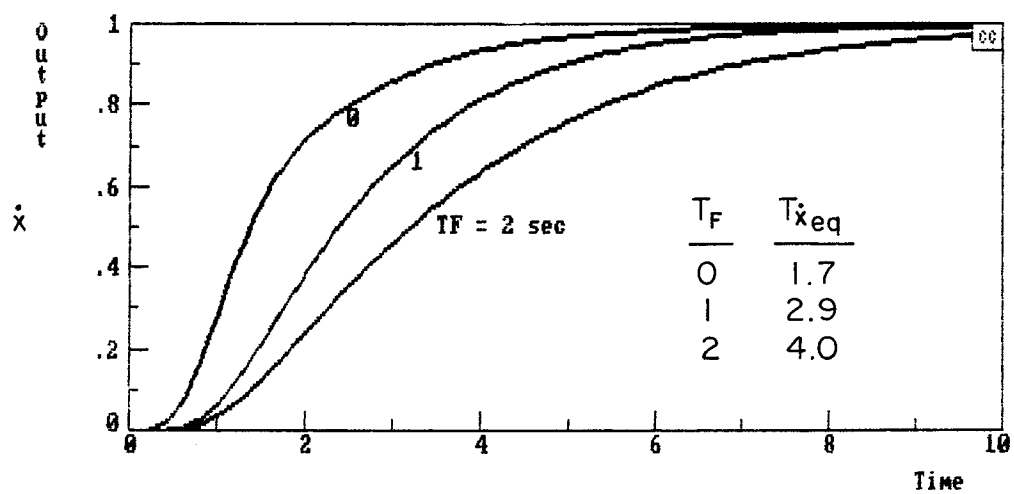
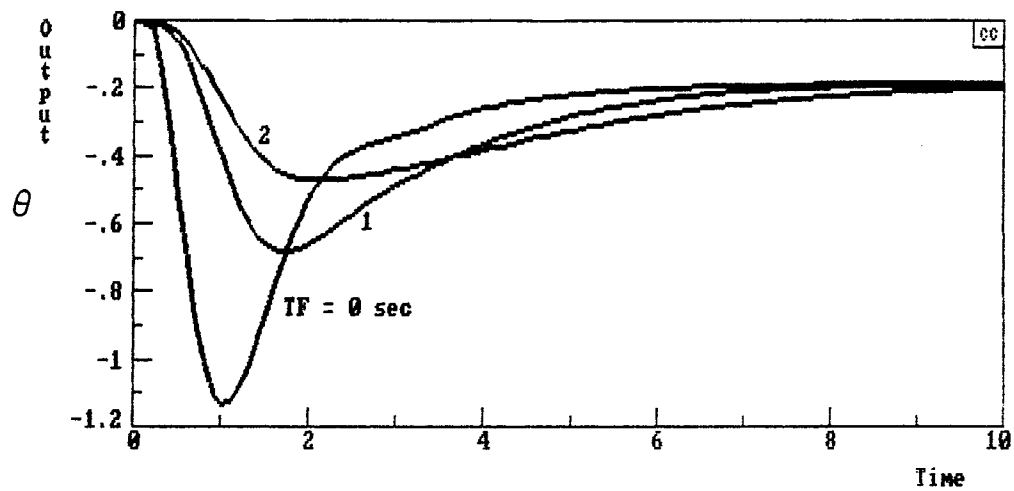
- $T_F=0$ : "Attitudes quite abrupt -- too annoying"  
"Could be PIO prone"  
"Do not feel tied to airplane"  
"Porpoises along -- almost intolerable"
- $T_F=1$ : "Noticeable improvement in abruptness"  
"Degrades precision, tones down attitudes"  
"Attitude response less abrupt, more tolerable"  
"More sluggish translational rate response"



TABLE 1(3.3.12). RESULTS OF TRANSLATIONAL RATE COMMAND STUDY  
(FROM REFERENCE 4)

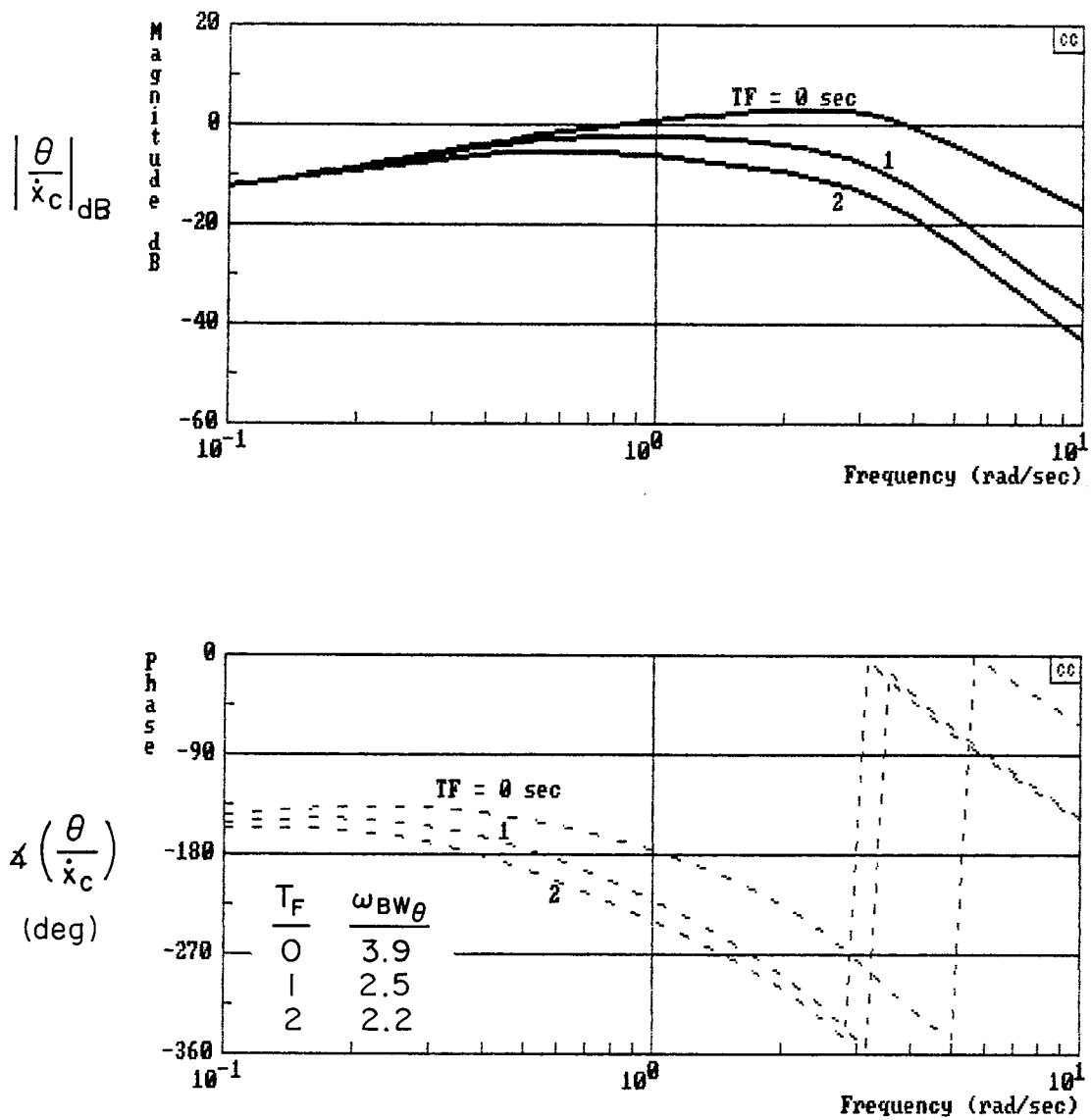
CASE NO.	INITIAL $X_c/\delta_{STK}$ (ft/sec/in.)	$T_F$ (sec)	PILOT RATINGS (PILOT: S,P)	COMMENTS
55	1.5	0	-2-3/2(P)	Very precise, attitude abrupt.
	1.5	1	-2/1.5(P)	Better than last one.
	3.0	0	-3/3(S)	Attitude abrupt; tempering input. Could be PIO prone in more severe conditions.
	3.0	1	-3/3(S)	Noticeable improvement in abruptness.
56	1.5	1	-3/3(P)	Degrades precision, tones down attitudes.
	3.0	1	-2/2(S)	Very good response; attitudes very mild.
56.1	1.7	0	3.5/3.5/3.5(P)	Can do it all but do not like the way it flies.
56.2	3.5	0	4/4/4(P)	Need limits on attitude. Do not feel tied to airplane.
		0	3/3/3(S)	Do not have enough control over attitudes at touchdown. Must be careful when removing inputs. Desired performance was attained.
56.3	1.7	0	3.5/3.5/3.5(P)	Much smoother than 56.2; gradient much too shallow.
56.4	1.7	0	3.5/4/3.5(P)	[Increased max. gradients]. Can see change in gradient; oversensitive in roll.
56.4	1.7	0	3/3/3(S)	Convenient to not worry about attitudes; concentrate on x,y during letdown.
56.5	5.6	0	4/4/4(S)	Like improvement in translational rate, but attitudes too abrupt -- quite annoying.
56.6	5.6	1	3/3/3(S)	Attitude response less abrupt, more tolerable. More sluggish translational rate response compensated for by larger stick inputs.
56.7	5.6	1	3/3/3(S)	Command shaping is objectionable -- stay away from knee of curve. Rating is for operating below knee of the curve. <u>Definitely would not allow this in spec.</u>
56.8	1.9	0	4/3.5/3.5(P)	Not much translational rate -- fly with stick on forward stop initially, $\theta$ is bobbling. No control over attitude.
56.8	1.9	1	4/4/6(S) [PR's for a/c characteristics]	Takes large inputs to go anywhere -- objectionable. Attitudes to get x are quite nice. Not in direct control of attitudes.
56.8	1.9	2	3/3-3.5/3-3.5(P)	Like this much better than $T_F=0$ . Do not have precise control over attitudes or translational rates at touchdown.
56.9	3.9	0	5/5/6(S)	Do not like this one at all. Nonlinearity is highly objectionable, attitudes very sloppy; aircraft abrupt.
56.9	3.9	0	7/3-3.5/3-3.5(P)	Much, much too abrupt. Porpoises along while driving up to ship -- almost intolerable.
56.9	3.9	1	4/4/6(S)	Extensive compensation to minimize attitude excursions.
56.9	3.9	2	3/2/2(P)	Much, much better than $T_F=0$ . Best system; feel attitude is tied to the stick.

- NOTES: 1) Case 55 is TRCPH; Case 56 is TRC.  
2) Extensions on case numbers (e.g., 56.1) refer to nonlinear input shaping (Figure 2).  
Cases without extensions (55,56) had linear shaping.  
3) Initial command gain,  $X_c/\delta_{STK}$ , refers to first segment of input shaping (Figure 2). For linear shaping, gain was constant for full command range.  
4)  $T_F$  is value of first-order prefilter time constant (Figure 1).  
5) All cases were symmetric in pitch and roll.  
6) Pilot ratings are: approach to LPH/stationkeeping/landing. For early evaluations, first task was not performed.  
7) Comments are edited from pilot commentary during evaluations.



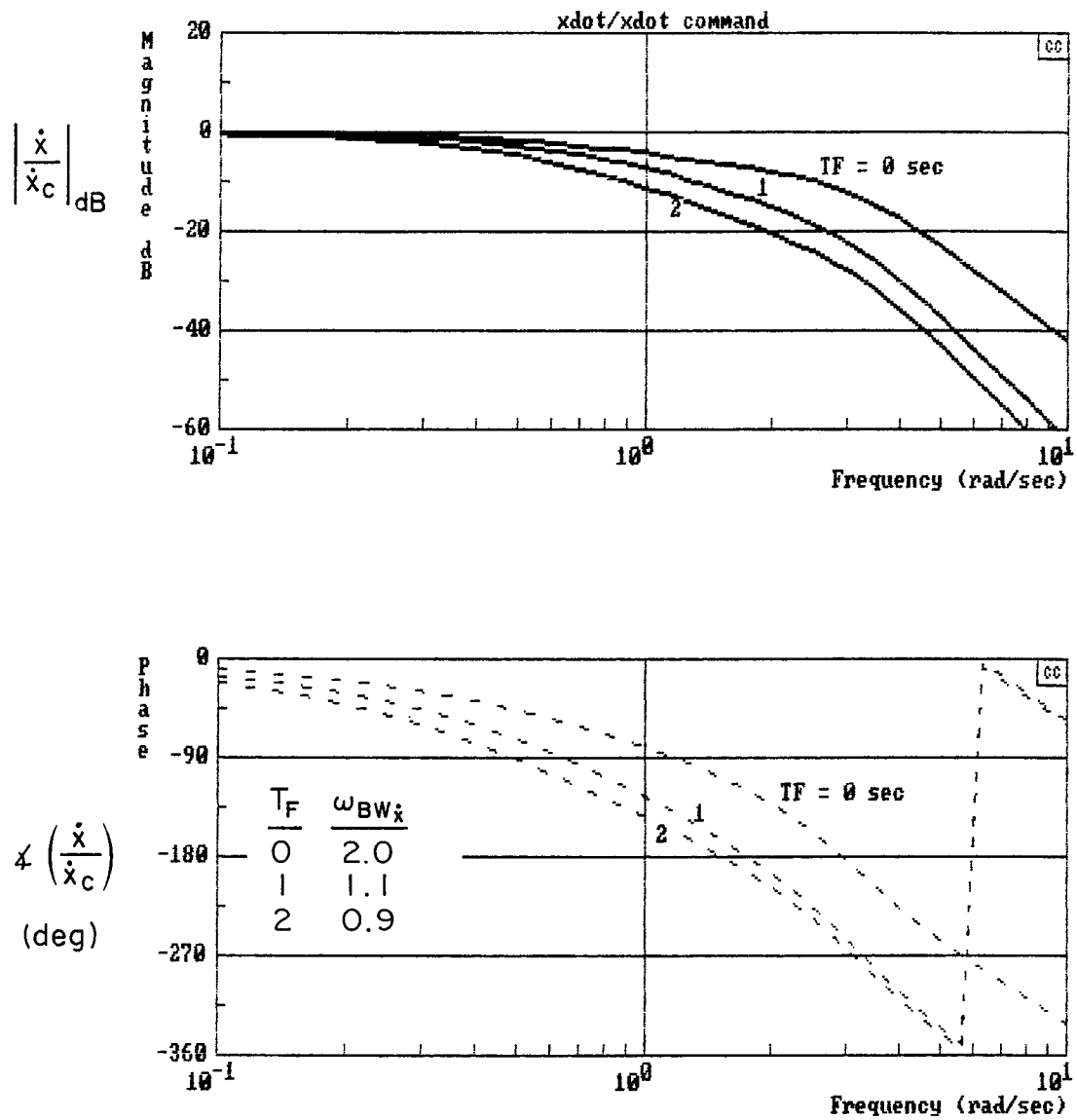
a) Time Responses to a Unit  $\delta_{STK}$  Step Input

Figure 4(3.3.12). Effect of Prefilter Time Constant on Response of TRC System of Reference 4



*b) Frequency Response of Pitch Attitude*

Figure 4(3.3.12). (Continued)



c) Frequency Response of Translational Rate

Figure 4(3.3.12). (Concluded)

- $T_F=2$ : "Like this much better"  
"Best system; feel attitude is tied to the stick".

This attitude abruptness has not always been encountered in other experiments (discussed more below), but the evidence here is sufficiently strong that it has been effectively outlawed in the specification by placing a limit on the lower value of  $T_{x_{eq}}^*$  at 2.5 sec. The data of Figure 4 and Table 1 suggest that 2.9 sec is close to the tolerable limit, while 1.7 sec is much too short. Without further data, it has been assumed here that it would be possible to devise an acceptable TRC Response-Type with  $T_{x_{eq}}^*$  as low as 2.5 sec, as long as the command/response shaping is properly mechanized.

The preference for longer rise times in the Reference 4 simulation meant that the pilot could adopt one of two strategies for gross maneuvering: 1) apply a moderate control input and wait for the translational rate to build up, or 2) overdrive the aircraft to achieve the desired translational rate, then back off on the input. The latter strategy is quite similar to that adopted for Attitude Response-Types, and was the preferred strategy in the simulation. It was found that a linear command gain was inadequate for both gross maneuvering and fine closed-loop control, that is, a gain sufficiently high to achieve an acceptably high translational rate would result in excessive sensitivity for landing, while a gain that was suitably low for landing would not provide enough translational rate with full stick input. The solution was to mechanize a nonlinear command shaping network.

For the Reference 4 simulation, a series of three-slope command schedules were evaluated, as sketched in Figure 3. The solid lines in Figure 3 indicate those command gains that received generally favorable comments (curves 2, 5, 6, and 9), while the dashed gains were not well received (curves 1, 3, and 8). The first slope in curve 4 was too low, and the higher slopes of curve 7 were much too high, and these portions are also dashed in Figure 3. Based on these data, the recommended command gain boundaries of Figure 10b(3.3) were drawn (indicated by dotted lines in Figure 3). There is not enough information to determine where the Level 2 limits might be.

A piloted simulation of the AV-8A Harrier with a TRC Response-Type (also using the NASA Ames VMS, Reference 48) had similar results. This system used velocity command in pitch, roll, and heave, and the pitch and roll command gain was 7.06 ft/sec/in. (linear command), with rise times of  $T_{x_{eq}}^* \approx 2$  sec,  $T_{y_{eq}}^* \approx 1.4$  sec. The model was flown in a shipboard landing task by one pilot in three Sea States with various HUD designs. In Sea State 0, and with no HUD, this TRC received an HQR of 2-1/2 -- the best rating of all the Response-Types evaluated (basic Harrier, Rate, Attitude, and TRC). In Sea State 4, however, the HQR jumped to a 5-1/2, and in Sea State 6, it became a solid 10 -- the worst ratings of any Response-Type in these Sea States. According to Reference 48, this Response-Type "resulted in a high workload in high sea states because the pilot reverted to attempting to control pitch

attitude.... This system tended to disorient the pilot.... [Emphasis added.]" For comparison, the Attitude Response-Type received HQRs of 3, 4-1/2, and 6-1/2 for the three Sea States, suggesting that the pilot was better able to adapt to the basic Attitude Response-Type in the more severe conditions. This is further support for demanding that TRC Response-Types behave in the short term like Attitude Response-Types, especially since TRC is specifically required in conditions of poor UCE, where atmospheric turbulence and winds may be the worst. The TRC of Reference 48 does not meet this specification (the rise times are too short and response gains too high), and this system would definitely not be acceptable in turbulent conditions.

The in-flight simulation of Reference 59, using the U.S. Navy variable-stability X-22A V/STOL, represents the only flight data available for comparison. The rise times and response gains for TRC were varied over a wide range, though a linear command gain was used, and there were very few Level 1 configurations; of 43 separate cases flown, only four received Level 1 average ratings -- and three of these were flown only once. Generally, short rise times and moderate command gains were preferred, and based on the Reference 4 simulation results, this suggests the pilots were searching for a compromise between excessive attitude abruptness and sluggish translational rate.

The forms of the evaluation tasks greatly influenced the results of the Reference 59 flight tests. The primary task consisted of tracking a HUD-displayed image of a landing pad: the pad was driven with a series of discrete position changes, corresponding to approximately 20-ft step-like jumps in pad position (first in x and y, then in x and z) with "rest periods" of 20 to 30 sec to allow the pilot to maneuver back to the pad. The pilots were required to maintain position over the pad in the presence of artificially added random turbulence. At the end of this sequence, a simulated vertical landing was performed. Actual landings could not be performed due to safety constraints on the X-22A, and the tasks took place at an altitude of 50-85 ft. The outside visual scene provided the attitude reference. A limited number of runs were performed with a ground-referenced task, requiring the pilot to rapidly maneuver between edges of a 75-ft-wide taxiway at an altitude of approximately 50 ft.

The HUD task was step-like, requiring large initial inputs to null position errors and only small inputs to maintain position in the presence of turbulence. A limited analysis of the pilot control techniques for this task (Reference 180) suggests that the pilot inputs were essentially discrete steps, with very little closed-loop control. In such a control strategy, it is to be expected that the problems with attitude abruptness and bobbling experienced in the Reference 4 and 48 experiments would not be as extreme, or as noticeable. In addition, the time histories of stick activity from the Reference 59 tests (published in Reference 180) indicate only small stick inputs (on the order of 1 to 2 inches). By contrast, it is likely that for the larger-displacement ground-reference task (involving 75-ft, as opposed to 20-ft, position changes), the pilots used considerably larger inputs. Although the

ground-reference task was flown only a few times, there is some evidence that the pilots preferred a higher command gain for this task -- as would be expected.

Figure 5 shows several of the command gains from the Reference 59 experiment compared with the Figure 10b(3.3) limits. These four gains were flown in both the HUD tracking and ground-reference tasks. Based on the discussion above, we have assumed that the HUD task used less than about 2 inches of stick, while the ground reference task required considerably higher inputs. In one case (with a gain of 4.7 ft/sec/in.), the pilot assigned an HQR of 3 for the HUD task and ratings of 7 and 7 for two separate runs in the ground reference task. For a hover maneuver only (in the presence of natural winds and turbulence), the pilot assigned a 3 -- identical to the HUD rating. In Figure 5, this has been interpreted as indicating that 4.7 ft/sec/in. is adequate for small inputs, but too low for larger inputs.

For a command gain of 5.8 ft/sec/in., similar ratings were given in the HUD task (3, 4, 4, and 4.5) and the ground reference task (4). The pilot comments, however, indicate a difference in the perception of the forces: for the HUD task, the forces "Felt good -- initial and final the same.... Comfortable pitch and roll," while for the ground reference task, the forces were "Comfortable initially but then too high in trying to keep a translation going." A case with a gain of 7.6 ft/sec/in. received HQRs of 4 and 4 in the HUD task, with a comment that "Abruptness of attitude response made me shy about aggressive corrections;" in the ground reference task, the HQR was a 2, and the forces were "comfortable, initial and final." Finally, with 11.5 ft/sec/in., HQRs were 3 and 5 for the HUD task and 4 for the ground reference task with comments about abruptness in both cases: "Airplane abrupt, really jittery [HUD].... I was a little bit careful with initial inputs to avoid abruptness [ground reference]...."

There are other moving-base simulation data that have been used in the past for criteria development (see, for example, Reference 12). These data, however, were generated using camera-model visual image generation, typically requiring rapid (but not very precise) translations, with no winds or turbulence. As a result, these simulation data generally specify quite large command gains (far above those that were found to be acceptable in the Reference 4 simulation or the Reference 59 flight test) and low time constants. These tasks are sufficiently different from those of either Reference 4 or Reference 59 that the data are not considered to be relevant for this specification.

The ADOCS flight control system development simulations of Reference 14 provide some information for limited-displacement sidestick controllers. Although these simulations did not address control command/response shaping, the command gains were modified several times throughout the program. There were three phases in the simulation: Phase 1 used the Boeing Vertol moving-base simulator, and Phases 2A and 2B used the NASA Ames VMS. Both TRC (designated LV/LV in Reference 14) and TRC with position hold (designated LV/PH) were evaluated for a

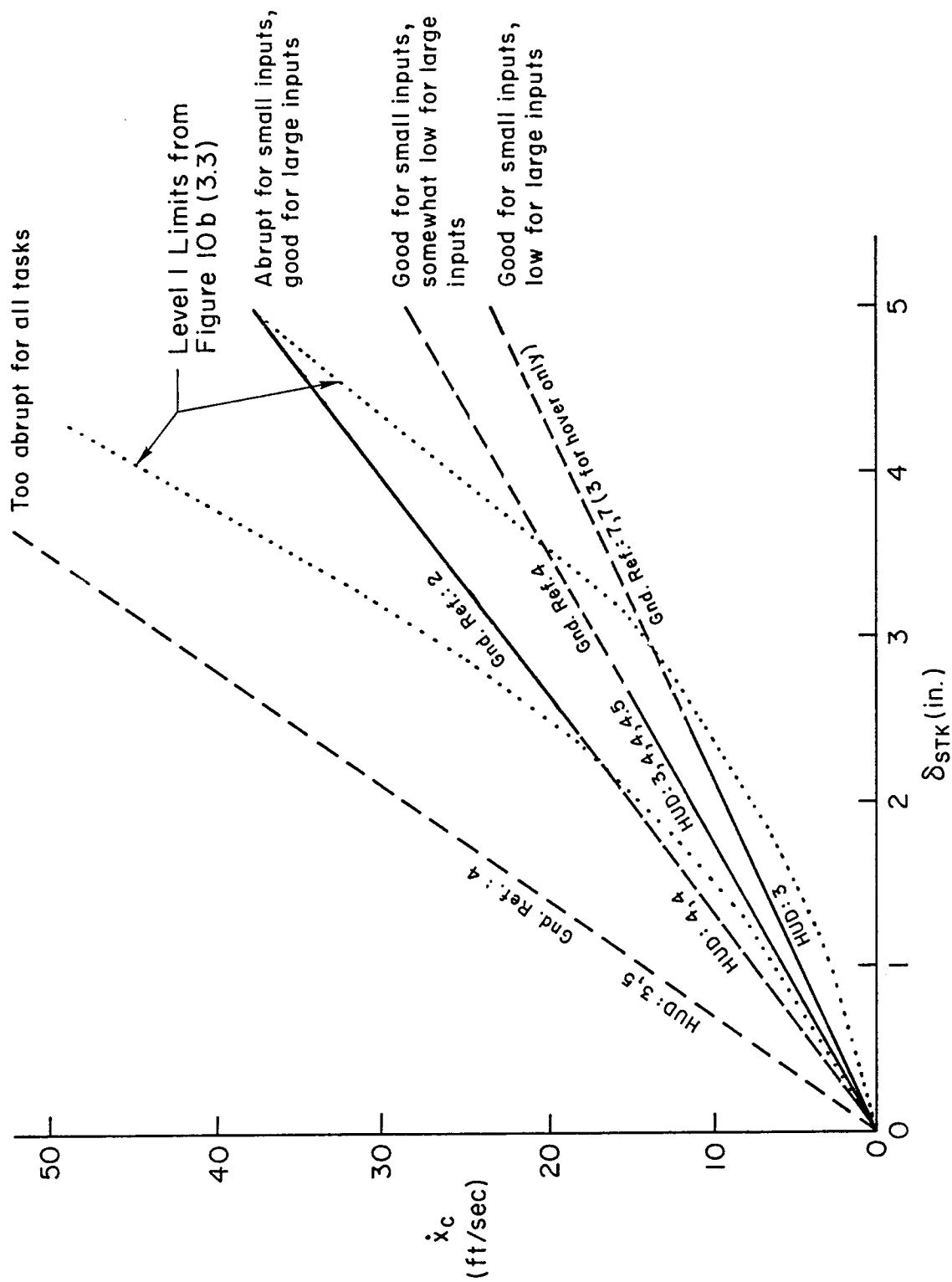


Figure 5(3.3.12). Command Gains Evaluated in HUD Tracking and Ground-Reference Tasks in V/STOL Flight Experiment (Reference 59)



limited number of runs, primarily for precision hover and bob-up in the presence of winds and turbulence. Almost all of the evaluations were in IMC conditions (using a simulated FLIR image and a helmet-mounted display to provide visual information). Translational Rate Command was the only Response-Type to receive Level 1 average HQRs for these tasks. The equivalent rise times in both pitch and roll met the requirements of this specification ( $T_{x_{eq}}^* = T_{y_{eq}}^* = 4.5 \text{ sec}$ ).

Figure 6 shows the command gains used for the ADOCS simulations. The Phase 1 curves were developed for the Boeing Vertol evaluations, and were satisfactory there. When flown on the VMS, with its greater motion fidelity and better visual cues, the gradients in both pitch and roll were much too sensitive around zero. The Phase 2A curves were then developed, but TRC was not formally evaluated during Phase 2A. When the system was flown during Phase 2B, the pitch gradient was still too high. There is no indication in Reference 14 of the range of control inputs required by the pilots; since the only tasks were hover and bob-up from a hover, the inputs were probably quite small. Therefore, the upper limit of the specification recommendation for sidestick control sensitivity (Figure 10c(3.3)) has been drawn to include the curves near zero force for both pitch and roll. For larger inputs, the boundary is somewhat arbitrarily drawn to reflect the Phase 2A and 2B pitch curves. The lower limit is simply half the slope of the upper limit.

#### d. Guidance for Application

Demonstration of compliance with this requirement consists of a series of step inputs, ranging from barely perceptible to full input (if practical), and the rise time measured as defined in Figure 10a(3.3). The command/response sensitivity can be obtained from these time responses by noting the control deflection and/or force and the steady-state translational rate achieved, and comparing these values with the appropriate limits of Figure 10b(3.3) or 10c(3.3).

The sensitivity limits of Figures 10b(3.3) and 10c(3.3) have several fundamental shortcomings common to any such "envelope" criteria: first, if a control system produces a response that lies just outside one limit for only a portion of the response, does that make the entire system Level 2? And second, if the actual response curve has small "wiggles" in it, but the overall curve still falls within the Level 1 limits, is this necessarily Level 1? In both cases, simulation or flight testing may be required to ascertain the answers. The intent here, however, is to assure that TRC Response-Type exhibits both the gradient required to fall in the Level 1 region, and the general shape of the limits as drawn. These intentions are the purpose for the qualitative requirement that "The gradient of translational rate with control input shall be smooth and continuous." Data reviewed for this requirement show a relatively strong relationship between desirable rise time and sensitivity: aircraft with longer rise times will probably require gradients near the upper boundary, and vice versa.

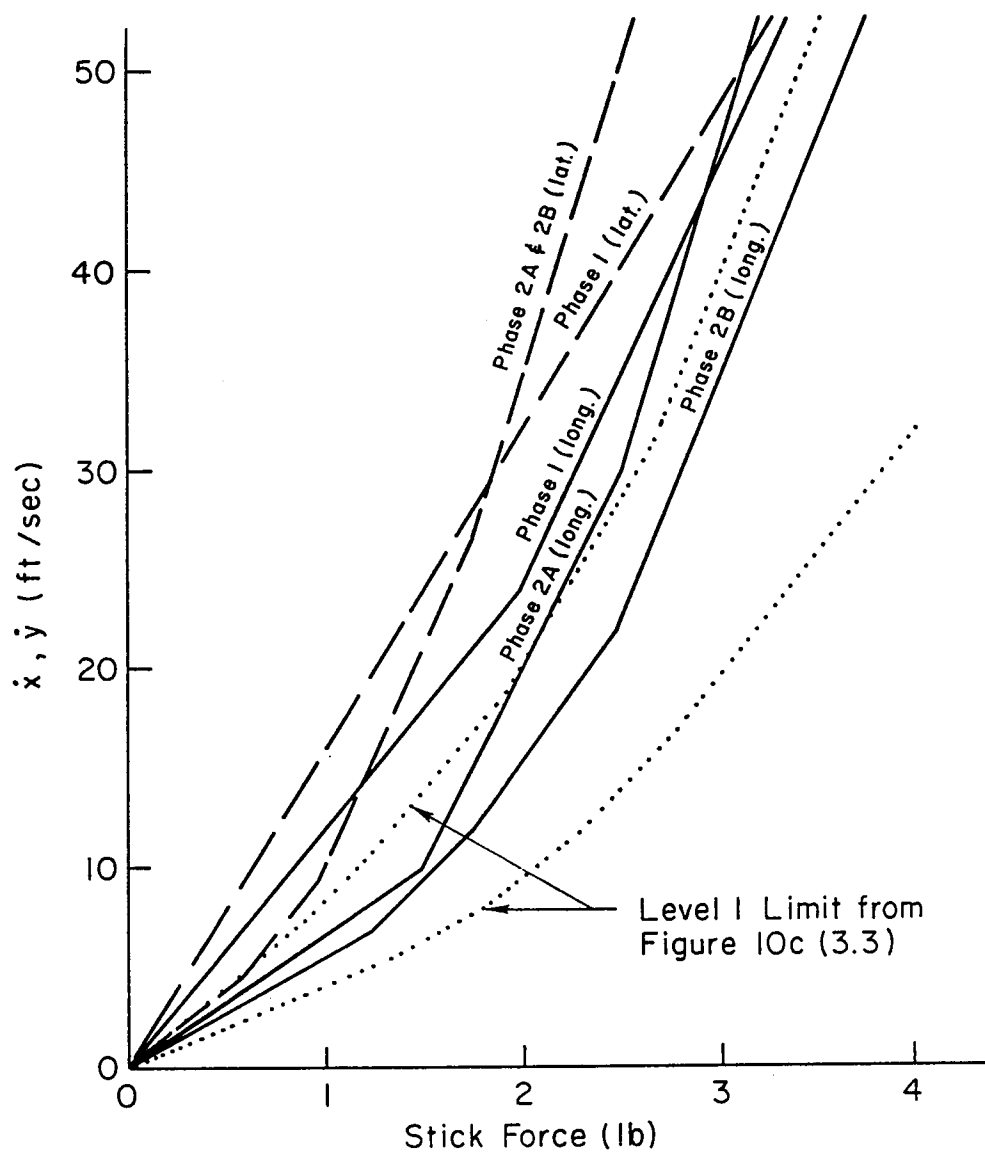


Figure 6(3.3.12). TRC Command Gains from ADOCS Simulation  
(Reference 14)

The slopes of the Level 1 sensitivity boundaries near zero and for large inputs are labeled on Figures 10b(3.3) and 10c(3.3). These labels make application of the criteria easier, especially for displacements or forces that are larger than those shown.

e. Related Previous Requirements

There are no requirements on Translational Rate Response-Types in any previous flying qualities specifications. Several researchers have proposed requirements, in most cases using the parameters applied here (equivalent first-order time constant and steady-state gain). The data supporting these proposed requirements have been based on simulations or flight tests where the control/response gearing was entirely linear. The Reference 4 simulation indicates the importance of a nonlinear gain to provide low sensitivity for small inputs and high control power for large inputs.

Figure 7 illustrates two of the more familiar time response requirements from References 12 and 59. The data upon which these boundaries are based were described in Supporting Data. For the Reference 12 boundary, the primary data came from a simulation on the NASA Ames Flight Simulator for Advanced Aircraft (FSAA). That simulation did not include turbulence, and the primary task involved large translations in position, using points on a terrain board as markers. There was little in the way of precision hovering involved, and the boundary drawn shows a tolerance for very high command gains, as would be required to perform the task. The Reference 59 limits come from the X-22A flight tests and reflect the relatively low gains and rapid responses preferred for small amplitude position changes during the HUD tracking task. These limits were drawn using a portion of the total flight data. In general, there were very few Level 1 points, and the overall data set does not agree with the boundaries very well. The few pilot ratings from the ground-reference task, which involved considerably larger translations and, hence, larger control inputs, tend to agree more with the Reference 12 limit.

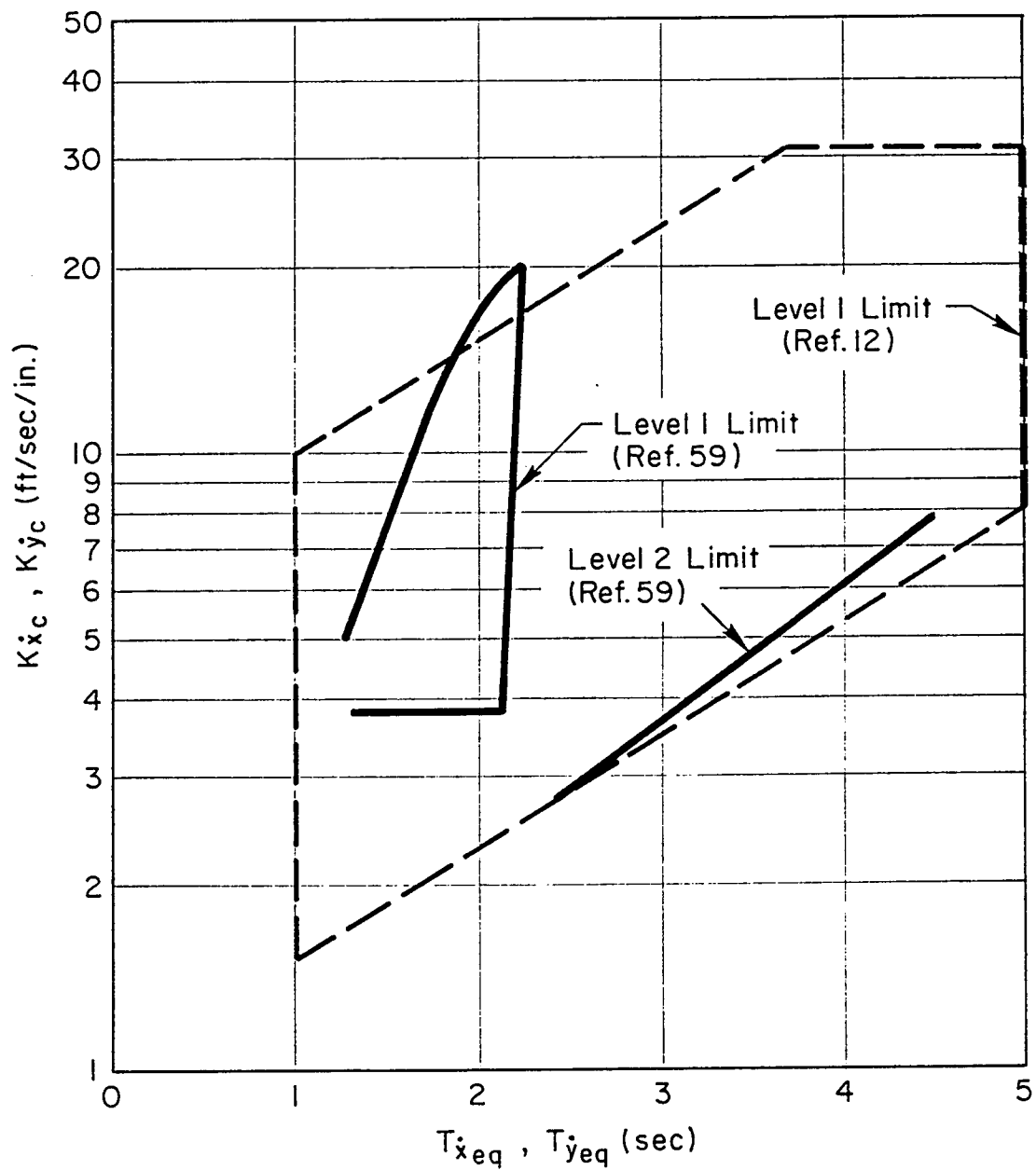


Figure 7(3.3.12). Proposed TRC Response Requirements from References 12 and 59

a. Statement of Requirement

3.4 FORWARD FLIGHT

The Forward Flight requirements apply to those Mission-Task-Elements designated as Forward Flight tasks in Paragraph 3.1.1 and apply over the applicable portions of the Flight Envelopes as defined by Paragraph 2.6.3.

b. Discussion

This paragraph specifies the applicability of the forward flight requirements. As defined in Paragraph 2.6.3, forward flight involves all operations at a ground speed greater than 45 kts.

a. Statement of Requirement

3.4.1 Pitch Attitude Response to Longitudinal Controller

3.4.1.1 Short-Term Response (Bandwidth). The pitch attitude response to longitudinal cockpit control force or position inputs shall meet the limits specified in Figure 1(3.4). The bandwidth ( $\omega_{BW_\theta}$ ) and phase delay ( $\tau_{p_\theta}$ ) parameters are obtained from frequency responses as defined in Figure 2(3.3).

It is desirable to meet this criterion for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish.

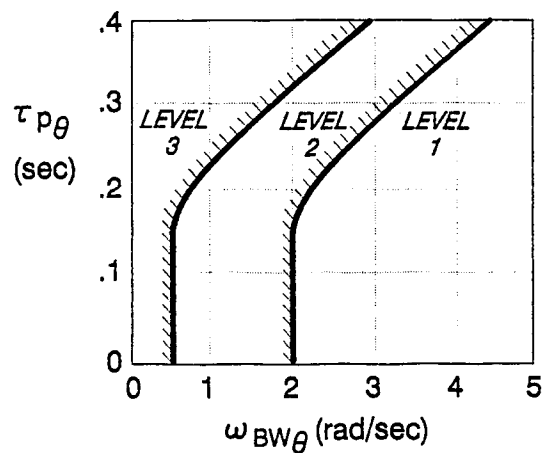
b. Rationale for Requirement

The approach taken in developing the requirements for this paragraph was basically the same as that for the hover and low speed requirements of Paragraph 3.3.2.1. Because the form of the pitch response is similar, it was expected that the form of the criteria would be. As the discussion in Supporting Data shows, the available data support the same limits as well. As is shown in Related Previous Requirements, these limits are considerably more stringent on rotorcraft response than are the MIL-H-8501A requirements (3.2.11 and 3.6.1.2).

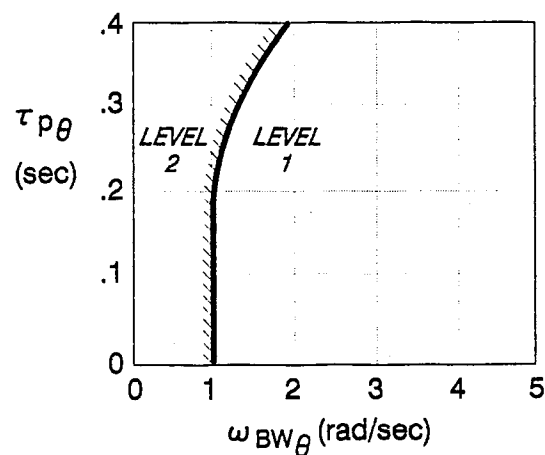
The limits for Air Combat, Figure 1a(3.4), are identical to the Hover and Low Speed limits for Target Acquisition and Tracking, Figure 1a(3.3), and are based on experiments conducted by the British Royal Aeronautical Establishment and by Systems Technology, Inc. The results of these experiments are documented in Supporting Data for this Paragraph.

The Bandwidth frequency limits for All Other MTEs -- VMC and Fully Attended Operations, Figure 1b(3.4), are supported by results of a flight test with a variable-stability CH-46, described in Supporting Data. Since no forward-flight data are available to specify the required variation in phase delay with Bandwidth, the boundary in Figure 1b(3.4) is identical to that in Figure 1c(3.3), which in turn was based on a low-speed flight experiment by the Canadian National Research Council as described in Supporting Data for Paragraph 3.3.2.1.

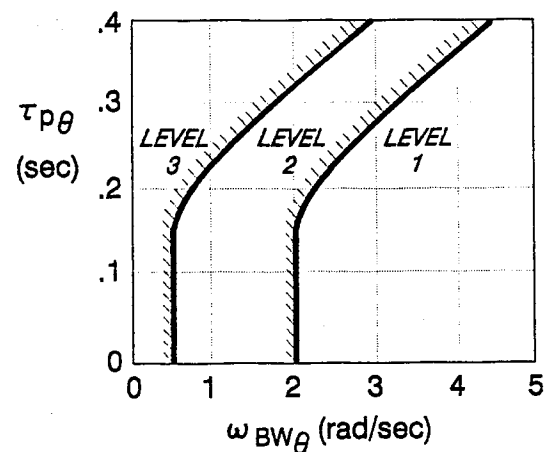
The requirements for All Other MTEs -- IMC and/or Divided Attention Operations, Figure 1c(3.4), are identical to the Air Combat requirements of Figure 1a(3.4). Flight and ground simulation data for instrument operations (constant-speed and decelerating approaches), documented in Supporting Data, indicate that the minimum Bandwidth for Level 1 is about 2 rad/sec, consistent with the Air Combat requirement; the only question for IMC operations was whether the shape of the boundaries with increasing phase delay should be stringent (as for Air Combat) or relaxed (as for VMC). In the absence of any data, it was decided that the more relaxed boundaries, such as in Figure 1b(3.4), simply allowed



a) Air Combat



b) All Other MTEs - VMC and Fully Attended Operations



c) All Other MTEs - IMC and/or Divided Attention Operations

Figure 1(3.4). Requirements for Small-Amplitude Pitch Attitude Changes -- Forward Flight

too much phase delay for precision instrument operations.

c. Supporting Data

Much of the available forward-flight data base was described in some detail in the discussion for Paragraph 3.2.2. The following paragraphs will rely on that discussion for background, with some additional information from the flight experiment of Reference 178 and from a fixed-base simulation.

1. Air Combat (Figure 1a(3.4))

The Air-to-Air Combat requirements of Figure 1a(3.4) are identical to those in the pitch requirement at low speeds (Paragraph 3.3.2.1). These limits are based on two investigations of bandwidth requirements for precision pitch tracking. While this is not a specific Mission-Task-Element as defined by Paragraph 3.2.2, the conditions of the tasks involved closely approximate those found in Air Combat, requiring rapid, precise changes in attitude and hence the results of these experiments are discussed here. The tasks in both experiments were confined to relatively small-angle commands and did not include the combination of large-amplitude target acquisition and small-angle fine tracking that are characteristics of air-to-air combat. Since the interest in this Paragraph is limited to small-angle maneuvering, this is not considered a shortcoming in the data.

The first experiment was an in-flight investigation of pitch attitude capture (or pitch pointing), conducted by the British Royal Air craft Establishment using a Puma flight research helicopter (Reference 178). Response dynamics of the Puma were varied by flying at four airspeeds (60, 80, 100, and 120 kts), with the stability augmentation system on and off, for a total of eight evaluation cases. In addition, the aircraft was flown at forward and aft center-of-gravity positions, SAS on, at 60 kts. The task consisted of following a series of pitch attitude changes using the Puma's attitude indicator. The pitch commands were read aloud by the flight test engineer, commanding a new attitude every three seconds. The pitch attitudes were restricted to a range of +5 to -7.5 deg, in 2.5-deg intervals (i.e., all attitude commands were either +5, +2.5, 0, -2.5, -5, or -7.5 deg). Each evaluation took approximately 90 seconds. Four pilots participated in the study. Dynamic characteristics of the evaluation cases were identified by performing frequency-sweep inputs at each test condition. Full documentation of the vehicle dynamics and HQRs (from four pilots) was provided by the author of Reference 178.

The results of the Reference 178 flight tests are shown in Figure 1. Initial trim airspeed is given in parentheses for each case, and actual handling qualities ratings are noted next to the symbols. The pilots were not allowed to assign half-ratings; the "+" symbols in Figure 1 indicate cases for which the pilots considered assigning worse ratings, and hence may be considered half-ratings for this analysis. The value of  $\omega_{BW}$  for the 120-kt, SAS-off case is an assumption for a



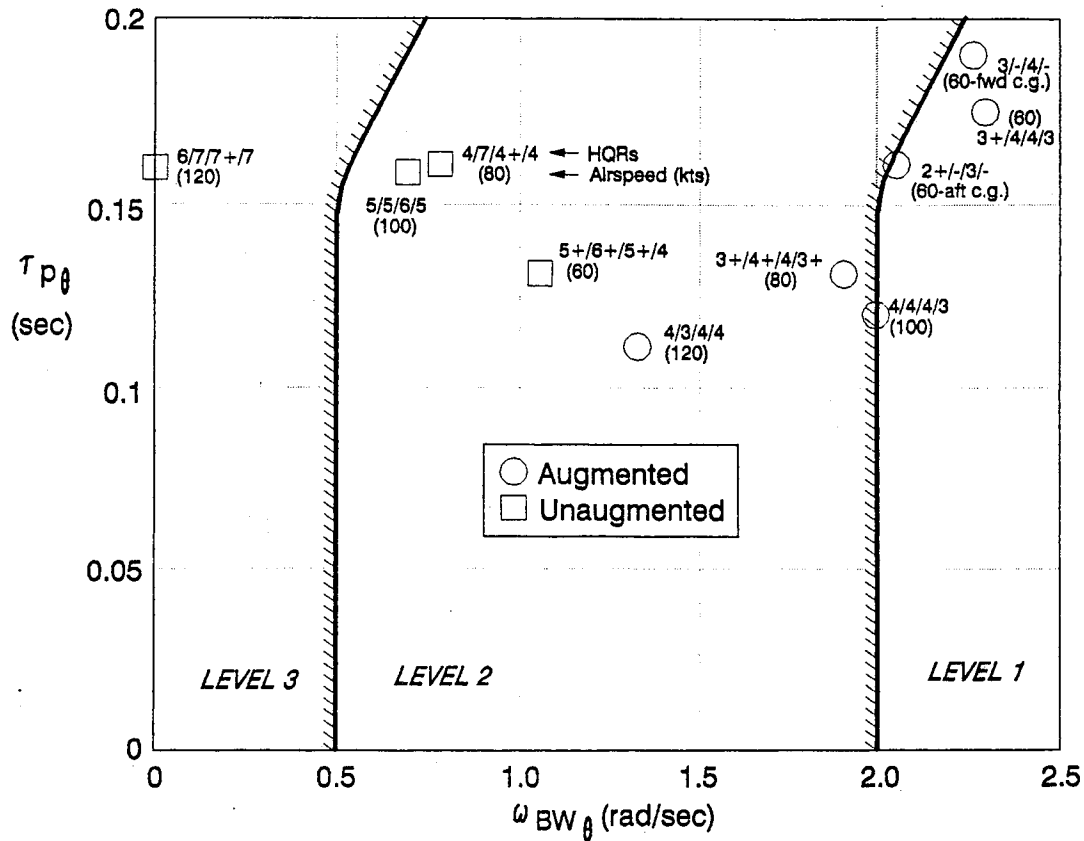


Figure 1(3.4.1.1). Results of In-Flight Pitch Tracking Experiment  
Conducted by RAE (Reference 178; Data Provided by  
Author of Reference 178)

"worst-case" value; it was difficult to obtain good low-frequency data for the Puma at this flight condition. The SAS produced a Rate Command/Attitude Hold Response-Type, with a limited-authority SAS servo; with the SAS off, the Puma is a Rate Response-type.

The Puma was rated as slightly worse than Level 1 at the higher speeds with the SAS on, which is consistent with the criterion boundaries in Figure 1. With the SAS off the ratings are solidly Level 2 or Level 3. In general, the ratings are in agreement with the Figure 1a(3.4) limits.

A fixed-base simulation was conducted at Systems Technology, Inc., in support of the requirements of this Paragraph. The simulation was designed to address three major questions: 1) What is the effect on handling qualities of very high levels of pure time delay (as much as 1/2 second) in combination with high bandwidths? Some such combinations

are Level 1 by the limits of Figure 1a(3.4). 2) Are there any significant differences, in terms of handling qualities, between high pure delays that produce large values of  $\tau_p$ , and cascaded lags that produce similar values of  $\tau_p$ ? If such differences were apparent, it might be necessary to include a separate limit on time delay along with the phase delay limits. 3) Do the limits on  $\omega_{BW}$  versus  $\tau_p$  properly account for high levels of pure time delay? That is, are the curves in Figure 1a(3.4) sufficiently steep to prevent a bad combination from passing the specification?

These questions were tested by devising twelve configurations that were spread over the range of bandwidth and phase delay. The desired variations were most easily obtained by using Attitude Command/Attitude Hold Response-Types. The models were simulated on an analog computer to remove any computational delays, and time delays of varying amounts were added to these systems (it was difficult to obtain pure delays; instead, the desired delays were mechanized using a second-order Pade approximation. This resulted in a small initial response to step inputs in the wrong direction, and although this response was noticeable to the pilots at times, the overall effect on pilot opinion was considered to be minimal.).

The twelve configurations are listed in Table 1. In most cases, the system damping ratio was held constant at a value of 1.0. One case with a relatively low damping ratio (0.35) was included as a check case on the effect of damping ratio. Natural frequencies of 1, 2, 3, 5, and 10 rad/sec were used. Time delay was varied from zero to 0.5 sec; a small amount of delay (0.033 sec) was included on the control cases, simulating actuator effects. The simulation setup was identical to that described in the supporting data for the mid-term response requirements of Paragraph 3.3.2.2 (also documented in Reference 159). The evaluation task consisted of tracking a flight-director-like pitch bar on a simulated attitude indicator. Pitch bar commands were provided by a pseudo-random noise generator. The two evaluation pilots were required to minimize the pitch error during runs of approximately 80 sec; two runs were performed for each configuration before Cooper-Harper pilot ratings were assigned and comments taken.

The cases are plotted in Figure 2, where the average handling qualities ratings are noted next to each symbol. The primary case for evaluating the effects of very high time delay (with  $\omega_n = 5$  rad/sec,  $\tau = 0.5$  sec) has a bandwidth of 2.7 rad/sec and phase delay of 0.3 sec (Table 1). This case was evaluated eight times, receiving an average rating of 4.75 (actual ratings ranged between 4 and 5.5, Table 1). Since this case lies in the Level 2 region on the Figure 1a(3.4) limits (Figure 2), this represents good agreement. A similar configuration, with a higher natural frequency (10 rad/sec) and the same time delay (0.5 sec) was evaluated once, and was rated a 4. This case is in the Level 1 region, near the Level 2 limit (Figure 2).

One case was devised with no time delay, but with a fifth-order characteristic equation, resulting in a high value of  $\tau_p$ . The second-order root had a damping ratio of 0.7 and natural frequency of 5 rad/sec, and three first-order roots at 5 rad/sec were added. This

TABLE 1(3.4.1.1). CONFIGURATIONS EVALUATED  
IN FIXED-BASE SIMULATION

$\theta/\theta_c^*$	$\omega_{BW\theta}$ rad/sec	$\tau_{p\theta}$ (sec)	HANDLING QUALITIES RATINGS		
			Pilot A	Pilot B	Avg
$\frac{1(.033)}{[1,1]}$	2.19	0.025	4	3	3.5
$\frac{1(0.2)}{[1,1]}$	1.62	0.152	—	3	3.0
$\frac{1(0.3)}{[1,1]}$	1.44	0.226	5,4	3,3	3.75
$\frac{4(0.033)}{[1,2]}$	4.04	0.026	4,4	3	3.67
$\frac{4(0.2)}{[1,2]}$	2.61	0.150	5,3	2,2	3.0
$\frac{9(0.4)}{[1,3]}$	2.46	0.275	4	4.5	4.25
$\frac{9(0.4)}{[0.35,3]}$	2.70	0.345	6	6	6.0
$\frac{25(0.033)}{[1,5]}$	8.50	0.025	3	—	3.0
$\frac{25(0.2)}{[1,5]}$	4.48	0.140	2.5,3	3,2	2.63
$\frac{25(0.5)}{[1,5]}$	2.73	0.297	5,5.5,5,4	5,5,4.5,4	4.75
$\frac{3125(0)}{(5)^3 [0.7,5]}$	2.77	0.271	3,4	4,4	3.75
$\frac{100(0.5)}{[1,10]}$	3.43	0.241	—	4	4.0

\* $(\tau) = e^{-\tau s}$ ; time delay approximated by a second order Pade,

$$e^{-\tau s} \doteq \frac{[-.866, 2\sqrt{3/\tau}]}{[.866, 2\sqrt{3/\tau}]}; (a) = (s + a); [\zeta, \omega_n] = [s^2 + 2\zeta\omega_n s + \omega_n^2]$$

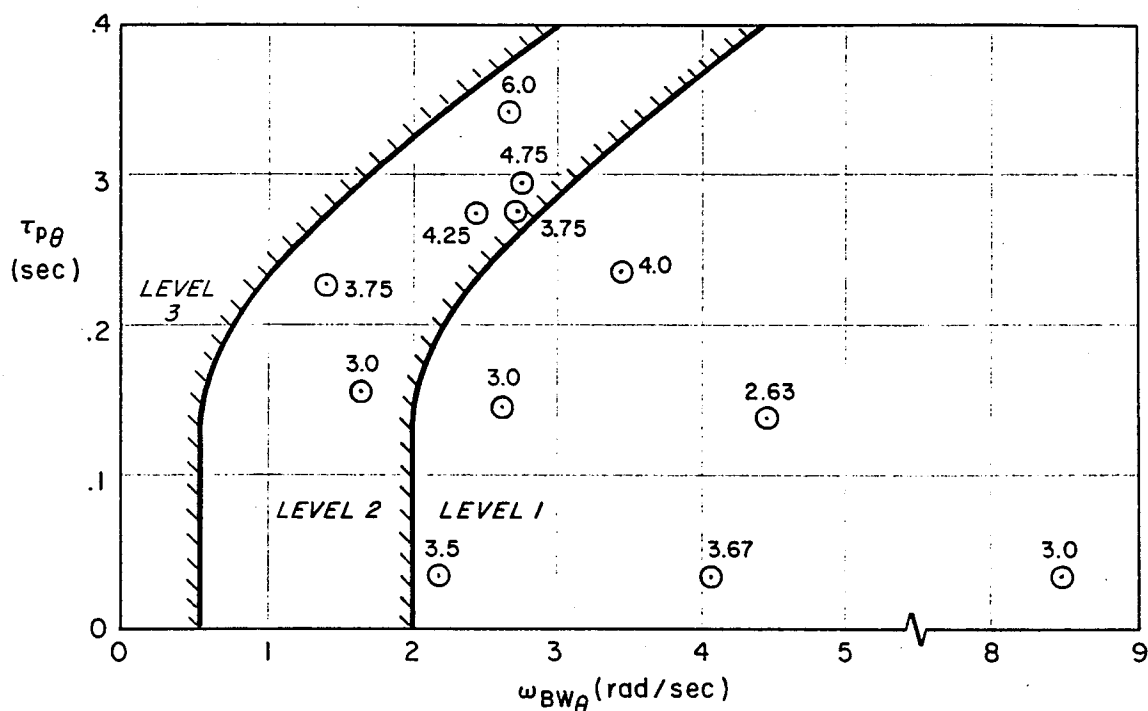
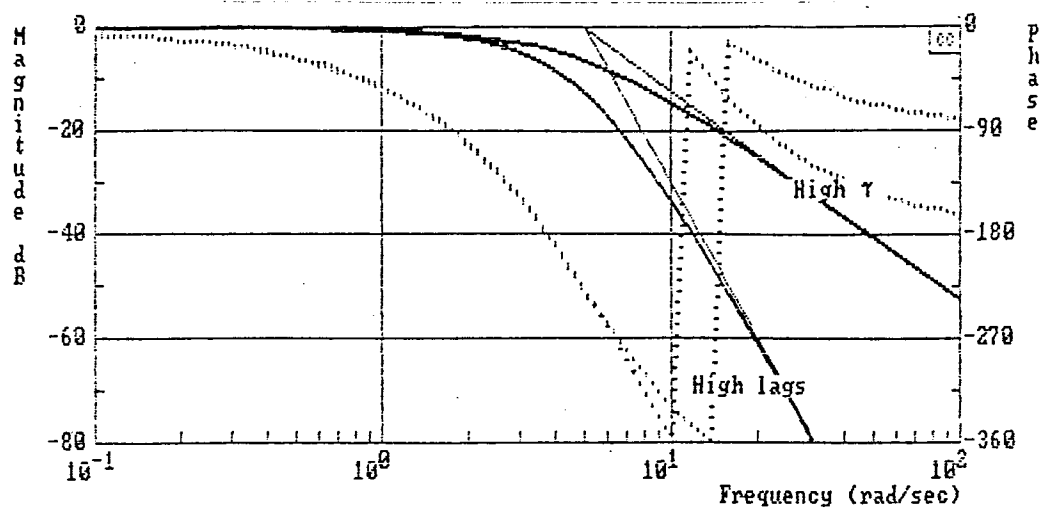


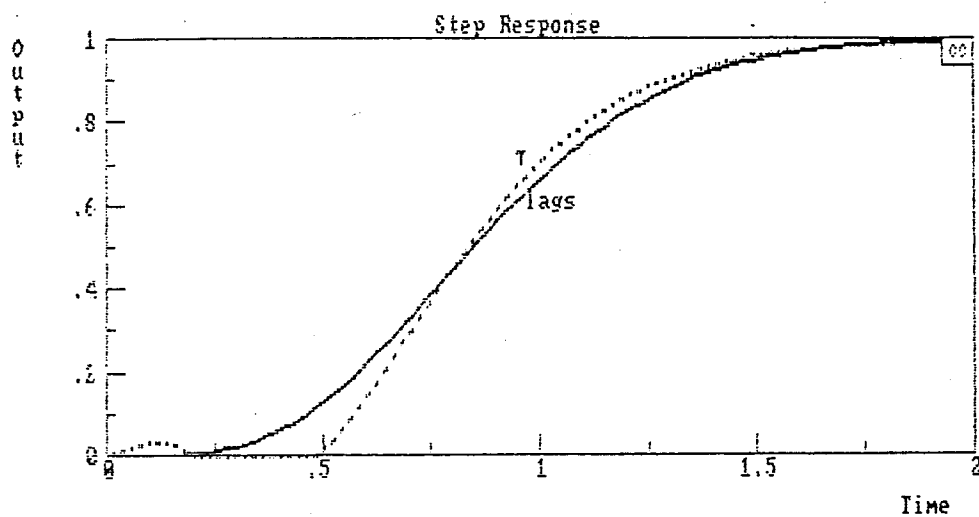
Figure 2(3.4.1.1). Average HQRs from Fixed-Base Pitch Tracking Experiment

configuration was designed specifically to approximate the phase characteristics of the high-delay case discussed above, up to a frequency of about 10 rad/sec. Figure 3 shows a comparison of the high-delay and high-lag cases. The frequency response in Figure 3a shows very similar phase curves (dotted lines) up to 10 rad/sec; beyond this frequency, the high-delay case exhibits considerably more phase lag. In addition, the magnitude characteristics (solid lines) are very different, since the high-lag case has a fifth-order denominator. The step responses in Figure 3b reflect these differences as well: for the first half-second, the responses are quite different, but beyond this they are very similar.

The bandwidth and phase delay values for these two cases are similar, as Table 1 indicates. The interest, therefore, is whether the pilots could detect significant differences due to either the large high-frequency phase differences, or the gain differences. The high-lag case lies just below the high-delay case on Figure 2, and it received an average rating of 3.75 (contrasted with 4.75 for the delay case). This one-rating difference can be at least partially attributed to the somewhat lower  $\tau_p$  for the high-lag case. In both instances, however, the pilots commented on the sluggish initial response, and they were especially aware of the initial hesitation in the high-delay case. Neither the ratings nor the comments reflect any significant differences between these two cases.



a) Frequency Response



b) Step Responses

$$\text{High } \tau: \quad \frac{\theta}{\theta_c} = \frac{25e^{-0.5s}}{(5)^2}$$

$$\text{High Lags:} \quad \frac{\theta}{\theta_c} = \frac{3125}{(5)^3 [0.7, 5]}$$

c) Dynamics

Figure 3(3.4.1.1). Comparison of High Time Delay and High Lag Cases

The simulation results of Figure 2 suggest that, for the three initial questions, 1) combinations of high bandwidth and high time delay are bad, but appear to be adequately covered by the current limits; 2) there are no significant differences, in terms of handling qualities, between high phase delays due to pure time delay and phase delays due to large lags; and 3) the limits of Figure 1a(3.4) appear to adequately account for very large values of  $\tau_p$ .

## 2. All Other MTEs in VMC and Fully-Attended Operation (Figure 1b(3.4))

This classification refers to tasks where the pilot's attention is not substantially distracted by communications, chart-reading, turbulence, etc. The only data set relevant to this sort of operation comes from Reference 25, described in the discussion for Paragraph 3.2.2. The short-term response characteristics of the configurations evaluated in Reference 25 are not known, but can be approximated using the values of  $M_q$  and  $M_\alpha$  given in Reference 25, as follows:

$$\frac{\theta}{-\delta_e} \dot{=} \frac{M_{\delta_e} (s + 1/T_{\theta 2})}{s [s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2]}$$

where

$$\begin{aligned} \frac{1}{T_{\theta 2}} &\dot{=} -Z_w \\ 2\zeta_{sp}\omega_{sp} &\dot{=} -Z_w - M_q \\ \omega_{sp}^2 &\dot{=} M_q Z_w - M_\alpha \end{aligned}$$

In Reference 25,  $Z_w = -0.38$  1/sec and  $M_{\delta_e} = 0.3$  rad/sec<sup>2</sup>/in. Figure 4 shows the approximate response characteristics of the Reference 25 configurations on a plot of  $2\zeta\omega_n$  vs.  $\omega_n$ . Several flying qualities criteria are also shown on Figure 4: the short-term response limits of the V/STOL specification, MIL-F-83300 (Reference 29), for forward flight; the requirement on normal acceleration from MIL-H-8501A (Reference 31); and a boundary based on a combination of minimum bandwidth of 1 rad/sec and minimum damping ratio of 0.35.

From Figure 4 it can be seen that a bandwidth of 1 rad/sec is an appropriate Level 1 limit for the tasks performed. This limit is consistent with the UCE = 1 limit of Paragraph 3.3.2.1.

## 3. All Other MTEs in IMC and/or Divided-Attention Operations (Figure 1c(3.4))

The IMC Mission-Task-Elements require a higher response bandwidth than VMC because of the natural increase in workload. By analogy, any high-workload environment (other than air combat, which is covered separately), either VMC or IMC, demands a high-bandwidth pitch response. The limits of Figure 1c(3.4) are identical to the similar requirement in the low-speed section of the specification, Paragraph 3.3.2.1. There is

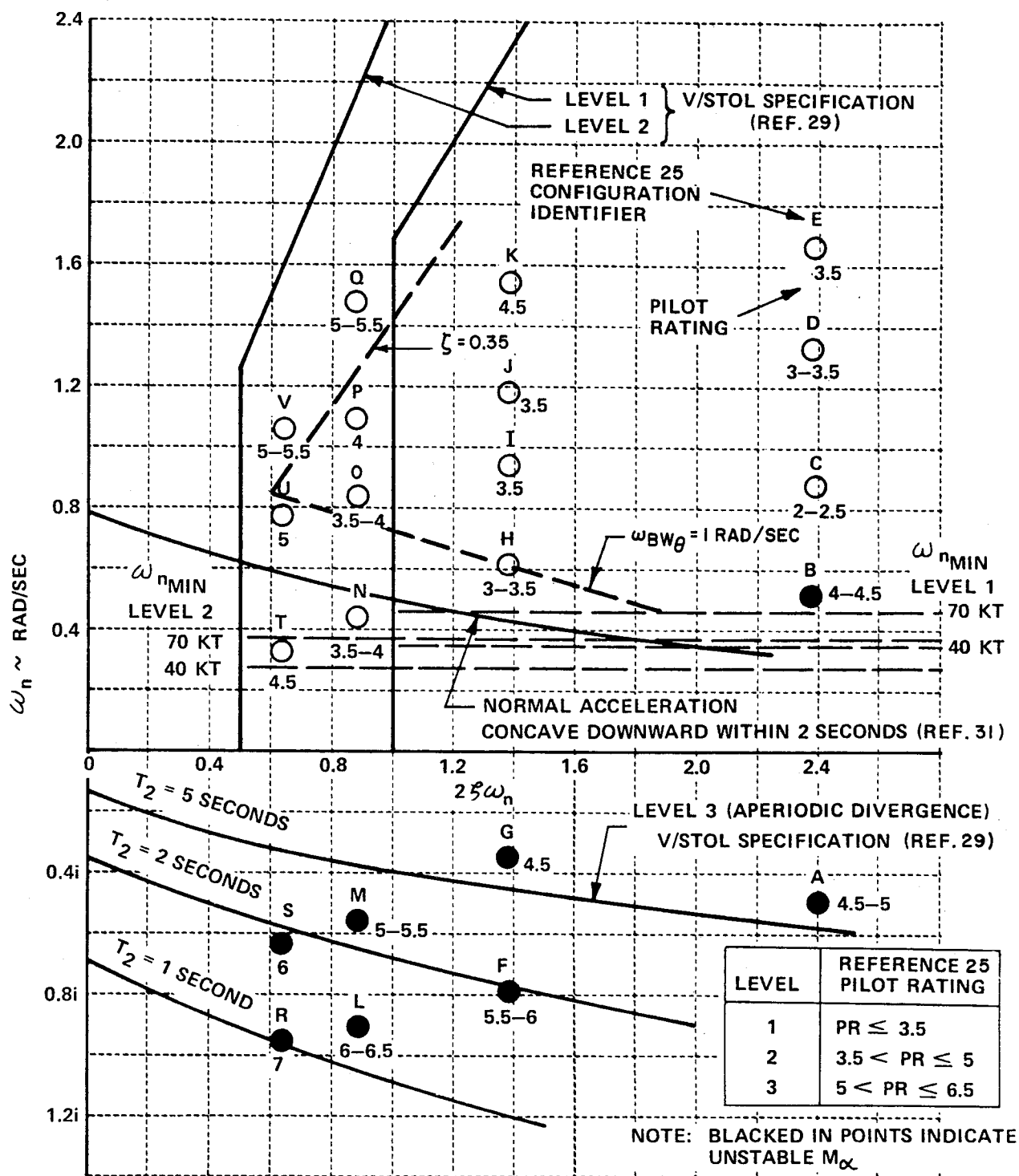


Figure 4(3.4.1.1). Comparison of Pilot Ratings with Approximate Short-Term Dynamics for CH-46 Flight Program of Reference 25 (Figure Taken from Reference 30)

less supporting information for the forward-flight requirement. As the discussion for Paragraph 3.2.2 reflects, almost all of the available data related to high-workload forward flight tasks pertains specifically to IMC operations. This data base can be divided into two distinct categories: 1) constant-speed IMC approaches, and 2) decelerating IMC approaches.

#### Constant-Speed IMC Approaches

The applicable data for this category are reviewed in some detail in the discussion for Paragraph 3.2.2. The pertinent pilot ratings have been extracted for this discussion.

The applicable pilot ratings for Rate Response-Types from References 3, 19, 20, 21, 22, and 23 are summarized in Figure 5. Configurations without an attitude reference, either unaugmented or rate-damped (i.e., rate augmented), received Level 2 (or worse) ratings at any bandwidth (Figure 2a). For Rate Response-Types with attitude feedback (RCAH), Figure 2b, the small data base suggests a minimum level of bandwidth somewhat less than 3 rad/sec. More data are clearly needed at the lower bandwidths; however, a Level 1 response limit of 2 rad/sec is suggested by the data. Relevant pilot ratings for Attitude Response-Types are summarized in Figure 6. All of the data of Figure 6 are based on 60-kt instrument approaches (ILS or MLS). The Reference 20 ratings are for actual flight tests, while the rest of the data are from simulations.

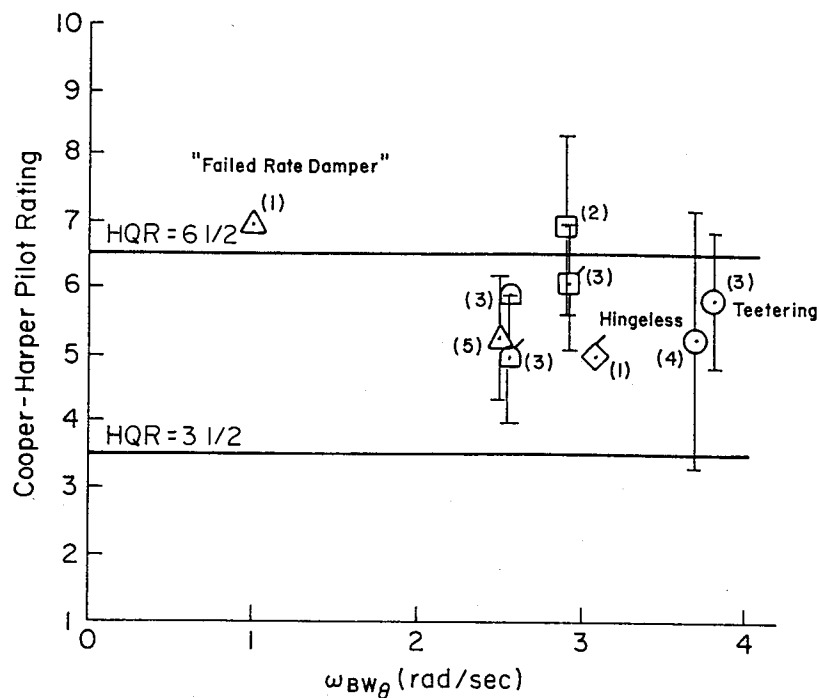
The relatively sparse set of data in Figure 6 is not conclusive, since the Reference 23 configuration was given Level 2 ratings with a bandwidth of about 3.1 rad/sec. These are, however, the only ratings for single-pilot operations and the approaches ended with a missed approach. A review of Reference 23 suggests that for dual-pilot operations -- consistent with all the other ratings on Figure 6 -- and with continued approaches, the pilot ratings would be Level 1. This suggests that the Level 1 bandwidth should be as high as 3 rad/sec for single-pilot operations, but it is felt that this is too small a data base to justify such a restriction at this time. A limit of 2 rad/sec is in agreement with the value for Rate Response-Types in Figure 5b. The Level 2 limit of 0.5 rad/sec is an extrapolation based on Figure 6.

The data of Figures 5 and 6 do not show any clear advantage when a three-cue flight director is available. This was also concluded in the discussion for Paragraph 3.2.2.

#### IMC Approaches with Deceleration on Glideslope

This Mission-Task-Element is specifically cited in Table 2(3.2) because of the demonstrated need for a three-cue flight director just to attain acceptable pilot ratings (i.e., Cooper-Harper ratings of 6 or better). Available pilot rating data for decelerating ILS or MLS approaches were not included in Figure 5 or 6 since the primary reason





a) Rate Damping (no Attitude Hold). Most Neutral Stick Control Position Gradient

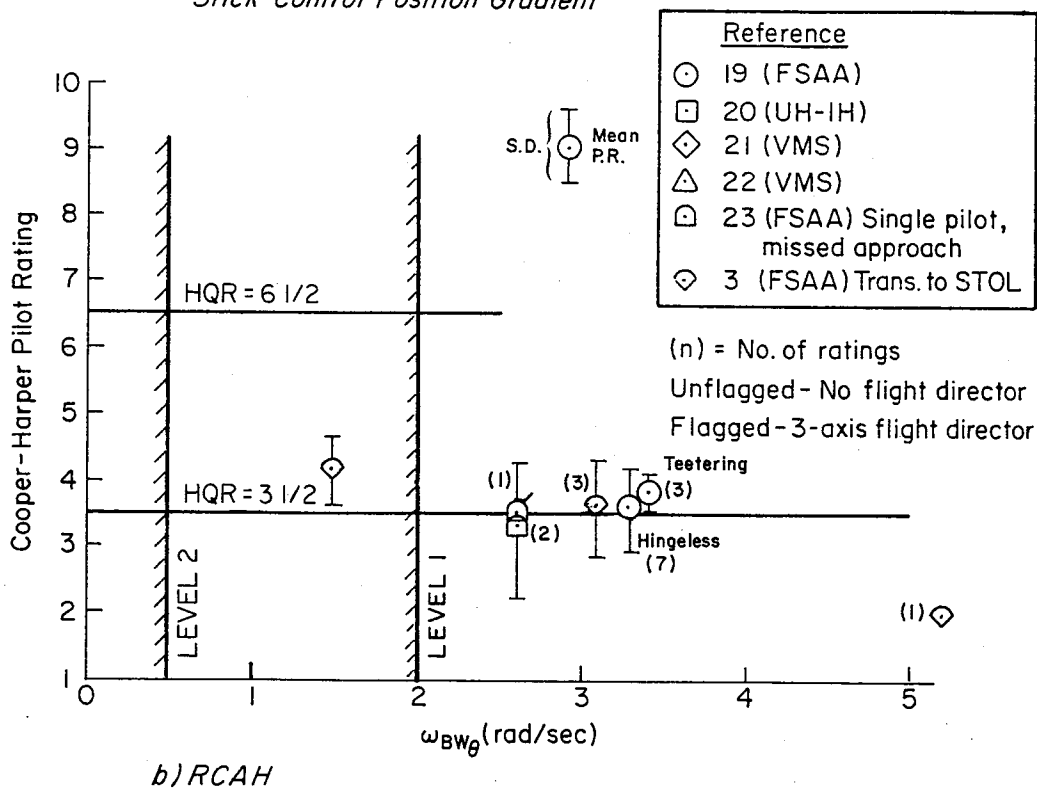


Figure 5(3.4.1.1). Summary of Pilot Ratings for Rate Response-Types in IMC (Constant Speed VOR or ILS Approach, Except Reference 3)

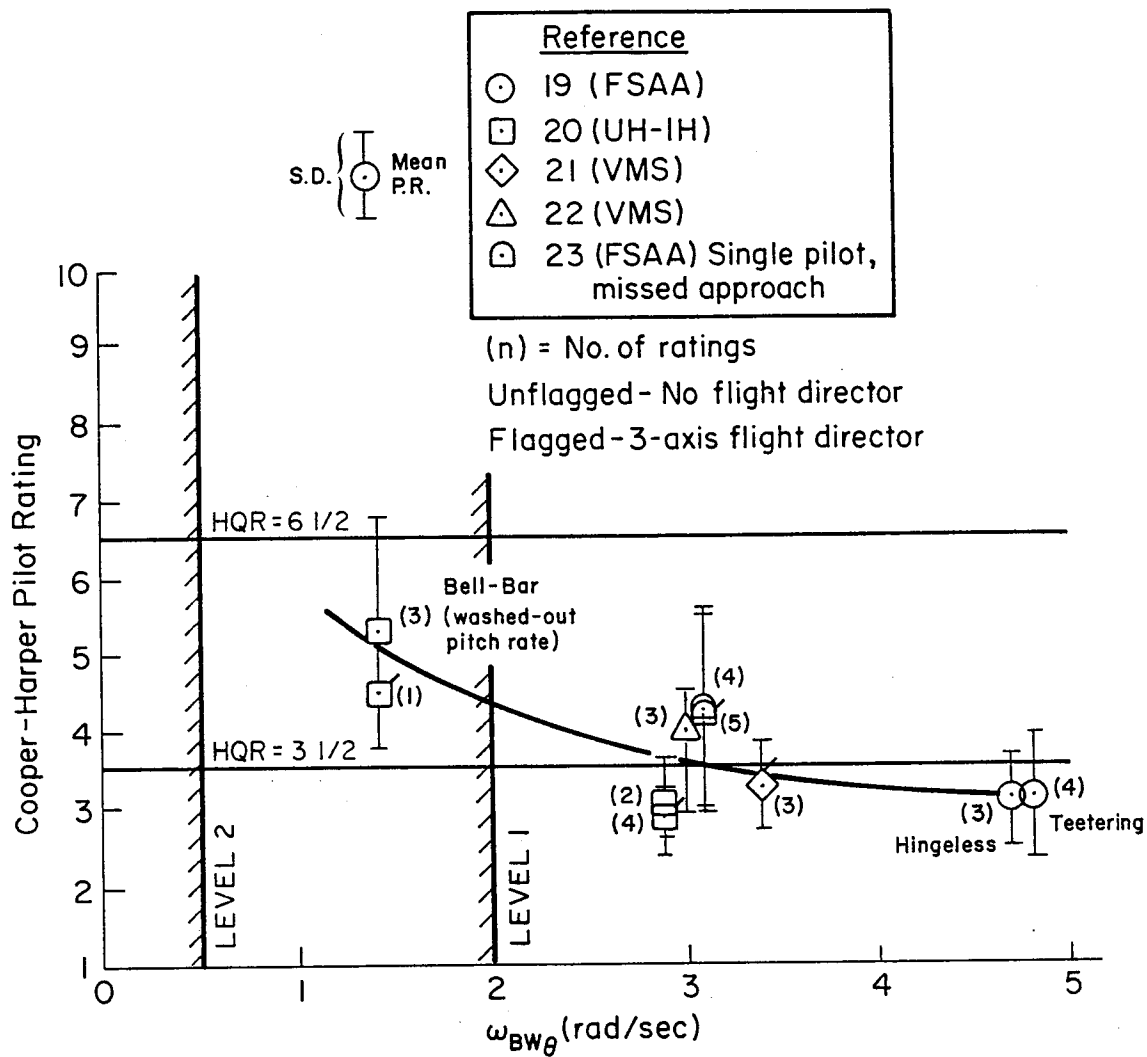


Figure 6(3.4.1.1). Summary of Pilot Ratings for Attitude Response-Types in IMC Constant-Speed Approach

for poor ratings is more task-related, rather than response-related. Relevant data for Rate Response-Types come from References 21, 37, and 38. Table 2 summarizes these data and Table 3 summarizes similar data for Attitude Response-Types.

TABLE 2(3.4.1.1). DATA FOR RATE RESPONSE-TYPES

REFERENCE	RESPONSE-TYPES	AVG. PR (No F.D.)	AVG. PR (with F.D.)	$\omega_{BW\theta}$
21 (All Decels)	Rate Damping	4.8	6.0	2.9
	RCAH	-	4.1	1.8
37	Rate Damping	9	7	2.7
38 (Heading Hold)	Rate Damping	6.6	3.7	?

TABLE 3(3.4.1.1). DATA FOR ATTITUDE RESPONSE-TYPES

REFERENCE	RESPONSE-TYPES	AVG. PR (No F.D.)	AVG. PR (with F.D.)	$\omega_{BW\theta}$
21 (All Decels)	ACAH	-	3.8	3.2
36	Bell-Bar (Washed out Pitch Rate)	8.3	7.3	1.4
	ACAH	6.7	5.3	?
	ACAH + Auto- Collective	6.0	4.0	
37	Attitude SAS	7	4	3.3
	Attitude CAS	7	3	2.4
38 (Heading Hold)	ACAH	6.4	4.4	?

With the exception of Reference 38, the ratings in Tables 2 and 3 are not conclusive. The possible reasons for the flight-director-off ratings of Reference 21 showing improvement (Rate Response-Type) concern the design of the flight director itself, as described in the background discussion for Paragraph 3.2.2. The bandwidth of the Reference 38 helicopter (the NAE variable stability Bell 205A) is not known, but appears to be less than 2 rad/sec.

It is possible that, as for aggressive VMC Mission-Task-Elements, the minimum response bandwidth should be increased for decelerating IMC approaches, but there is no data upon which to base such a requirement. Instead, Table 2(3.2) in Paragraph 3.2.2 requires the use of Attitude Hold and a three-cue flight director.

#### 4. Comparison with Operational Helicopters

Figure 7 shows the bandwidths of several operational helicopters as a function of airspeed compared with the limits of Figure 1(3.4). Based on this figure, most unaugmented helicopters for which augmentation is available (i.e., the AH-1G, UH-1H, CH-53D, and CH-47B) are Level 2 or 3. For IMC operations, all of the augmented helicopters shown are Level 2 as well, with the exceptions of the CH-53D and the UH-60 (data for some of the augmented helicopters are not available).

##### d. Guidance for Application

Since the requirements are similar, the reader should refer to the discussion in Paragraph 3.3.2.1 (Hover and Low Speed) and to Appendix A.

##### e. Related Previous Requirements

The longitudinal dynamic stability requirements of MIL-H-8501A, Paragraphs 3.2.11 and 3.6.1.2, express stability in terms of periods, cycles, and times to half or double amplitude of oscillatory modes. In addition, the "concave-downward" requirements of Paragraph 3.2.11.1 can be interpreted in terms of short-term damping ratio and undamped natural frequency. The following discussion will compare these limits with the limits on response specified by bandwidth, as well as the short-term requirements from the V/STOL and CTOL specifications, MIL-F-83300 and MIL-F-8785C, respectively.

Figure 8 shows a comparison plot of the short-term response requirements from MIL-H-8501A, MIL-F-83300, and MIL-F-8785C with a second-order interpretation of bandwidth. The "concave downward" normal acceleration requirement of MIL-H-8501A Paragraph 3.2.11.1 has been converted to damping ratio and frequency by Seckel (Reference 32); as shown by the root locus of Figure 9, this requirement can be interpreted as well as a limit on  $M_w$ , or maneuver margin. Thus, the "concave downward" requirement is intended to regulate against excessive instability with respect to angle of attack; positive values of  $M_u$  can produce oscillatory instabilities, which the oscillatory response limits

Aircraft	Ref.	
⊙ OH-6A	28	Analytical Models (derivatives and transfer functions) 125 ms delay added to simulate actuators, etc.
⊠ BO-105C	28	
△ AH-1G	28	
◇ UH-1H	28	
⊡ CH-53D	28	
◊ CH-47B	146	
■ UH-60	5	Flight Data (frequency sweeps)
■ Bell 214ST	117	

Note: Flagged symbol indicates SCAS on

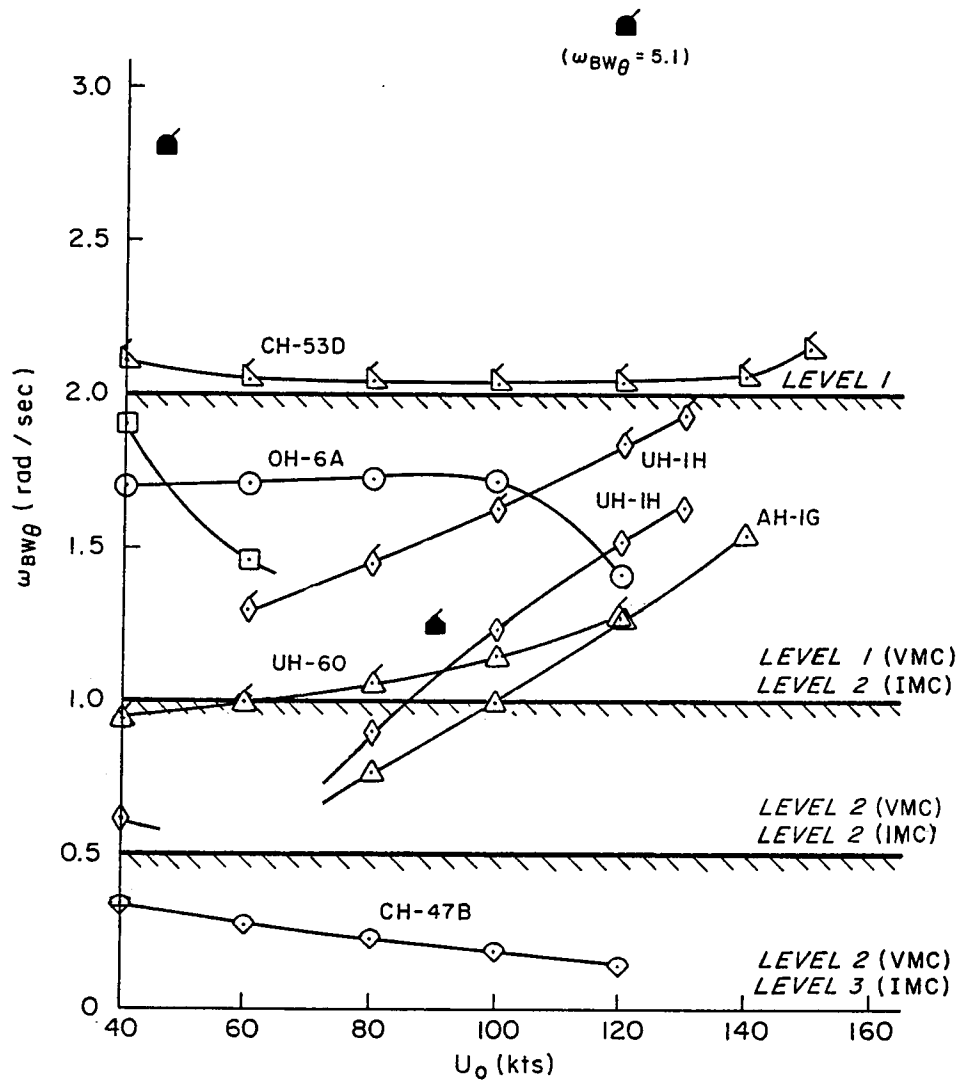


Figure 7(3.4.1.1). Pitch Attitude Bandwidths of Operational Helicopters

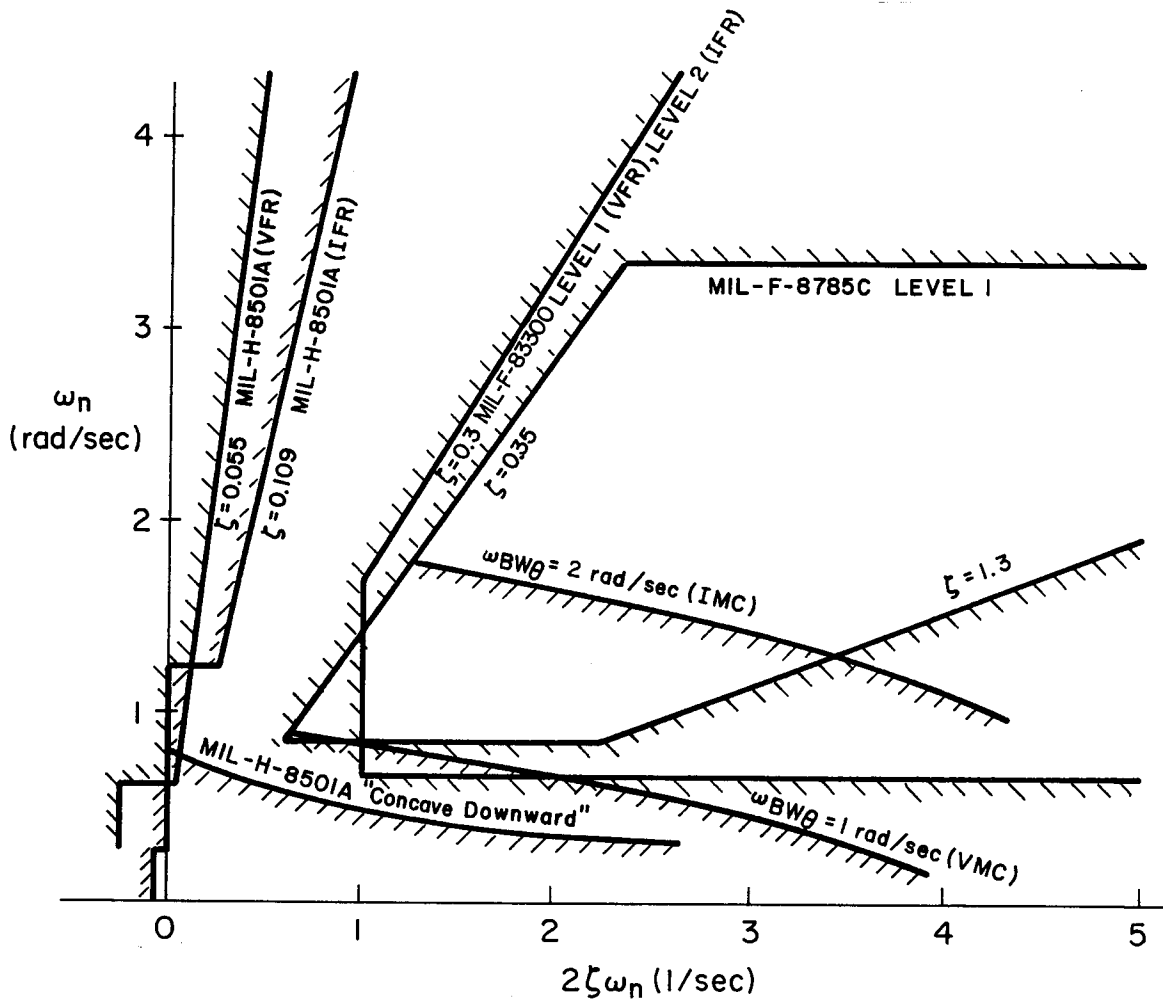


Figure 8(3.4.1.1). Comparison of Response Requirements from MIL-H-8501A, MIL-F-83300, and MIL-F-8785C with Bandwidth Requirements. (Limits Based on a Helicopter at 70 kt with  $1/T_{\theta 2} = 1.0$  rad/sec)

of MIL-H-8501A regulate against. It is important to remember that MIL-H-8501A is the only specification not written in terms of Levels of flying qualities, and is instead a "pass/fail" specification.

The V/STOL forward flight requirement (Reference 29, Paragraph 3.3.2) specifies that "all roots of the longitudinal characteristic equation of the aircraft shall be stable" and that the short-term mode that defines angle-of-attack response to pitch controller meet the oscillatory requirements shown in Figure 8. These requirements are seen in Figure 8 to be more stringent than the MIL-H-8501A limits.

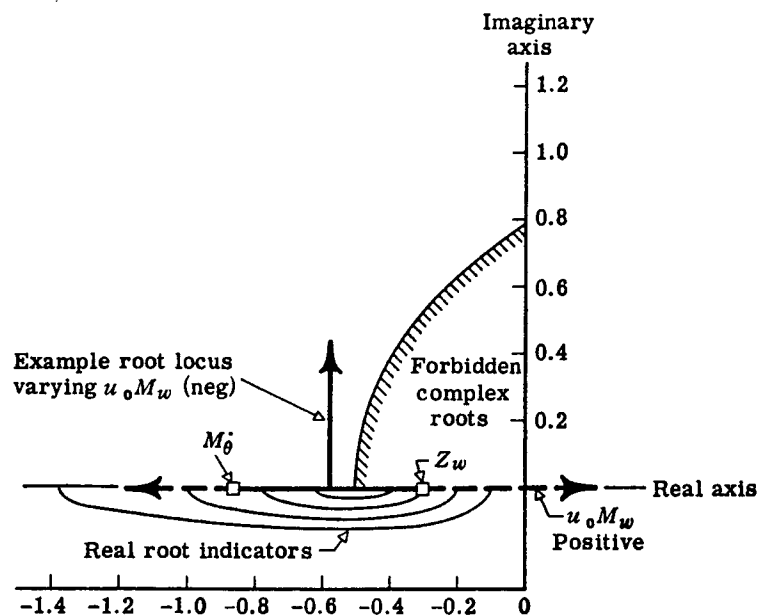


Figure 9(3.4.1.1). Concave Downward Requirements  
(from Reference 32)

Since the characteristics of a helicopter flying at high speeds are very airplane-like, it would be expected that the MIL-F-8785C requirements for conventional airplanes could be applied for forward flight. The boundaries shown from MIL-F-8785C in Figure 8, for Level 1 flying qualities, are close to the MIL-F-83300 limits [the MIL-F-8785C frequency limits are functions of  $n/\alpha = (V/g) (1/T_{\theta_2})$  for Figure 8, values of  $1/T_{\theta_2} = 1$  rad/sec and  $V = 70$  kt were used]. Again, these limits are considerably more demanding than MIL-H-8501A.

The final lines on Figure 8 are for a simple second-order interpretation of constant bandwidth (1 and 2 rad/sec) for Rate-Response-Types. Figure 8 shows that the Level 1 VMC limit in the specification is close to MIL-F-83300 and MIL-F-8785C, while the IMC limit is more stringent than any of the other criteria.

a. Statement of Requirements

3.4.1.2 Mid-Term Response to Control Inputs. The mid-term response characteristics apply at all frequencies below the bandwidth frequency obtained in Paragraph 3.4.1.1. Use of an Attitude Hold Response-Type (Paragraph 3.2.6) constitutes compliance with this paragraph, as long as any oscillatory modes following a pulse controller input have an effective damping ratio of at least 0.35. If Attitude Hold is not available, the applicable criterion (Paragraph 3.4.1.2.1 or 3.4.1.2.2) depends on the degree of divided attention necessary to complete the Mission-Task-Elements according to Paragraph 3.1.1.

3.4.1.2.1 Fully Attended Operations. For those Mission-Task-Elements specified in Paragraph 3.1.1 as required fully attended operation (Paragraph 2.5.1), the mid-term response shall meet the requirements of Figure 3(3.3).

3.4.1.2.2 Divided Attention Operations. For those Mission-Task-Elements specified in Paragraph 3.1.1 as requiring divided attention operations (Paragraph 2.5.2), the limits of Figure 3(3.3) shall be met, except that the Level 1 damping shall not be less than  $\zeta = 0.35$  at any frequency.

b. Rationale for Requirements

These requirements are intended to insure adequate mid-term stability (for divided-attention operations), or minimize the mid-term instability (for fully attended operations). They are identical to Paragraphs 3.3.2.2, 3.3.2.2.1, and 3.3.2.2.2.

c. Supporting Data

There are no data specifically oriented toward forward flight. There is, however, no reason to expect the fundamental requirements from the pilot's point of view to be different from low-speed flight -- for which there is data (see Supporting Data for Paragraph 3.3.2.2).

d. Guidance for Application

See discussion for Paragraph 3.3.2.2.

e. Related Previous Requirements

None.



a. Statement of Requirement

3.4.1.3 Mid-Term Response -- Maneuvering Stability

a) Control Feel and Stability in Maneuvering Flight at Constant Speed. In steady turning flight at constant airspeed, and in pullups and pushovers, for Levels 1 and 2 there shall be no tendency for the aircraft pitch attitude or angle of attack to diverge aperiodically. For the above conditions, the incremental control force required to maintain a change in normal load factor and pitch rate shall be in the same sense (aft - more positive, forward - more negative) as those required to initiate the change. These requirements apply for all local gradients throughout the range of operational and service load factors defined in 2.9.1 and 2.9.2.

b) Control Forces in Maneuvering Flight. The variations in longitudinal cockpit control force with steady-state normal acceleration shall have no objectionable nonlinearities throughout the Operational Flight Envelope. At no time will a negative local gradient be permitted. In addition, deflection of the pilot's cockpit controller must not lead the control force at any frequency below 5 rad/sec. For Level 1 the following requirements apply:

- For centerstick controllers, the local force gradient,  $F_s/n$ , shall be no less than 13 N/g (3 lb/g) and no greater than 67 N/g (15 lb/g).
- For sidestick controllers, the local force gradient shall be no less than 13 N/g (3 lb/g) and no greater than 26 N/g (6 lb/g).
- The local slope of  $F_s$  vs.  $n$  should be relatively constant over the range of normal accelerations within the Operational Flight Envelopes. A variation of more than 50 percent shall be considered as excessive.

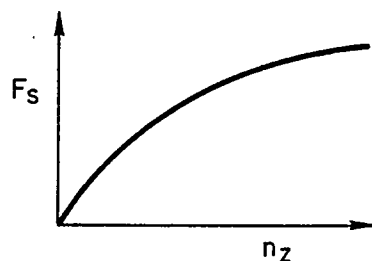
b. Rationale for Requirement

During the specification development process, several of the helicopter manufacturers' representatives have expressed the need for a requirement that addresses maneuvering stability in forward flight. In MIL-H-8501A, the so-called "concave downward" requirement (see Related Previous Requirements discussions for this requirement and for Paragraph 3.4.1.1) is intended to assure an adequate maneuver margin. The combination of control feel and control force requirements stated here also addresses maneuvering stability. Use of stick force per g instead of a "concave downward" requirement at forward flight speeds has been recommended in the past (e.g., References 44 and 122), and has been adopted here for several reasons:

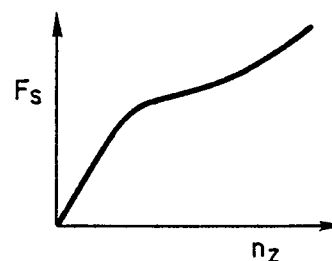
- Both  $F_s/n$  and "concave downward" assure maneuvering stability; however, the latter does not assure a positive indication to the pilot, i.e., control sensitivity and feel;
- It is felt that the  $F_s/n$  requirement is easier to apply and demonstrate compliance with;
- The "concave downward" requirement applies specifically only to very small inputs (see Related Previous Requirements), and the effort involved to extend it for larger inputs, and to revise the limits for inflection time to reflect current stability requirements, is not justified.

The requirements stated in this paragraph are based on requirements in MIL-F-8785C (Reference 66). The wording used here more closely follows the recommended revision to MIL-F-8785C, Reference 1. In both References 1 and 66, limits on  $F_s/n$  are stated as functions of limit load factor,  $n_L$ , and the ratio of normal acceleration to angle of attack,  $n/\alpha$ . Reference 1 discusses some of the shortcomings to such a functional dependence, and the approach taken here has been to specify upper and lower limits as a function of controller type only.

Local variations in the stick force vs. normal acceleration plot manifest themselves as stick force lightening as the pilot "pulls in" more load factor. This is undesirable and produces a tendency to overshoot the target load factor or to "dig in." MIL-F-8785C is written to prevent more than a 50% change in the local slope of stick force vs. normal acceleration to prevent this effect. Experience has shown, however, that a long term monotonic variation in slope is not particularly objectionable, and that a requirement limiting the change in slope to less than 50% may be overly restrictive. This is illustrated in the following sketch.



This is ok as long as  
local slope is  $\geq 3 \text{ lb/g}$



This not ok even if  
local slope is  $\geq 3 \text{ lb/g}$

Changes in local slope are restricted to less than 50% over a lg range of  $n_z$  to eliminate the undesirable effect.

The last part of this requirement prohibits the cockpit control deflection from leading the control force. Controllers where deflection leads the force tend to occur with a poorly designed feel system, and always result in a strong PIO tendency (see Reference 1, page 275). The criterion only applies to the frequency range of piloted control, which is conservatively set with an upper limit of 5 rad/sec for this requirement.

### c. Supporting Data

There is considerable support in the fixed-wing community for the maneuvering stability requirements. Reference 1 presents this support. The major observations from that reference, plus some limited data for rotorcraft, will be discussed here to justify the specified values of  $F_s/n$ .

Support for  $F_s/n$  as specified in MIL-F-8785C comes entirely from flight investigations of short-period characteristics using the USAF/Calspan NT-33A (see References 1 and 82). In these experiments, various combinations of short-period frequency and damping,  $\omega_{sp}$  and  $\zeta_{sp}$ , were evaluated with  $F_s/n$  chosen by the pilots. Thus, the useful information was basically a byproduct of the tests, since pilot ratings could reflect an interrelationship of  $\zeta_{sp}$ ,  $\omega_{sp}$ , and  $F_s/n$ . A review of these data for centerstick controllers, and for a variety of tasks and flight conditions (Reference 1), shows:

- There is a wide range of  $F_s/n$  that is considered acceptable -- i.e., Level 1 in terms of Cooper-Harper pilot ratings;
- There is a definite preference for low values of  $F_s/n$  for fighter aircraft, especially at high forward flight speeds, with an optimum value of around 4-6 lb/g;
- In most cases, Level 2 pilot ratings occur when  $F_s/n$  is greater than about 10 lb/g. This does not necessarily mean that this is an excessive gradient, since, for the data in question, some other aircraft characteristic may have been marginal, resulting in Level 2 ratings, and thus the gradient chosen may have been an attempt to suppress this undesirable characteristic.

Additional insight from the fixed-wing world comes from simulation and flight experience with the B-1 bomber (Reference 123) in terrain flight. It was found that 17 lb/g was too high based on pilot fatigue, and that a good design target is 7-8 lb/g. These numbers are based on a conventional airplane where pitch is the only available short-term

flight path controller; for a rotorcraft with collective,  $F_s/n$  on the order of 17 lb/g may not be considered excessive.

Supporting data for rotorcraft are available in References 124, 125, and 126.

In Reference 124, several longitudinal and lateral feel springs were evaluated on a training helicopter. Maneuvers included hovering transitions to forward flight, quickstops, and pullups and windup turns in forward flight. One configuration included a 14 lb/g bobweight with a viscous damper. No pilot ratings were given, but comments in Reference 124 indicate that the bobweight resulted in desirably light forces in hovering flight, and supplied much-needed force feel in forward flight. This supports the requirement for  $F_s/n$ , and suggests that 14 lb/g is in the acceptable range.

Reference 125 documents the development of a feel augmentation system for the CH-53D helicopter. This paper is a good summary of the advantages and shortcomings of various forms of feel systems discussed in Guidance for Application. The feel augmentation system chosen, using a combination of pitch rate and longitudinal cyclic deflection and rate as feedbacks, provided a gradient of about 15-18 lb/g at 80 kt and above, decreasing to about 2 lb/g at hover. Again no pilot ratings are available, but the feel system was a vast improvement over the basic CH-53D.

Reference 126 contains extensive pilot commentary and pilot ratings from development of a feel system for the S-67. The implementation of this system closely parallels that used for the CH-53D in Reference 125. Figures 1 and 2 illustrate the stick force per g characteristics of the final feel system. As Figure 2 shows,  $F_s/n$  increases linearly from 18 lb/g at 80 kt to 25 lb/g at 170 kt. Pilot comments by three government evaluation pilots reflect the desirability of some force feel, but also indicate that the chosen level of  $F_s/n$  was too high: "The force gradient in the high 'q' regime was assessed to be too high in high g maneuvering flight, but acceptable during low g maneuvers (below 1.5g) in the same regime.... The value of high stick forces as limit load factor is approached is obvious and desirable.... Overall rating of the FAS system [Feel Augmentation System] is determined to be 4 (Cooper-Harper)...." From the second pilot, "During more severe maneuvers, the benefits of the FAS became readily apparent. The positive maneuvering stability in terms of cyclic stick force per g provided the necessary cue that has always been missing in helicopter control characteristics for maneuvering flight.... However, the force required at load factor levels greater than approximately 1.5g was considered to be slightly high." Reference 126 notes in summary that "it was felt by the Sikorsky pilots that reducing the force gradient to the 10 to 15 pound per g range would eliminate the complaint of high forces at high load factors." This is the primary basis for specifying a 15 lb/g upper limit on  $F_s/n$ .

The lower limit of 3 lb/g appears to be universally accepted for all types of aircraft and controllers: it is the limit for conventional airplanes (MIL-F-8785C) and V/STOLs (MIL-F-83300) with both wheel and

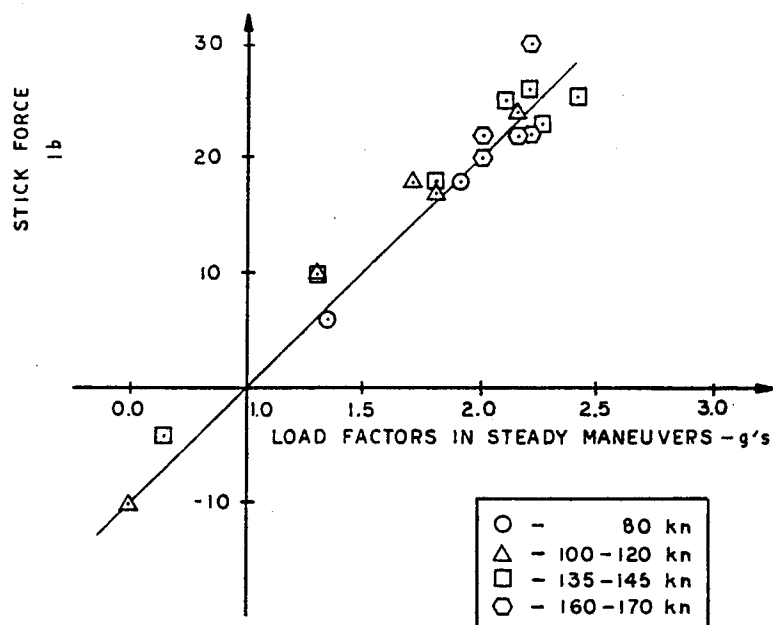


Figure 1(3.4.1.3). Pitch Control Forces During Constant Load Factor Maneuvers (from Reference 126)

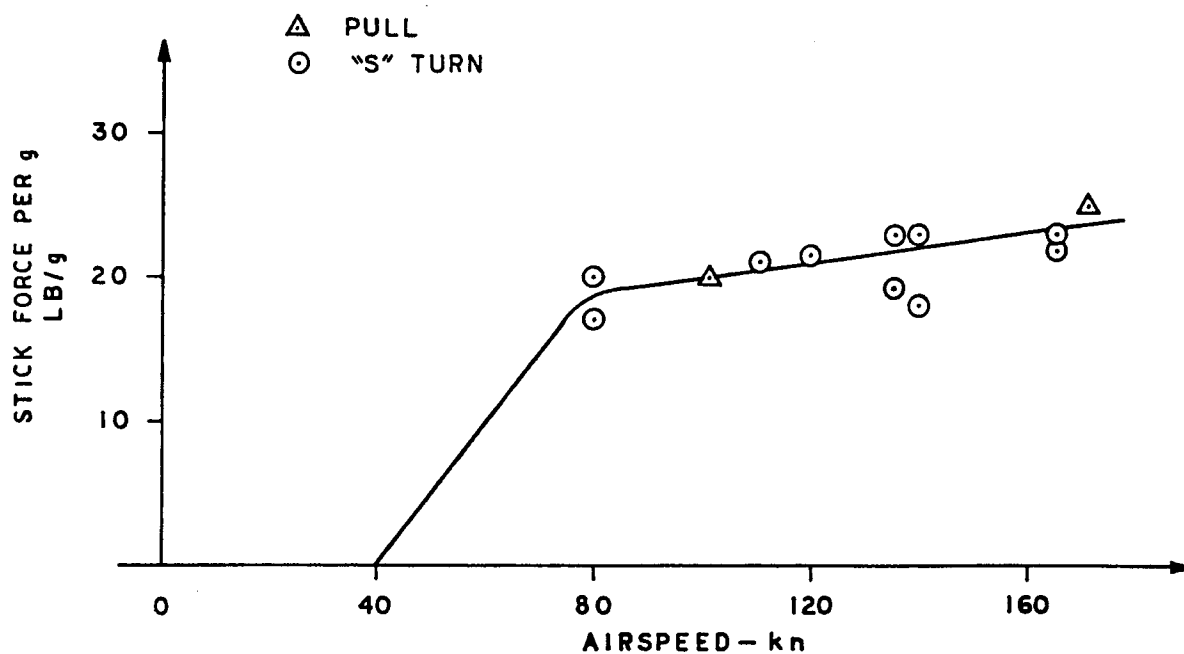


Figure 2(3.4.1.3). Force Gradient in Steady Maneuvers for S-67 (from Reference 126)

centerstick controllers. In addition, it is supported by side stick controller data for conventional airplanes, as the following discussion shows.

Support for the sidestick limits of 3-6 lb/g for Level 1 comes from an in-flight evaluation of force/response and force/deflection tradeoffs conducted by the USAF Test Pilot School, Edwards AFB, using the variable-stability NT-33A and reported in Reference 64. Results of this evaluation are documented in Figure 3 (reproduced with minor changes from Reference 64).

The lateral force/deflection characteristics were varied to maintain control harmony for the air-to-air evaluations (consisting of 2g bank-to-bank and 3.5g wind-up turns). If the pilot comments indicated that control harmony detracted from the rating given, variations in control harmony were evaluated.

With the exception of the light  $F_s/n$  and large  $F_s/\delta_s$  (Configurations 1 and 2), there is not a substantial variation in pilot ratings over the test matrix.

In general, pilots preferred increased control stick motion with decreased control force gradients, and decreased control stick motion with increased control force gradients. Control Configurations 13, 14, and 15 of Figure 3 yielded the best results, in both pilot ratings and comments. Pilots indicated that control motions were noticeably large but not uncomfortable. These configurations were on the edge of the test matrix; thus, the extent of this favorable region was not determined, but suggests that a conservative range of  $F_s/n$  is somewhere between 3 and 8.6 lb/g. A lower limit of 3 lb/g is consistent with all other flight data, while an upper limit of 6 lb/g is suggested by the Figure 3 data.

#### d. Guidance for Application

The primary concern with application of this new (to the helicopter community) requirement is in testing for compliance. However, it is appropriate to also address issues related to mechanization of artificial feel systems.

### TESTING FOR COMPLIANCE

The following discussion is concerned with the flight testing aspects of determining control force characteristics.

There are several methods for obtaining the required control force data. The best method to use depends primarily on the speed range under consideration. A major factor in determining the appropriate method for a given speed range is that load factor control gradients are defined for constant speed and collective. The method selected must therefore result in zero or small speed changes with  $n$ , or at least include a means for eliminating the effects of any speed changes.

Initial Longitudinal Control Force/Response Gradient (lb/g)		Control Force/Deflection Gradient (lb/deg)					
Very Light (3.0)	Light (4.0)	Medium (5.0)	Heavy (8.6)	1.1	1.4	2.0	5.0
				1	5	6	7
				9	10	11	12
				13	14	15	16
No pitch bobble tendency but imprecise positioning. Avg of CH 3.7	Pitch and lateral are both too sensitive. Avg of CH 4.4	Pitch and lateral both a little too sensitive. Avg of CH 5.1	Pitch extremely sensitive. Lateral fair. Avg of CH 6.7	Pitch a little sensitive. Lateral wandering and sensitive. Avg of CH 6.0	Pitch a little sensitive. Lateral slow to respond. Avg of CH 5.0	Pitch very stable at higher g's, but forces tiring. Avg of CH 4.1	
Pitch and lateral steady and responsive. Motion noticeably large. Avg of CH 2.9	Pitch a little sensitive. Lateral bobble. Avg of CH 4.3	Slight pitch bobble. Better at higher g's. Lateral sluggish (cont. harmony). Avg of CH 4.5	Pitch too sensitive. Lateral wandering and sensitive. Avg of CH 6.0	Pitch steady once on tgt. Lateral forces high (cont. harmony) Avg of CH 3.85	Pitch steady but forces too heavy. Lateral forces too heavy. Tiring. Avg of CH 4.3		
Motion noticeably large. No pitch bobble, slightly sluggish. Avg of CH 3.3	Very slight pitch bobble tendency, but good. Large lateral corrections difficult. Forces high & bobble. Avg of CH 4.4	Pitch steady once on tgt. Lateral forces high (cont. harmony) Avg of CH 3.85	Pitch a little sensitive. Lateral slow to respond. Avg of CH 5.0	Pitch very stable at higher g's, but forces tiring. Avg of CH 4.1			
A/C very sluggish and forces uncomfortable. Avg of CH 5.0	A/C sluggish but stable. Forces heavy. Avg of CH 4.5	Pitch steady but forces too heavy. Lateral forces too heavy. Tiring. Avg of CH 4.3	Pitch very stable at higher g's, but forces tiring. Avg of CH 4.1				

Figure 3(3.4.1.3). Pilot Comments for NT-33A with Sidestick Flying Air-to-Air Tasks (from Reference 64)

One method is to use a series of alternating symmetric pullups and pushovers, sequenced to minimize the airspeed and altitude changes. The control is held fixed after each input until the short-term motion becomes steady state, and measurements are taken at a near-level attitude. An alternate version of this method is to stabilize the rotorcraft holding various amounts of out-of-trim control force in straight flight, then suddenly release the control, and measure the resulting normal acceleration increment.

Another method is to perform a series of stabilized turns after trimming the rotorcraft in level flight. The load factor can be changed by changing the bank angle, and the airspeed held constant by using a different rate of descent for each load factor. The collective and trim controller should be left at their trim settings throughout the maneuver to minimize the possibility of introducing extraneous inputs. The gradients obtained in this manner will not be quite as linear as with the symmetric pullup method because of the difference in pitch rate between pullups and turns. But with the possible exception of a more stable slope near 1g in the turns, the differences are generally small enough to be within the measurement errors.

A third method that is sometimes used involves a windup turn. After trimming in level flight, a turn of a certain number of g's is initiated, and the speed is allowed to decrease slowly as the g-level and altitude are held constant. The test is then repeated at several other g-levels until the complete range is covered. In this way, control gradient data can be obtained rapidly for several speeds. Again, the trimmer and collective should be left at the trim settings, and the rate of change of airspeed controlled by changing the rate of descent. The major disadvantage of this method is that it is less accurate because more careful pilot technique is required.

In general, the symmetric pullup method will work well at high speeds, but the airspeed changes will be excessive if the method is used at low speeds. On the other hand, the turn methods work well at low speeds, but can cause excessive altitude changes at high speeds. Also, it is impossible to obtain data for  $n$  less than 1.0 using turn methods.

#### PRACTICAL DESIGN CONSIDERATIONS

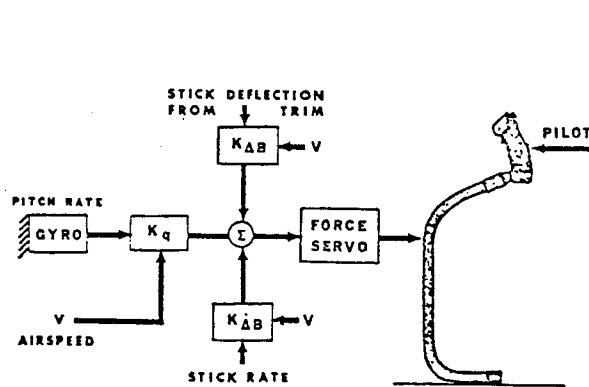
Reference 125 represents a summary of the design considerations for implementing an artificial feel augmentation system. Table 1 is a very condensed summary of these considerations; the interested reader should refer to Reference 125. The final feel system proposed for the CH-53D in Reference 125, as well as for the S-67 in Reference 126, uses pitch rate to replace normal acceleration as a feedback; i.e., at constant speed  $U_0$ , normal acceleration  $n$  is proportional to pitch rate  $q$  by:

$$n \approx U_0 q$$

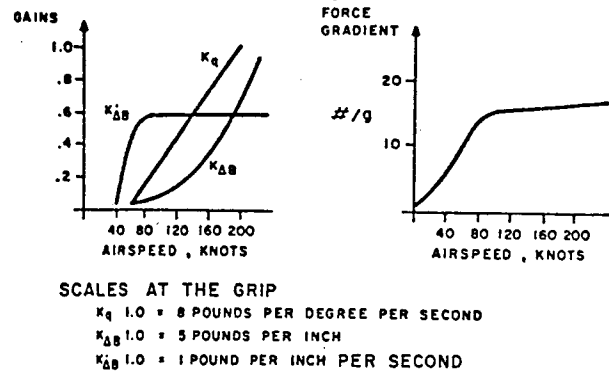


TABLE 1(3.4.1.3). DESIGN APPROACHES FOR CYCLIC FEEL AUGMENTATION (ADAPTED FROM REFERENCE 125)

- BOBWEIGHT: Provides  $F_s/n$  that is invariant with airspeed; appropriate for fixed-wing aircraft, not for rotorcraft with collective -- bobweight does not discern between load factors induced by cyclic inputs from load factors induced by collective inputs. Leads to undesirable collective-to-cyclic control coupling.
- Q-SPRING: Spring gradient (pounds per inch of deflection) reflects cyclic activity, but forces felt by the pilot do not correlate well with the resultant load factor during maneuvers.
- STICK RATE (DAMPER): Forces proportional to rate of control displacement are zero in a steady state maneuver; low forces are desirable in hover but inadequate in forward flight.
- ANGLE-OF-ATTACK FEEDBACK: Power adjustments via collective result in angle-of-attack variations unrelated to pitch inputs.
- Q-SPRING AND DAMPER: Inadequate in forward flight due to problems noted above, but improved feel in hover.
- BOBWEIGHT AND DAMPER: Improve some of problems of each alone, but collective-induced load factor changes still reflected in cyclic forces.
- Q-SPRING AND BOBWEIGHT: Improved stick force cues, but still produces unacceptable collective-pitch coupling.



a) Block Diagram



b) Gain Programs and Resultant Force Gradient

Figure 4(3.4.1.3). Feel Augmentation System Implemented for CH-53D (from Reference 125)

A multiplying gain varied as a function of airspeed provides the desired feedback. Additional feedbacks of stick deflection and stick rate were employed. Figure 4 shows a schematic for the implementation of this system, as well as the gain schedule and resulting force gradient as a function of airspeed.

Both Figures 2 and 4 emphasize an important point concerning stick force gradients: it is not the intent of this requirement to suggest that any gradient between 3 and 15 lb/g is acceptable; indeed, it will probably be necessary to provide a gradient at or near the upper limit at high speeds, and at or near the lower limit at the low end of the forward flight regime.

#### e. Related Previous Requirements

This requirement replaces the so-called "concave downward" requirement of MIL-H-8501A, which reads as follows:

3.2.11.1 The following is intended to insure acceptable maneuver stability characteristics. The normal acceleration stipulations are intended to cover all speeds above that for minimum power required; the angular velocity stipulations shall apply at all forward speeds, including hovering.

(a) After the longitudinal control stick is suddenly displaced rearward from trim a sufficient distance to generate a 0.2 radian/sec pitching rate within 2 seconds, or a sufficient distance to develop a normal acceleration of 1.5g within 3 seconds or 1 inch, whichever is less, and then held fixed, the time-history of normal accelera-

tion shall become concave downward within 2 seconds following the start of the maneuver, and remain concave downward until the attainment of maximum acceleration. Preferably, the time-history of normal acceleration shall be concave downward throughout the period between the start of the maneuver and the attainment of maximum acceleration. Figure 5(a) is illustrative of the normal acceleration response considered acceptable.

(b) During this maneuver, the time-history of angular velocity shall have become concave downward within 2.0 seconds following the start of the maneuver, and remain concave downward until the attainment of maximum angular velocity; with the exception that for this purpose, a faired curve may be drawn through any oscillations in angular velocity not in themselves objectionable to the pilot. Preferably, the time-history of angular velocity should be distinctly concave downward throughout the period between 0.2 second after the start of the maneuver and the attainment of maximum angular velocity. Figure 5(b) is illustrative of the angular velocity response considered acceptable.

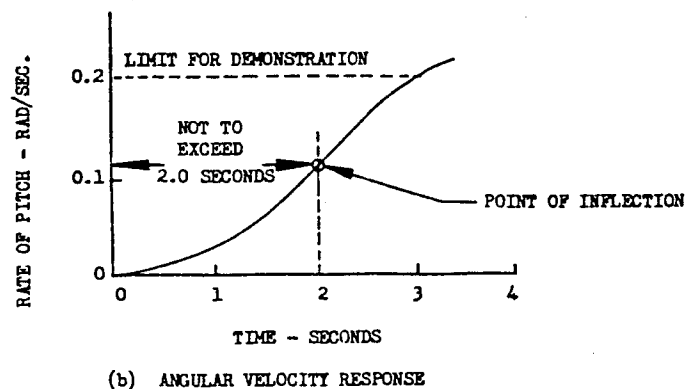
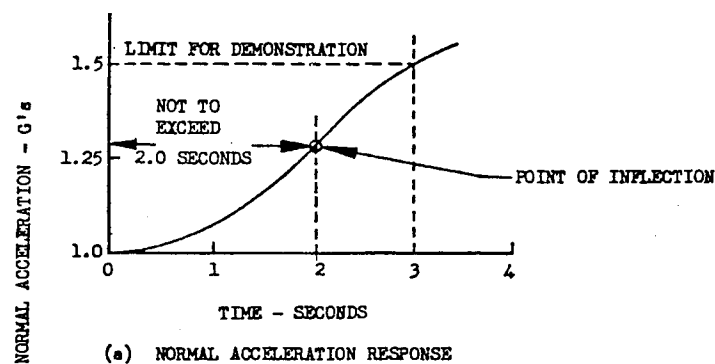


Figure 5(3.4.1.3). Typical Normal Acceleration and Pitch Rate Response. (In this Sample the Control Input was Limited by Normal Acceleration.) (from Reference 31)

This requirement was discussed in the Related Previous Requirements discussion for Paragraph 3.4.1.1, where it was shown that the concave downward limit on normal acceleration is equivalent to maneuver margin (see also Reference 32). Reference 127 shows that meeting the concave downward requirement places limits on the stability derivatives  $Z_w$ ,  $M_q$ , and  $M_w$ ; for any inflection time  $t_I \leq 2$  sec, compliance requires:

$$Z_w M_q - \mu M_w \geq 2/t_I^2$$

where  $\mu$  is the rotor advance ratio,  $V \cos \alpha / \Omega R$ . The inverse of the normal acceleration response gives the longitudinal cyclic stick displacement per g. From Reference 127, the steady-state response of the short-period approximation is

$$\frac{\theta_{1s}}{a_z/g} = \frac{g (Z_w M_q - \mu M_w)}{-\mu M_{\theta_{1s}} Z_w}$$

where  $\theta_{1s}$  is the longitudinal cyclic pitch angle and  $M_{\theta_{1s}}$  is pitch sensitivity. At low speed this gradient is a measure of pitch control sensitivity  $M_q/M_{\theta_{1s}}$ , and at high speed it is a measure of angle-of-attack stability  $M_w$ .

Combining the two equations given above yields

$$\frac{\theta_{1s}}{a_z/g} \geq \frac{g}{-\mu M_{\theta_{1s}} Z_w} \frac{2}{t_I^2}$$

and the concave downward and control displacement per g requirements are related. Additional terms can be added to this equation to relate  $\theta_{1s}$  to cockpit control force,  $F_s$ .

a. Statement of Requirement

3.4.2 Pitch Control Power. For Level 1, it shall be possible from trimmed, unaccelerated flight to achieve the steady load factor specified in the Operational Flight Envelopes. For Level 2, there shall be sufficient pitch control authority to accelerate from 45 knots (23 m/s) to the maximum level flight airspeed, and to decelerate back to 45 knots (23 m/s) at constant altitude.

b. Rationale for Requirement

This requirement simply states that there must be sufficient pitch attitude control power to maneuver the rotorcraft throughout the Operational Flight Envelope. It is required that sufficient pitch control power be available to achieve the normal load factor boundary. While there is no data to support such a requirement, it is felt that there should be at least enough pitch control power to achieve the operational load factor for helicopters that meet this specification. The Level 2 limit only requires that the helicopter remain controllable up to the maximum level flight airspeed. There are no restrictions on the use of other controllers (e.g., collective) for either Level 1 or Level 2.

c. Supporting Data

None.

d. Guidance for Application

The requirement does not specify the allowable combination of collective and cyclic controls that may be used to demonstrate compliance. It is felt that as long as the required load factor can be attained, it does not matter which controller is used. As a practical matter, the use of large collective inputs may require more pitch control power (due to coupling) than if a more conventional combination of collective and cyclic is used.

e. Related Previous Requirements

MIL-H-8501A (Paragraph 3.2.1) requires that "throughout the speed range a sufficient margin of control power, and at least adequate control to produce 10 percent of the maximum attainable pitching moment in hovering shall be available at each end to control the effects of longitudinal disturbances." The current requirement is more stringent for most, if not all conventional rotorcraft, and in no case should the requirement result in control power less than 10 percent of the pitching moment available in hover.

a. Statement of Requirement

3.4.3 Flight Path Control. The vertical rate response shall have a qualitative first-order appearance for at least 5 seconds following a step collective input. The limits on the parameters defined by the following equivalent first-order vertical-rate-to-collective transfer function are given in Table 1(3.4).

$$\frac{\dot{h}}{\delta_c} = \frac{K e^{-\tau_{\dot{h}} s}}{T_{\dot{h}} s + 1}$$

TABLE 1(3.4). MAXIMUM VALUES FOR HEIGHT RESPONSE TO COLLECTIVE CONTROLLER

LEVEL	$T_{\dot{h}}$ (sec)	$\tau_{\dot{h}}$ (sec)
1	5.0	0.20
2	$\infty$	0.30

The equivalent system parameters are to be obtained using the time domain fitting method defined in Figure 8(3.3). The coefficient of determination,  $r^2$ , shall be greater than 0.97 and less than 1.03 for compliance with this requirement.

b. Rationale for Requirement

Collective control may be used to augment longitudinal cyclic inputs for flight path control. This is especially true in near-earth operation, where collective inputs become a necessary part of the pilot's control strategy to clear obstacles while minimizing altitude overshoot and airspeed excursions. As in Paragraph 3.3.10.1 (height control at hover and low speed), this requirement effectively sets the minimum heave damping requirements for the rotorcraft. The heave damping requirements for low speed have been extended to forward flight speeds based on the flight data presented below.

c. Supporting Data

Extension of the low speed and hover requirement to forward flight is based on the variable stability Bell 205A flight tests reported in Reference 174 (see backup material for Paragraph 3.3.10.1). Those tests included a moderate speed (60-80 kt) dash which required aggressive vertical maneuvering to clear obstacles and remain close to the terrain. The values in Table 1(3.4) are supported by the pilot ratings for that forward flight task.

d. Guidance for Application

None.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.4.4 Interaxis Coupling. Control inputs to achieve a response in one axis shall not result in objectionable responses in one or more of the other axes while performing any of the Mission-Task-Elements specified in Paragraph 3.1.1. This shall hold for control inputs up to and including those required to perform any of the specified Mission-Task-Elements. Specific limits on interaxis coupling are given in Paragraphs 3.4.4.1 and 3.4.4.2.

b. Rationale for Requirement

This requirement covers all interaxis coupling not specifically addressed by 3.4.4.1 and 3.4.4.2 due to lack of data. It is identical to Paragraph 3.3.9 in the Hover and Low Speed section.

c. Supporting Data

The collective-to-attitude and pitch/roll coupling requirements of Paragraphs 3.4.4.1 and 3.4.4.2 are the only requirements for which adequate supporting data exists. Several of the lateral and directional requirements, however, are also aimed at roll/yaw coupling, especially turn coordination and lateral control in steady sideslips.

d. Guidance for Application

None.

e. Related Previous Requirements

None.



a. Statement of Requirements

3.4.4.1 Collective-to-Attitude Coupling

3.4.4.1.1 Small Collective Inputs (Less Than 20% of Full Control). The peak change in pitch attitude,  $\theta_{\text{peak}}$ , occurring within the first 3 seconds following an abrupt change in collective,  $\delta_{\text{COL}}$ , shall be such that the ratio  $|\theta_{\text{peak}}/n_{z_{\text{peak}}}|$  is no greater than 3.0 deg/m/s<sup>2</sup> (1.0 deg/ft/sec<sup>2</sup>), where  $n_{z_{\text{peak}}}$  is the peak normal acceleration.

3.4.4.1.2 Large Collective Inputs (Greater Than or Equal to 20% of Full Control). The peak change in pitch attitude,  $\theta_{\text{peak}}$ , occurring within the first 3 seconds following a step change in collective,  $\delta_{\text{COL}}$ , shall be such that the ratio  $|\theta_{\text{peak}}/n_{z_{\text{peak}}}|$  is no greater than 1.5 deg/m/s<sup>2</sup> (0.5 deg/ft/sec<sup>2</sup>) in the up direction and 0.76 deg/m/s<sup>2</sup> (0.25 deg/ft/sec<sup>2</sup>) in the down direction. In addition, during an autorotation to touchdown from any point in the Operational Flight Envelopes,

- It shall not require more than 15 percent of full (stop-to-stop) cockpit pitch controller travel or force to hold pitch attitude constant.
- There shall be at least 5 percent of the total (stop-to-stop) pitch controller effectiveness remaining throughout the maneuver.

b. Rationale for Requirements

These requirements are intended to preclude excessive coupling between the collective controller and pitch attitude. Excessive coupling will interfere with precision altitude control, and could result in loss of attitude control for large collective changes, e.g., autorotation. Paragraphs 3.4.4.1.1 and 3.4.4.1.2 have been included to insure that these deficiencies are not allowed by the specification.

c. Supporting Data

Support for these requirements comes from the V/STOL simulation study of Reference 3. Some qualitative support is also found in Reference 83.

A fixed-base simulation experiment was conducted as a part of the Reference 3 study, using a model of a fixed-wing V/STOL. While this simulation focused on pitch attitude coupling due to throttle during decelerating approaches, it is considered applicable to helicopters in forward flight, at least as a measure of the allowable coupling between pitch attitude and the primary flight path controller.

Each simulation run was divided into two tasks. The first task began at a range of 6000 ft, 50 ft above the nominal 4 deg glide slope. The mean airspeed was 80 kt, with a moderate turbulence level of 5 kt (rms). The pilot acquired and tracked the glide slope using guidance signals on a heads-up-display. A three-window computer-generated display of the landing area (a destroyer) was also available for reference. An initial deceleration to 40 kt was completed at a range of 3000 ft, with a final deceleration to hover next to a shipboard helicopter pad. The pilot assigned a separate Cooper-Harper rating for the approach tracking and final deceleration tasks. (The final deceleration task was accomplished using the visual display.) Seventeen configurations were evaluated; one set of (two) ratings was given for each configuration. Several runs were completed for each case before the ratings were assigned. Pilot comments were recorded and later transcribed. One pilot performed all evaluations, and the results should be weighed accordingly.

The Reference 3 simulation used a transfer function representation of the aircraft characteristics so that the vertical rate and pitch attitude responses could be independently varied. The tested variations in pitch attitude to a 10% altitude-rate controller are shown in Figure 1. Similarly, the tested variations in altitude-rate response to a 10% input are shown in Figure 2.

The pilot rating correlations showed that pilot opinion depended almost entirely on the height response bandwidth with little dependency on attitude coupling per se. That is, in terms of Cooper-Harper pilot rating, the collective-to-attitude coupling mattered only insofar as it affected flight path control. However, the pilot commentary does provide clues as to when the coupling itself becomes undesirable. The limits established in this requirement are based on the pilot commentary, which is excerpted in Table 1. The interpretation of this commentary is that the pitch attitude to altitude-rate-controller coupling ( $\theta_{\text{peak}}/\delta_{\text{ARC}}$ ) was not a factor for an adverse value of -0.072 deg/percent, but was worse than Level 1 at -0.23 deg/percent (see Table 1) -- "extremely objectionable". The Level 1 boundary was taken as the midway point between these values, e.g.,  $\theta_{\text{peak}}/\delta_{\text{ARC}} \geq -0.15$  deg/percent.

The pilot rating data are plotted against flight path-to-collective bandwidth in Figure 3. The collective-to-attitude coupling ( $\theta_{\text{peak}}/\delta_{\text{ARC}}$ ) is also shown on Figure 3 where it can be seen that large adverse values result in a low flight path bandwidth, and tend to be non first order looking (recall that Paragraph 3.4.3 requires the h response to look qualitatively first order for approximately 5 seconds). Physically, it can be seen that if the rotorcraft pitches in the opposite sense to the collective, in forward flight, the flight path response will be degraded. The Figure 3 data, combined with the pilot commentary related to adverse coupling in Table 1 indicate that degraded flight path control (low bandwidth and non first order response) is the limiting factor. On that basis, a separate limit on adverse coupling does not seem necessary, although it is accounted for in the final criterion, as discussed below.

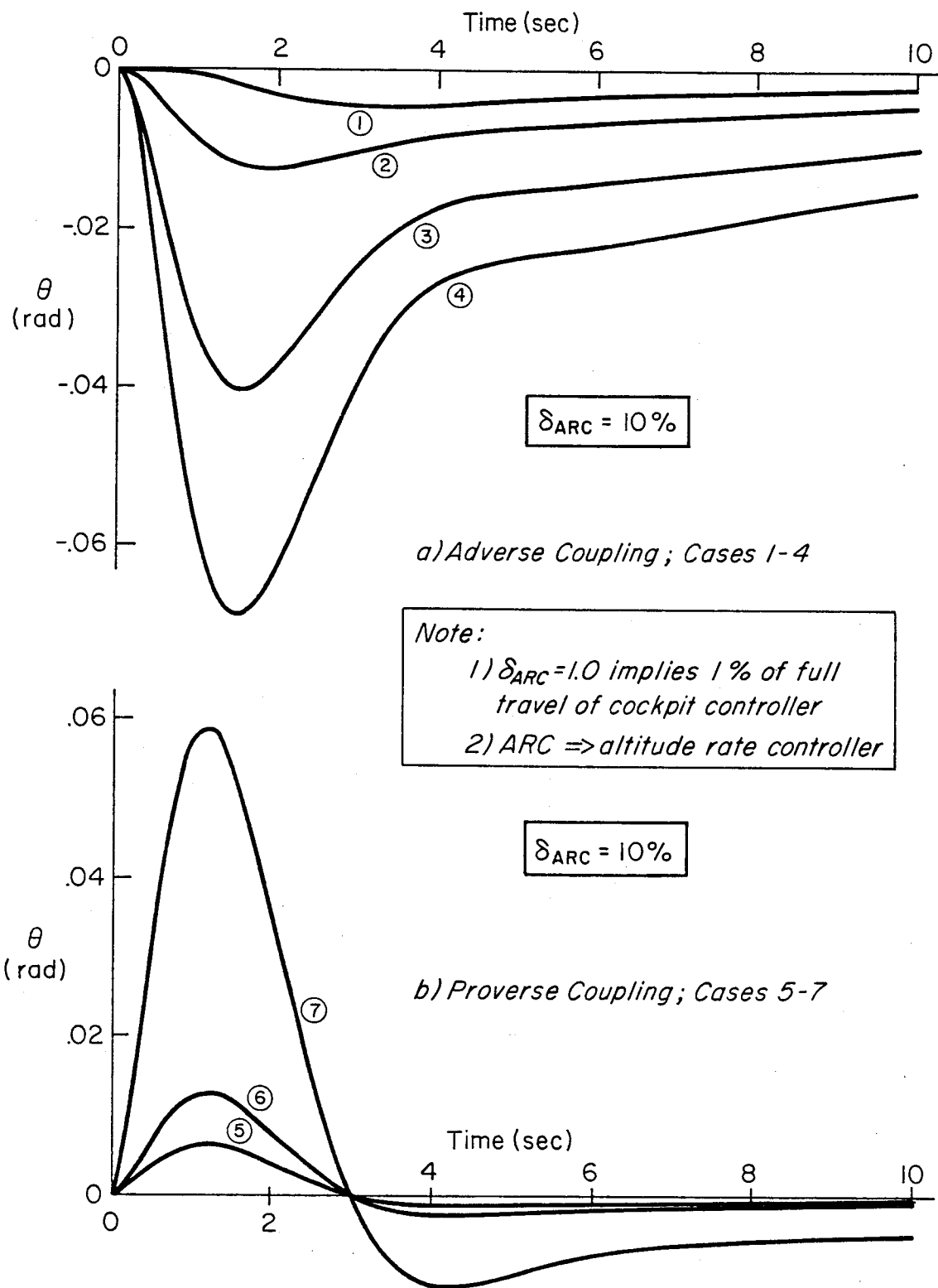


Figure 1(3.4.4.1). Pitch Attitude Responses to Step Altitude Rate Controller Input for  $\theta/\delta_{ARC}$  Cases for Table 1

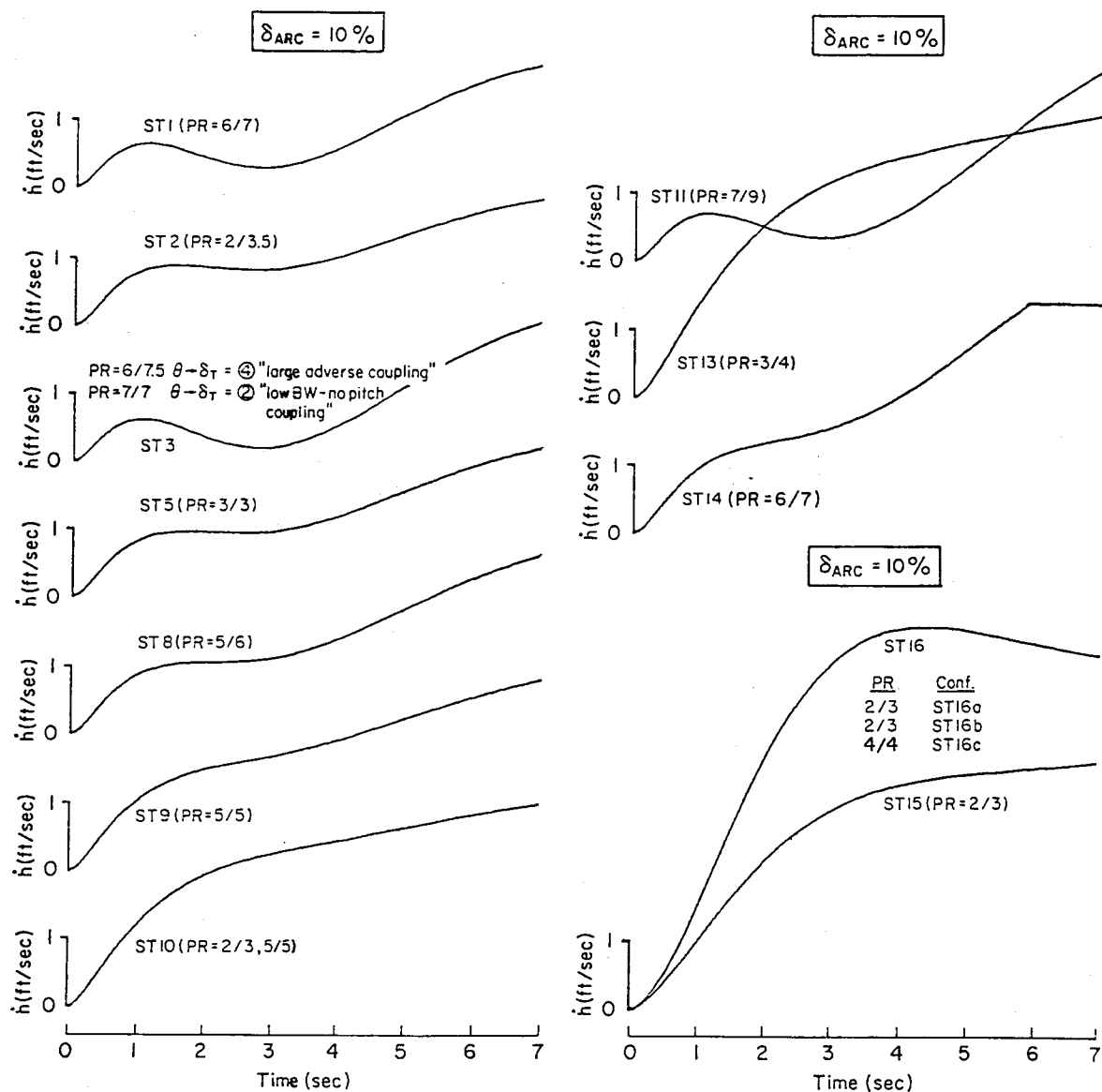


Figure 2(3.4.4.1). Altitude Rate Responses to Step Altitude Rate Controller Inputs for Configurations Evaluated in Reference 3 Simulation. Cooper-Harper Pilot Ratings (PR) are Noted Next to Responses

TABLE 1(3.4.4.1). EXCERPTS FROM PILOT COMMENTARY REGARDING  
PATH/ATTITUDE COUPLING FOR FIXED-BASE  
SIMULATION OF REFERENCE 3

$\theta/\delta_{\text{ARC}}$ deg/%	CASE NO.	PILOT COMMENTS
-0.023 ① *	ST 10	I do not see any coupling in pitch (first run) there was very little attitude coupling that I could perceive (repeat run).
	ST 13	I did not see any noticeable coupling in attitude.
	ST 15	I did not see any pitch coupling that time.
-0.072 ② *	ST 2	The coupling is pretty low on this one. I had a little bit of adverse coupling that was mildly unpleasant but not a big factor.
	ST 3a	Very minimal pitch coupling.
-0.23 ③ *	ST 11	That had some real bad adverse coupling. I found this to be extremely objectionable.
-0.40 ④ *	ST 1	The adverse pitch coupling on that is pretty extreme. The pilot compensation was pretty extreme so a pilot rating of 6 on the glide slope. Inside 200 ft (deceleration to hover) the pilot rating gets a little worse because the pitch coupling is a lot more noticeable (PR = 7). Pilot technique is to use power-to-stick crossfeed to cancel coupling but the workload is very high. This results in adequate performance.
0.034 ⑤ *	ST 16a	The pitch coupling is in a desirable direction.
0.072 ⑥ *	ST 16b	It is almost better than no pitch coupling at all.
0.34 ⑦ *	ST 16a	The pitch coupling is getting to be a little bit excessive. Pitch coupling is a minor but annoying deficiency making it a pilot rating of 4 (on glide slope and near the ship).

\*Numbers in circle refer to attitude coupling cases shown in Figure 1

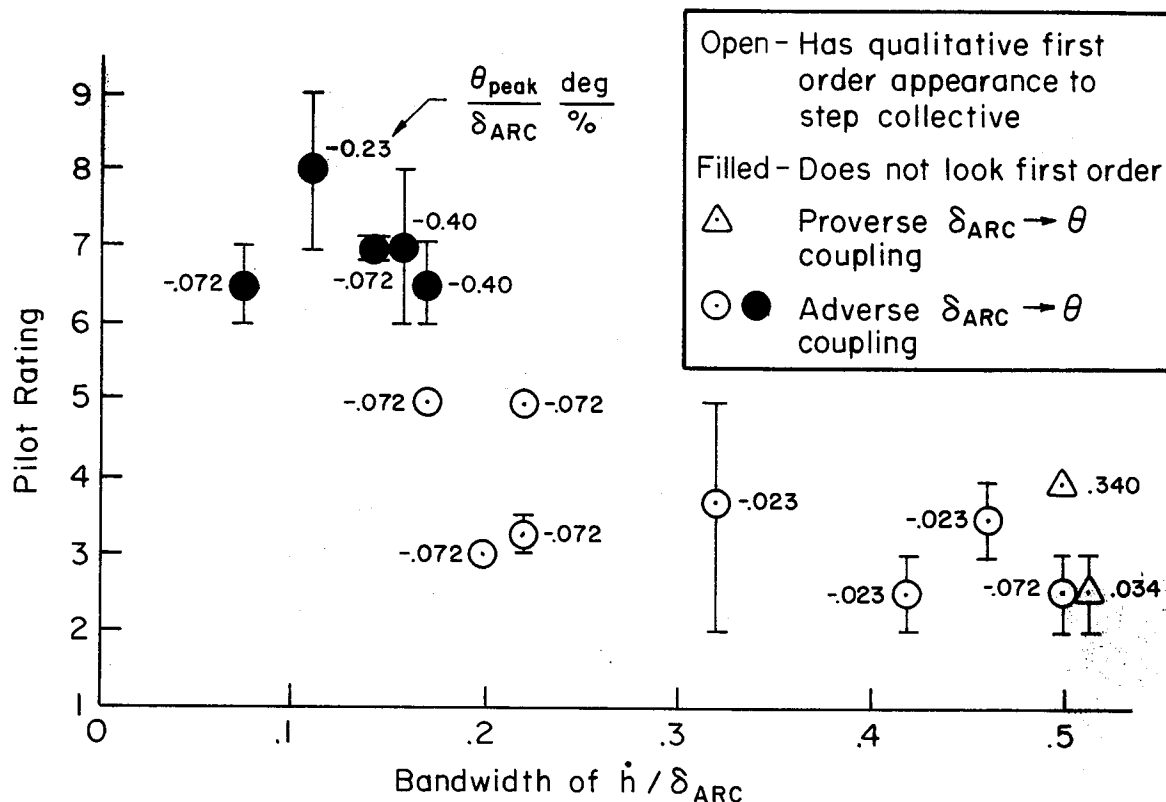


Figure 3(3.4.4.1). Effect of  $\delta_{ARC} \rightarrow \theta$  Coupling and  $\dot{h}/\delta_{ARC}$  Bandwidths on Pilot Ratings from Reference 3 Experiment

Two proverse coupling cases were tested in Reference 3, both with the same (good) flight path bandwidth, but with a factor of 10 difference in  $\delta_{ARC} \rightarrow \theta$  coupling. The pilot ratings (HQRs) degraded from 2/3 to 3.5/4 when  $\theta_{peak}/\delta_{ARC}$  was increased from .034 to 0.34 deg/percent (see Figure 3). The related pilot commentary indicate that the increased coupling was a "minor but annoying deficiency" (HQR=4). Taking this as the Level 1 limit yields a value of .35 deg/percent.

The percent-of-control-travel is not an effective normalizing variable to generalize the results of Reference 3, as it is influenced by the geometry of the controller (i.e., VSTOL throttle vs. collective). The peak normal acceleration ( $n_{z_{peak}}$ ) resulting from a step altitude-rate-controller input was selected to normalize the results, as it yields a parameter which represents the unwanted response divided by the desired response ( $\theta_{peak}/n_{z_{peak}}$ ). The peak normal accelerations were estimated from the maximum slopes of the altitude-rate responses in Figure 2 resulting in a change in the value of the Level 1 coupling boundary from .35 deg/percent to 1.0 deg/ft/sec<sup>2</sup>. One result of normalizing with respect to peak normal acceleration is to penalize adverse coupling. For example, adverse coupling case 2 exhibits a significantly lower value of  $\theta_{peak}/\delta_{ARC}$  than proverse coupling case 7 in Figure 1. Yet, the value of  $\theta_{peak}/n_{z_{peak}}$  for ST 2 (which uses adverse coupling,

case 2) is  $-0.80 \text{ deg/ft/sec}^2$  compared to  $+1.0 \text{ deg/ft/sec}^2$  for ST 16 (which uses proverse coupling, case 7).<sup>\*</sup> On the basis of these results, a limit of  $1 \text{ deg/ft/sec}^2$  has been selected for proverse and adverse coupling.

Comments from industry have indicated that  $1.0 \text{ deg/ft/sec}^2$  is excessively lenient. These comments are based on experience with previous rotorcraft, and inhouse simulations, and probably reflect the effects of large collective inputs. Since the Reference 3 data only apply to small amplitude tracking, it seems reasonable to tighten the requirement for large controller inputs (taken as greater than 20% of full travel) to values recommended by the manufacturers. One manufacturer suggested a value of  $-0.25 \text{ deg/ft/sec}^2$  for down collective and  $0.50 \text{ deg/ft/sec}^2$  for up collective, and another recommended  $0.10 \text{ deg/ft/sec}^2$  for both directions. The former recommendation has been adopted as it seems reasonable, i.e., a 0.5g collective change would result in a peak attitude change of 4 degrees for down collective, and 8 degrees for up collective. The signs (negative attitude for down and positive attitude for up collective) probably relate to the fact that conventional helicopters pitch down for down collective and up for up collective. An absolute value has been used in the criterion to account for possible non-standard coupling.

The manufacturers also recommended that the proposed cockpit control travel limit to hold pitch attitude constant in an autorotation was too high (25% of full (stop-to-stop) travel). That limit was taken from the old MIL-H-8501A. A reduction to 15% was suggested based on the manufacturer's experience, and has been adopted herein. In addition to total travel, it is important to insure that the longitudinal cyclic controller does not hit a limit during the autorotation. Therefore a 5% pad on pitch controller effectiveness has been incorporated.

#### d. Guidance for Application

The tests required to determine the coupling involve rapid changes in collective from one constant value to another, with the cockpit pitch controller free. If allowing the cockpit pitch controller to be free results in a pitch divergence (due to a reversible control system) it may be held fixed at the trim value. The data for autorotations could be obtained in a non-obtrusive way during testing to determine the height-velocity envelope.

#### e. Related Previous Requirement

None.

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<sup>\*</sup>Case numbers refer to the level of collective-to-attitude coupling, and are denoted as circled numbers in Figure 1 and Table 1. The "ST" configurations refer to the vertical rate response in Figure 2.

a. Statement of Requirement

3.4.4.2 Pitch-to-Roll and Roll-to-Pitch Coupling During Aggressive Maneuvering. The following requirement applies for the Ground Attack, Slalom, Pull-Up/Push-Over, Assault Landing, and Air Combat Mission-Task-Elements. The ratio of peak off-axis response to desired response,  $\theta_{pk}/\phi$  ( $\phi_{pk}/\theta$ ), following an abrupt lateral (longitudinal) cyclic step input, shall not exceed the limits specified in Table 2(3.4) for at least 4 seconds after the input is initiated. This shall hold for control input magnitudes up to and including those required to perform the specified Mission-Task-Elements.

TABLE 2(3.4). MAXIMUM VALUES FOR PITCH-TO-ROLL  
AND ROLL-TO-PITCH COUPLING

PARAMETER	LEVEL 1	LEVEL 2
$(\theta_{pk}/\phi)\delta_A$ or $(\phi_{pk}/\theta)\delta_B$	$\pm 0.25$	$\pm 0.60$

b. Rationale for Requirement

Some amount of pitch-to-roll and roll-to-pitch coupling is unavoidable in most helicopters. The significance of this coupling on handling qualities depends upon the mission: for the most aggressive maneuvers, only a relatively small level of coupling can be tolerated. It is desirable to derive limits on pitch/roll coupling that are functions of both MTE and response frequency, but there is simply not enough data to do so. Instead, the small amount of data available for this Paragraph and for the similar low-speed requirement, Paragraph 3.3.9.2, were analyzed to develop an upper limit on coupling in the first five seconds following the control input.

The parameters used here were chosen for two reasons: 1) they are easy to obtain from flight test data, and 2) they are related to the classical coupling parameters,  $L_q/L_p$  and  $M_p/M_q$ , as the discussion for Paragraph 3.3.9.2 shows.

c. Supporting Data

References 83 and 87 contain supporting data for the coupling requirements. In addition, some insight can be obtained from the study reported in Reference 162, though no handling qualities ratings are available.



Reference 87 reports on a flight experiment using the NASA Ames UH-1H helicopter. The task flown consisted of s-turns around markers spaced along a runway at a speed of 60 kts and an altitude of 100 ft. The pitch-to-roll and roll-to-pitch coupling evaluated, represented by the ratios of damping derivatives,  $M_p/M_q$  and  $L_q/L_p$ , respectively, was set at 0, 0.25, and 0.50, with  $M_q$  constant at -2 rad/sec and  $L_p$  varied from -2 to -8 rad/sec. Two levels of yaw rate damping,  $N_r = -1.2$  and -3.5 rad/sec, were used; the latter represents the better value, so the results analyzed here will be for this level of damping. The coupling parameters required for this Paragraph were calculated using these values of derivatives in combination with the derivatives given in Reference 28 for the UH-1H at 60 kts. There is, as a result, some possibility for error, since the analytical models of Reference 28 do not always accurately represent the level of coupling present (see, for example, the discussion for Paragraph 3.3.9.2). According to Reference 87, however, the variable-stability system of the UH-1H was adjusted to match the Reference 28 derivatives.

Figure 1 shows the variation in handling qualities ratings with pitch-to-roll coupling; this parameter was chosen because the slalom task is primarily lateral in nature, and thus it would be expected that unwanted pitch attitudes due to lateral cyclic inputs would be the dominant coupling. There is not a very strong trend in ratings with coupling, with only a slight degradation for the highest level of coupling. These data do suggest that coupling causes some degradation at the highest level, and the Level 1 limit at 0.25 reflects this trend more than the actual data.

It is interesting to note that the baseline (uncoupled) cases of Reference 87 received Level 2 average ratings, even though the roll bandwidths were Level 1 (Figure 1). Figure 2 shows all of the cases investigated in the Reference 87 experiment on a crossplot of roll bandwidth and control sensitivity. The results for the uncoupled cases are discussed in supporting data for both the small-amplitude (Paragraph 3.4.5.1) and large-amplitude (Paragraph 3.4.5.3) requirements. The point of interest here is that the coupling cases, with the nominal UH-1H control power, were Level 2 because of both bandwidth and control power. For the highest bandwidth (2.8 rad/sec), the best uncoupled configuration received an HQR of 3.5 from one pilot. If the data of Figure 1 for this bandwidth (triangles) were shifted downward so that the uncoupled case is at an HQR of 3.5, the case with the highest coupling would be at an HQR of approximately 5, or solidly Level 2. This level of coupling is halfway between the Level 1 and 2 limits, lending some support to these limits.

The Reference 83 study used the NASA Ames S-19 fixed-base simulator with a combined camera/model visual system to simulate nap-of-the-earth flight. Rotor system dynamics were varied to produce the desired variations in coupling, and to vary the damping derivatives  $M_q$  and  $L_p$ . Three different courses were flown: a longitudinal dolphin course of 50-ft-high barriers irregularly spaced between 700 and 1400 ft apart; a lateral-directional slalom course of trees approximately 75 ft high spaced in a straight line similar to the dolphin course; and a combination course. Airspeeds varied between 50 and 90 kts. The coupling

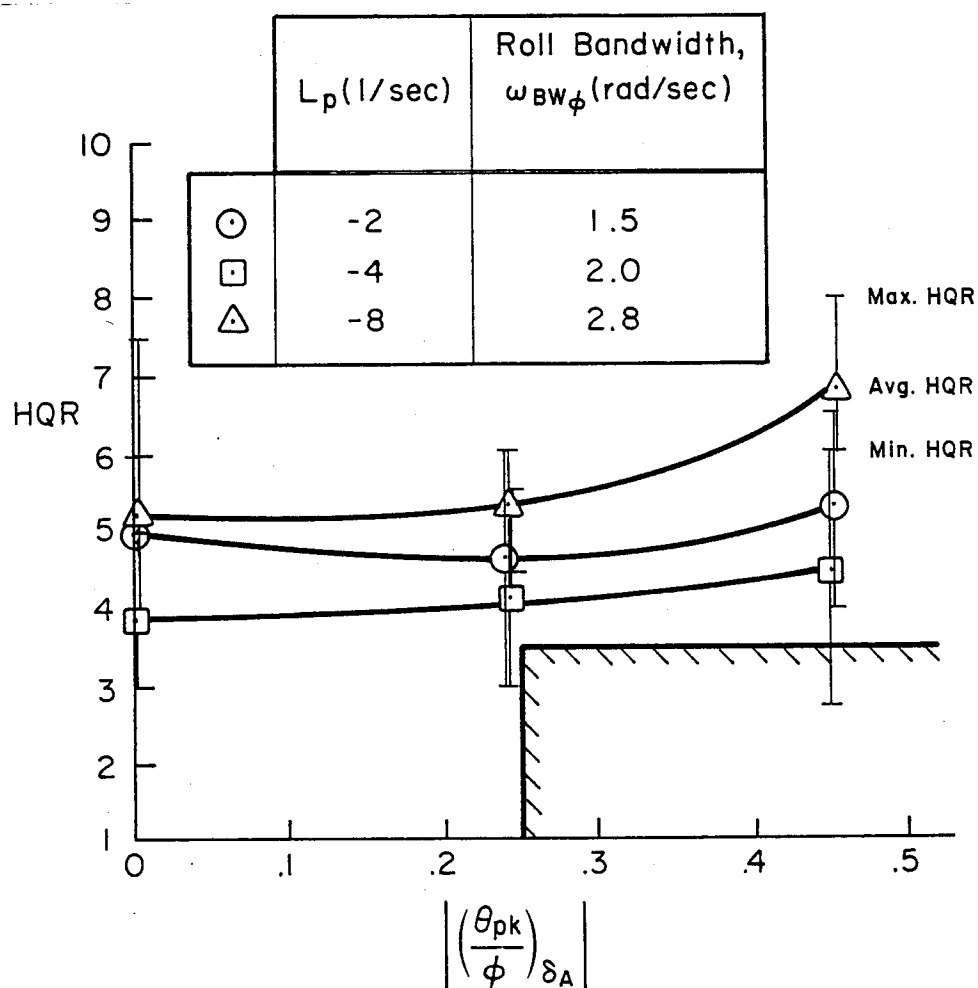


Figure 1(3.4.4.2). Variation in HQR with Pitch-to-Roll Coupling for Flight Experiment of Reference 87 (Control Sensitivity Fixed at Non-optimum Value)

evaluations used the combination course.

Two pilots participated in the Reference 83 simulation and flew the task at different speeds. Pilot A, who flew at relatively low speed, generally assigned Level 1 and 2 ratings to the coupling configurations, while Pilot B, who performed the task at higher speeds, gave Level 2 and 3 ratings. Figure 3 shows the handling qualities ratings from the two pilots as a function of the pitch-to-roll coupling parameter (estimated from derivatives given in Reference 83). The trend lines indicate that Pilot B was essentially unaware of any degradation in handling qualities from what he perceived as poor handling qualities to start with (i.e., he gave Level 2 ratings even for the lowest levels of coupling). While there is not strong correlation between coupling level and HQR for either pilot, the ratings of Pilot A at least indicate that low levels of coupling are not objectionable, and that noticeable handling qualities degradations occur for coupling values above about 0.25.

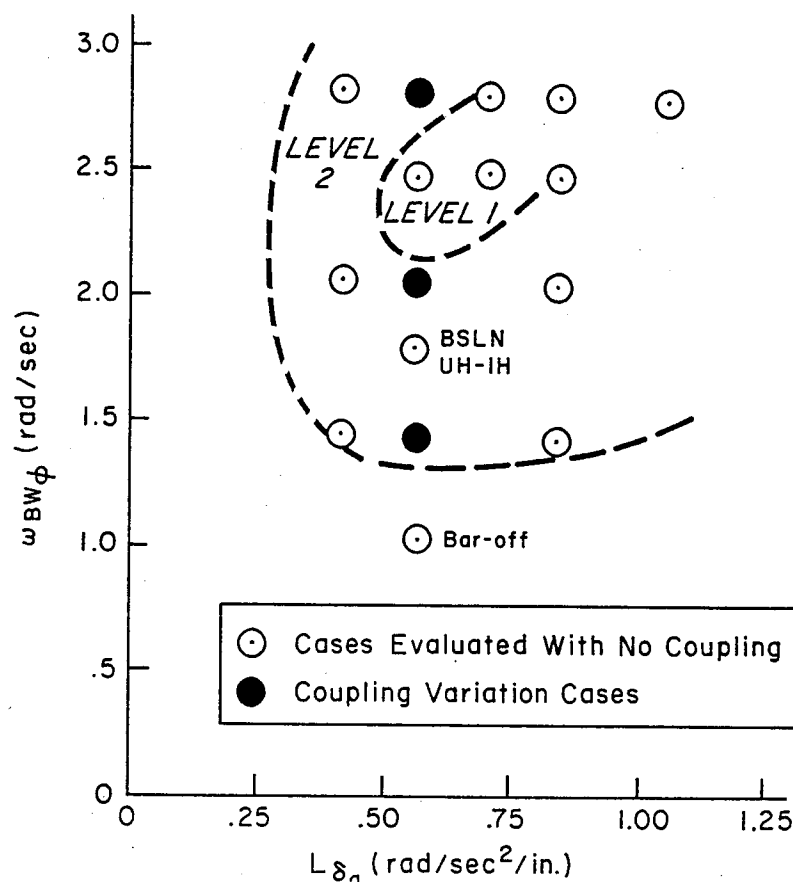


Figure 2(3.4.4.2). Evaluation Cases for Reference 87 Flight Experiment and Trends in Flying Qualities Level for Uncoupled Cases (Paragraph 3.4.5.1)

The final source for coupling information comes from the Reference 162 evaluations of a rudimentary hingeless-rotor helicopter. No pilot ratings are available, but an analysis of pilot comments provides some insight. The characteristics of this helicopter are described in the supporting data for Paragraph 3.3.9.2. The coupling was quite large, with  $(\phi_{pk}/\theta)\delta_R = 0.9$  and  $(\theta_{pk}/\phi)\delta_A = -1.2$ , both clearly Level 3 by this Paragraph. For evaluations at approximately 65 kts, Reference 162 reports that "The cross-coupled pitch response to a roll control input is objectionable to the pilot. The pitch up with a left roll (and pitch down with a right roll) resulted in undesirably large normal-acceleration and flight-path changes if control correction was not applied.... The pilots considered that accurate control of pitch attitude during a roll maneuver was difficult because of the large magnitude of the coupling.... [On the other hand,] The coupling of roll with pitching velocity in forward flight was considered by the pilots to be of negligible magnitude." The latter conclusion is as expected since the pilots evaluated the helicopter at constant speed -- i.e., pitch inputs were minimal.

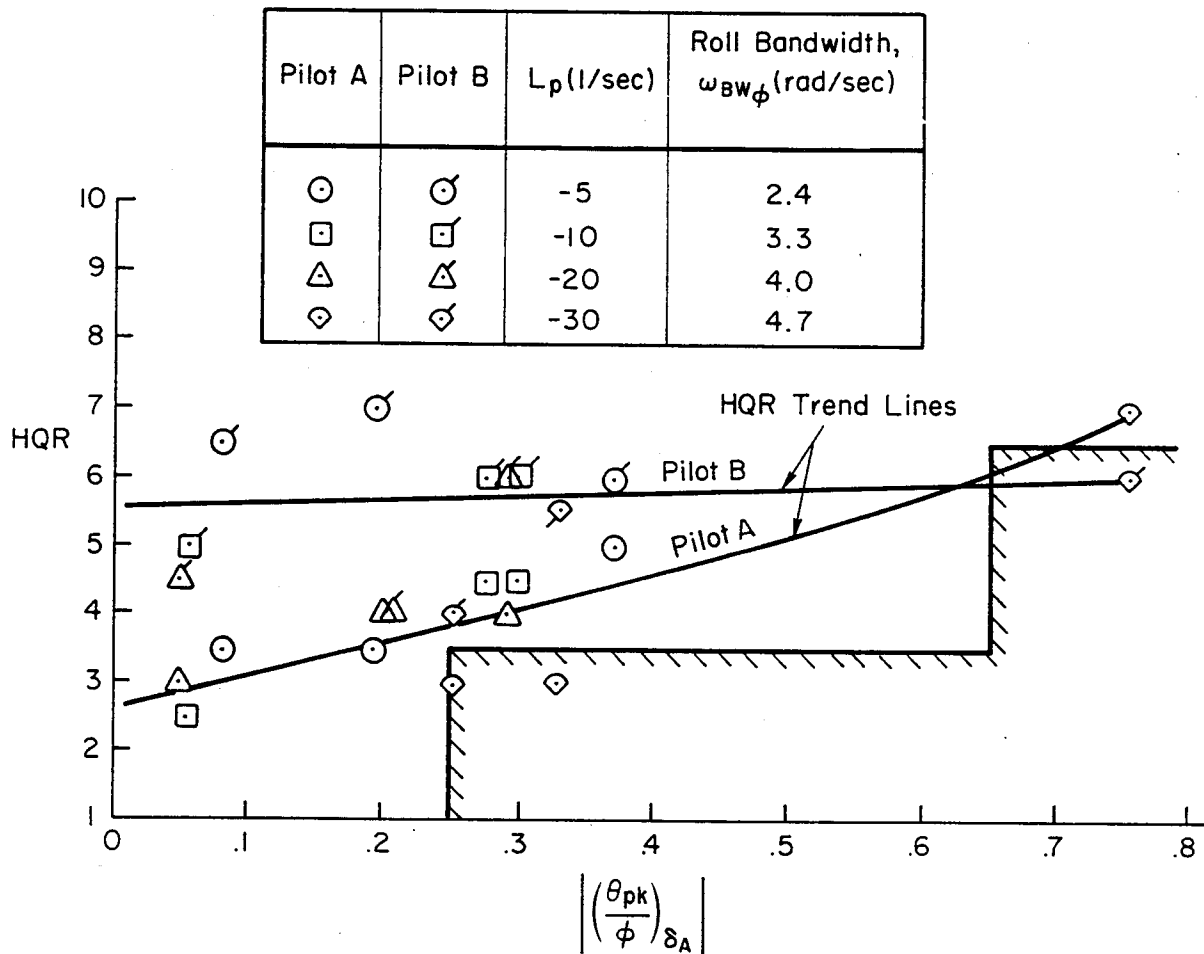


Figure 3(3.4.4.2). Pilot Ratings from Fixed-Base Simulation of Reference 83 (Combination Dolphin/Slalom Course; Control Sensitivity Varied with  $L_p$ )

For this requirement, where little supporting data is available, it would be desirable to analyze the characteristics of as many operational helicopters as possible. It is difficult, however, to find the time-history data necessary to obtain the required parameters; and a review of the analytical models of Reference 28 shows that these derivative-based models do not accurately represent the actual aircraft. A graphic example of this is given in the supporting data for Paragraph 3.3.9.2.

d. Guidance for Application

This Paragraph is identical in form to the low-speed requirement, Paragraph 3.3.9.2; the discussion there should be consulted.

e. Related Previous Requirements

While there are no requirements in the V/STOL or helicopter specifications concerning coupling, investigators in the past have developed recommended limits based on the ratio of the coupling and damping derivatives,  $L_q/L_p$  and  $M_p/M_q$ . For example, the authors of Reference 164 recommend a ratio less than 0.3 for satisfactory handling qualities and 0.5 for acceptable handling qualities, based primarily on the investigations of References 161 and 162. The authors of References 83 and 87, from which the supporting data for this Paragraph came, suggest similar limits. In simple theory (as shown in the discussion for Paragraph 3.3.9.2), the derivative ratios are approximately equal to the response ratios used in this Paragraph, provided the additional control coupling ( $M_{\delta_A}$  and  $L_{\delta_B}$ ) is negligible.

a. Statement of Requirement

3.4.5 Roll Attitude Response to Lateral Controller

3.4.5.1 Small-Amplitude Roll Attitude Response to Control Inputs (Bandwidth). The roll attitude response to lateral cockpit control force or position inputs shall meet the limits specified in Figure 2(3.4). The bandwidth ( $\omega_{BW\phi}$ ) and phase delay ( $\tau_{p\phi}$ ) parameters are obtained from frequency responses as defined in Figure 2(3.3).

It is desirable to meet the requirement for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish.

Any oscillatory modes following a pulse controller input shall meet the requirements of Figure 9(3.4).

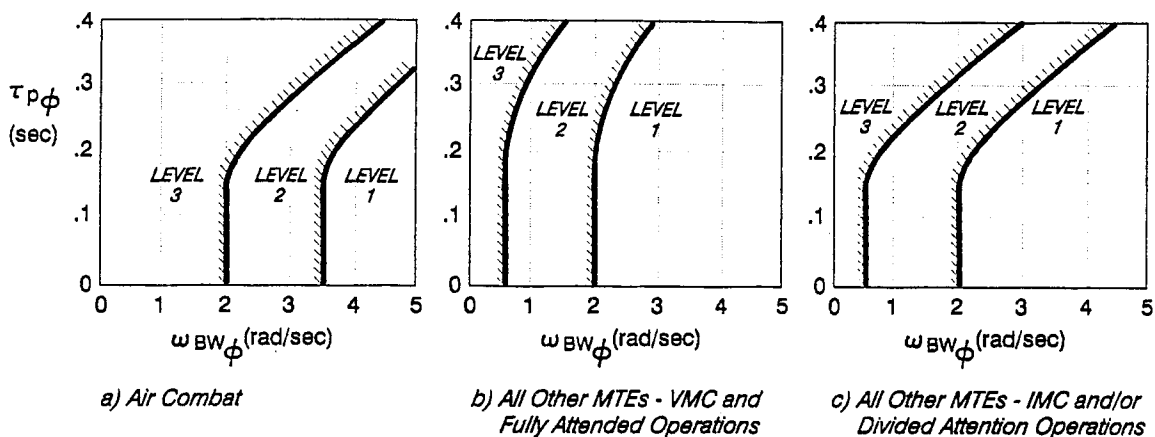


Figure 2(3.4). Requirements for Small-Amplitude Roll Attitude Changes -- Forward Flight

b. Rationale for Requirement

The theory of manual control (Reference 116) states that for continuous, precise control, the pilot demands a high response bandwidth. For larger, and hence less frequent and less precise, inputs, the pilot control structure changes, and lower response bandwidths will be acceptable. When the response requirements become very large, the pilot is not so much concerned about bandwidth as with control power. This structure of piloted control was verified in the simulation study of Reference 93, and is reflected in the structure of the roll response requirements.

For short-term, fine maneuvering, the primary parameter of interest to the pilot is the rapidity of the response. This is quantified throughout the specification in terms of the response bandwidth. The requirements for small-amplitude maneuvering in this Paragraph are written explicitly in terms of the bandwidth frequency, defined in Figure 2(3.3) of the specification. For larger responses, where bandwidth may not be so easy to determine due to nonlinearities, the requirements use the ratio of peak roll rate divided by attitude change, which is related to bandwidth frequency (Paragraph 3.4.5.2). Finally, for the limiting case, limits are placed on the minimum roll rate (or, for Attitude Response-Types, bank angle) achievable (Paragraph 3.4.5.3).

Three sets of limits are specified in Figure 2(3.4). The more stringent limits, Figure 2a(3.4), apply to the Air Combat MTE, and the more relaxed boundaries of Figure 2b(3.4) cover all other MTEs. For divided attention operations (or specifically IMC flight) the more relaxed bandwidth boundaries (Figure 2b(3.4)) are combined with the more stringent  $r_p$  requirement (Figure 2a(3.4)). The Air Combat limits are based on a simulation study of yaw requirements for air combat, discussed in the Supporting Data for Paragraph 3.4.7.1, and support for the limits for other MTEs is presented below. These limits are identical to Figures 1b(3.3) and 1d(3.3), respectively. Supporting data for Paragraph 3.3.2.1 provides more information on these limits.

#### c. Supporting Data

Several flight test and ground simulation programs have been aimed at determining the control/response requirements for helicopters in low-level, high-speed flight. As with most requirements in this specification, the most consistent supporting data comes from the flight experiments.

The flight tests of Reference 87 used the Ames UH-1H helicopter for most of the evaluations. A limited number of runs were made with an AH-1G and an OH-6A. The primary variables were roll control sensitivity and damping. Some variations were made in the level of yaw damping,  $N_r$ , and in cross-coupling, represented by the ratios  $M_p/M_q$  and  $L_q/L_p$ . For the purposes of this discussion, only the data with zero cross-coupling are of interest. The evaluation task (Figure 1) was a slalom flown at Crows Landing Naval Air Station, California, around ground markers spaced 1000 ft apart along a runway. The pilots were instructed to fly at a reference airspeed of 60 kt and altitude of 100 ft. The ground markers did not extend to flight altitude, so the pilots were instructed to turn about imaginary piers located at the markers. In addition, the task was specifically designated as a series of S-turns. Both of these factors should be kept in mind when other flight test data are discussed below.

Pilot rating results are shown on a plot of  $\omega_{BW_\phi}$  vs.  $L_{\delta_a}$  in Figure 2. Actual bandwidths for the test configurations are not known, so the bandwidth values used in Figure 2 are based on derivatives given in Reference 87 and an equivalent time delay,  $\tau_e$ , of 0.17 sec -- a reasonable value for the UH-1H in forward flight. The Cooper-Harper

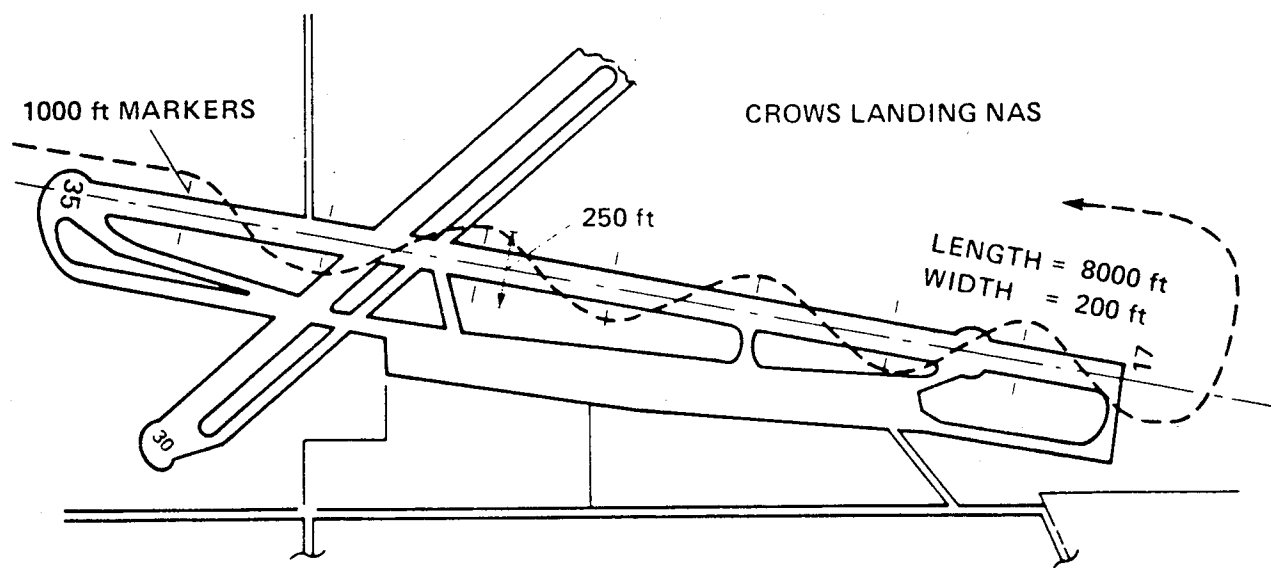
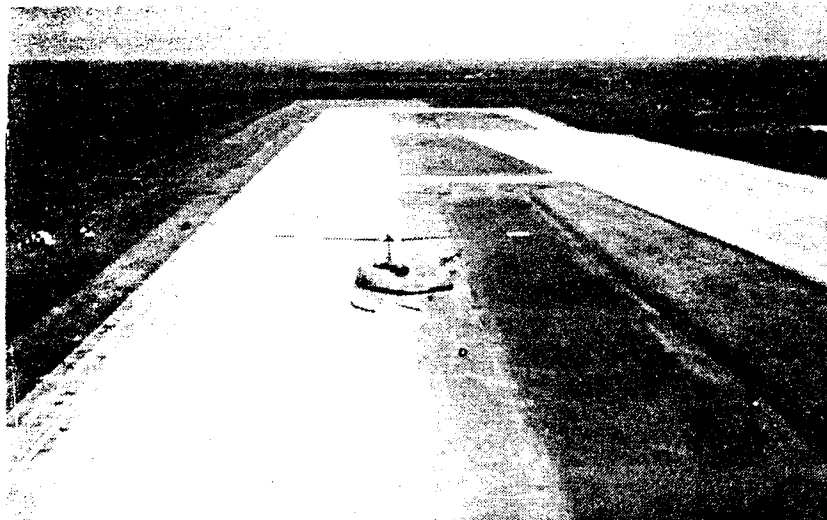


Figure 1(3.4.5.1). Slalom-Course Task for Flight Tests of Reference 87. Nominal Altitude, 100 ft AGL; Airspeed, 60 kt (from Reference 87)



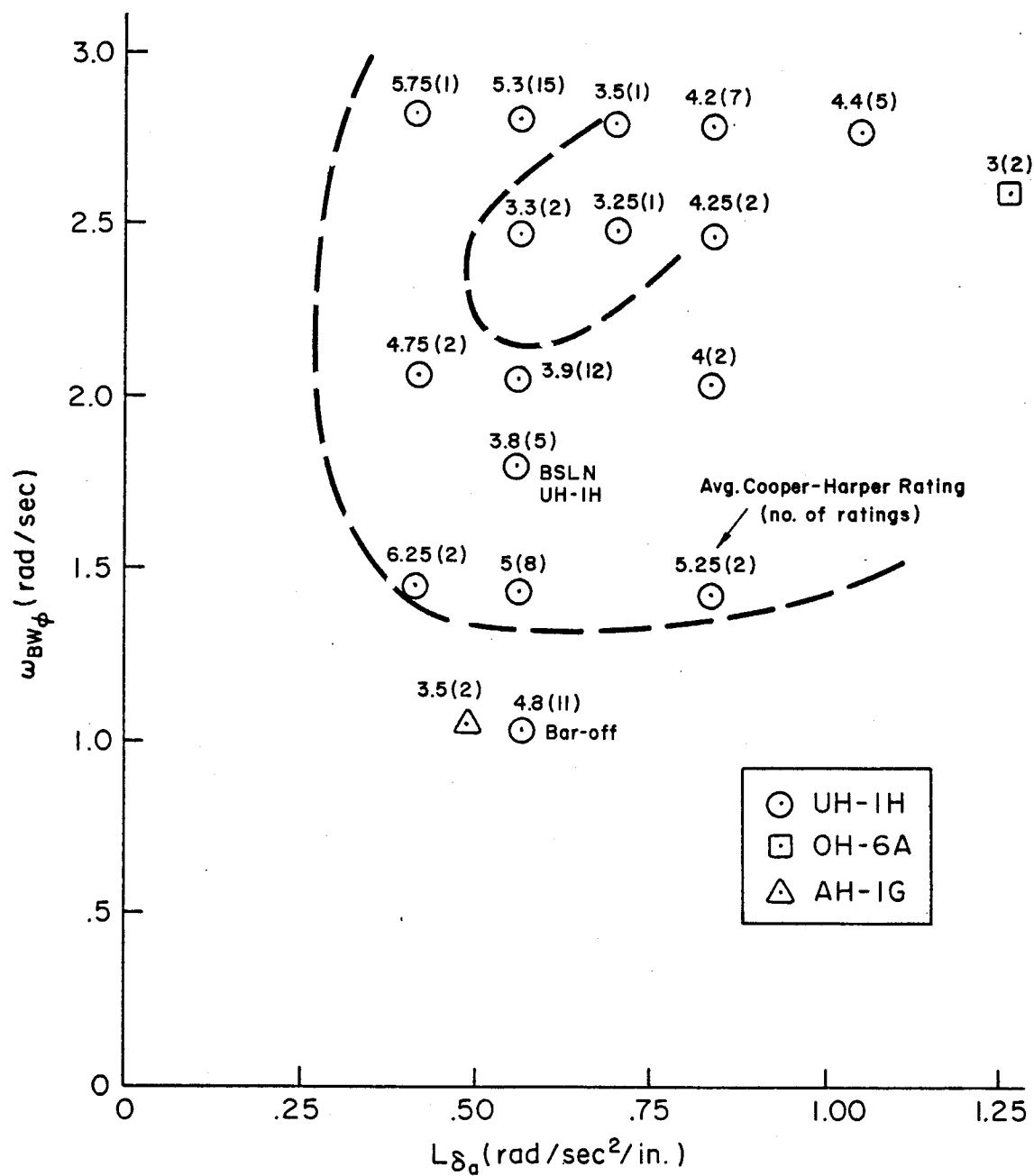


Figure 2(3.4.5.1). Pilot Ratings from NASA Army Flight Tests of Reference 87. No Cross-Coupling;  $N_r = -3.5$  1/sec

ratings given represent averages from one or more of four pilots; the number of ratings for each point is given in parentheses. The ratings show that a bandwidth of about 2.1 rad/sec is necessary for Level 1.

A series of flight tests were conducted by the German Aerospace Laboratories (DFVLR) in parallel with the U.S. tests of Reference 87. Results of some of these tests are documented in References 88 and 89. In Reference 88, pilot ratings are given for two pilots performing the slalom task shown in Figure 3. The tests were conducted at an airspeed of 60 kt and an altitude of 30 ft, rather than the 100 ft used in Reference 87. In addition, the pilots attempted to fly the shortest path between ground markers. This is very different from the Reference 87 tests, where S-turns were used, and would be expected to demand higher control sensitivity, and perhaps higher bandwidths as well.

Figure 4 summarizes the pilot ratings from Reference 88, again with an estimated value of  $\omega_{BW}$  (using the derivatives of References 88 and 28, and an assumed delay of 0.17 sec). Average ratings are shown for two pilots; individual ratings are not available from Reference 88. These data support the observation from Figure 2 that a bandwidth of around 2 rad/sec is necessary for Level 1. Considerably more control sensitivity ( $L_{\delta}$ ) was required in the Reference 88 tests, however, reflecting the different nature of the task. It is significant that the bandwidths required were not substantially different, indicating the constancy of bandwidth for these tasks.

In the flight tests reported in Reference 89, two helicopters (the DFVLR BO-105 and UH-1D) were flown through two courses by three evaluation pilots. The slalom course is similar to that used in the NASA tests of Reference 87 (Figure 1); this course was flown at altitudes of 100 ft (replicating the NASA tests) and 30 ft. In addition, a "German Slalom" (jinking) course, shown in Figure 5, was flown around 10-m (33-ft) obstacles at 30 ft. This course required the pilots to fly along the centerline at 60 kt until committed to turn around the obstacles. Response characteristics of the test helicopters were not varied. Overall pilot ratings were obtained using the Cooper-Harper rating scale, and from two modified scales for pilot stress and task performance. A summary of pilot ratings is given in Figure 6.

The ratings of Figure 6 show little variation between the 100-ft and 30-ft slalom. Ratings are degraded somewhat for the jinking task. The ratings for the BO-105 in the slalom were included on Figure 4 and are reasonably consistent with the DFVLR data of Reference 88.

The Level 2 bandwidth limit of 0.5 rad/sec is based on the data of Figure 2 and needs to be evaluated further.

Figure 7 shows the bandwidths of several operational helicopters as a function of airspeed. Most of the helicopters meet the Level 1 requirement.

All four of the helicopters in the Level 2 region of Figure 7 have some form of augmentation system; the AH-1G is in the Level 2 region

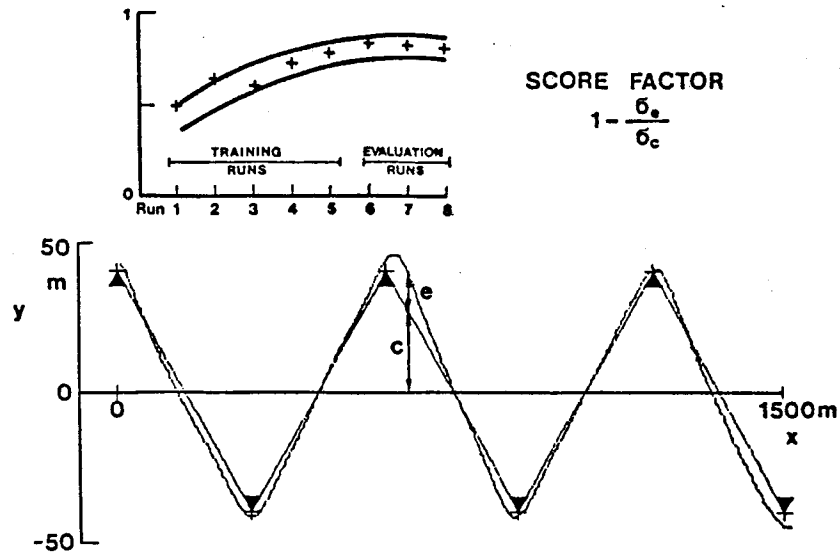


Figure 3(3.4.5.1). Slalom Task Used in DFVLR BO-105 Flight Tests (Reference 88)

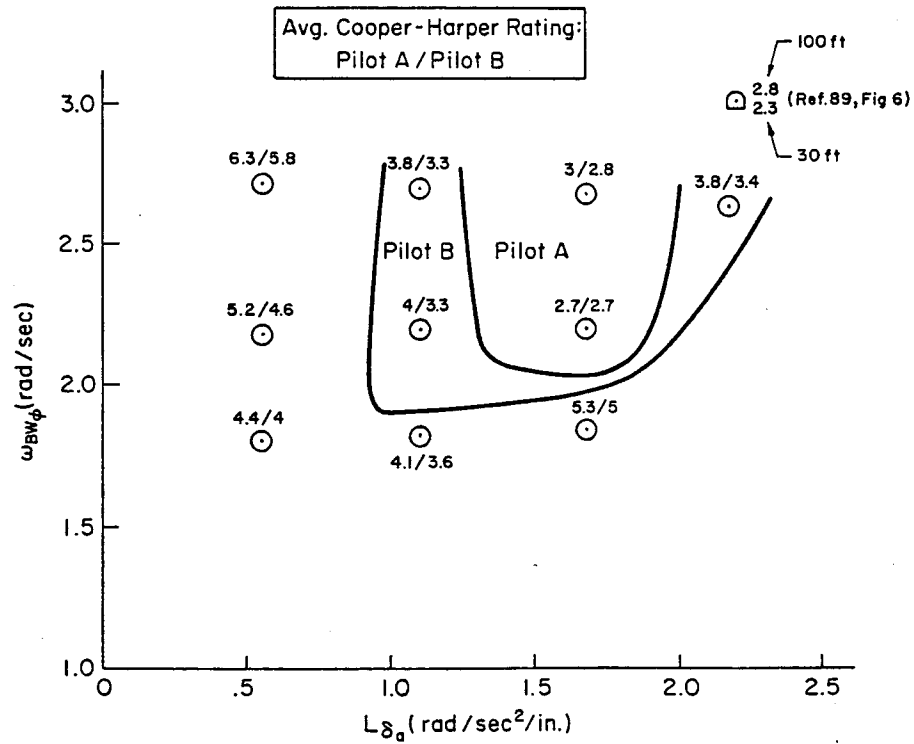


Figure 4(3.4.5.1). Pilot Ratings from DFVLR Tests (References 88 and 89)

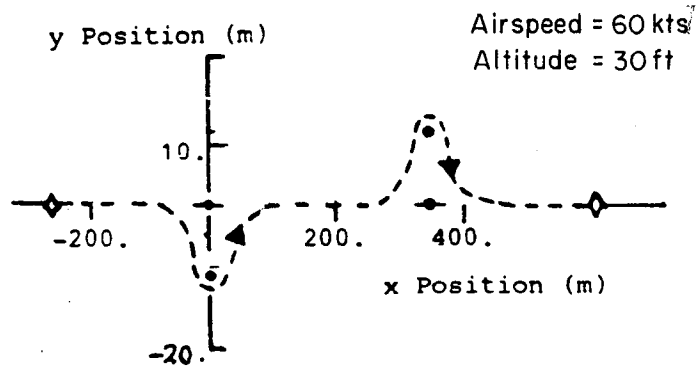


Figure 5(3.4.5.1). "German Slalom" (Jinking) Course  
from Reference 89 Flight Tests

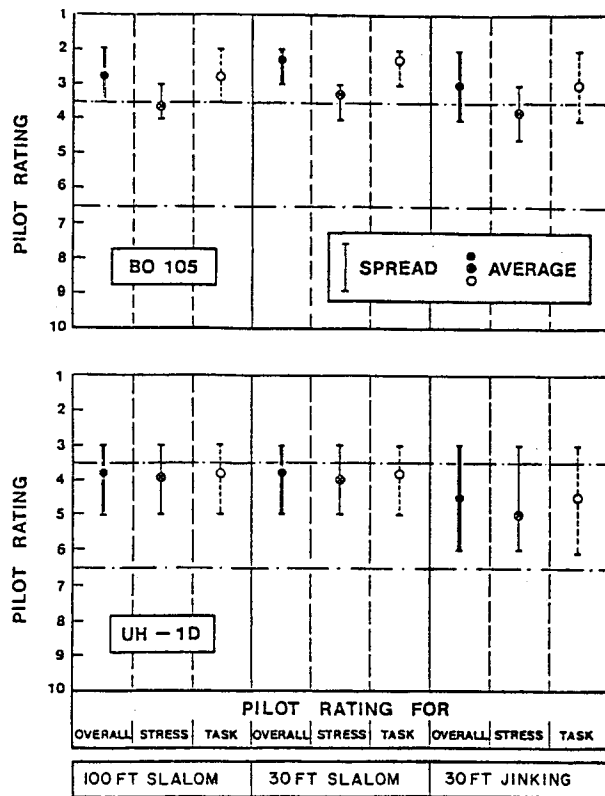


Figure 6(3.4.5.1). Pilot Rating Summary from  
DFVLR Flight Tests (Reference 89)

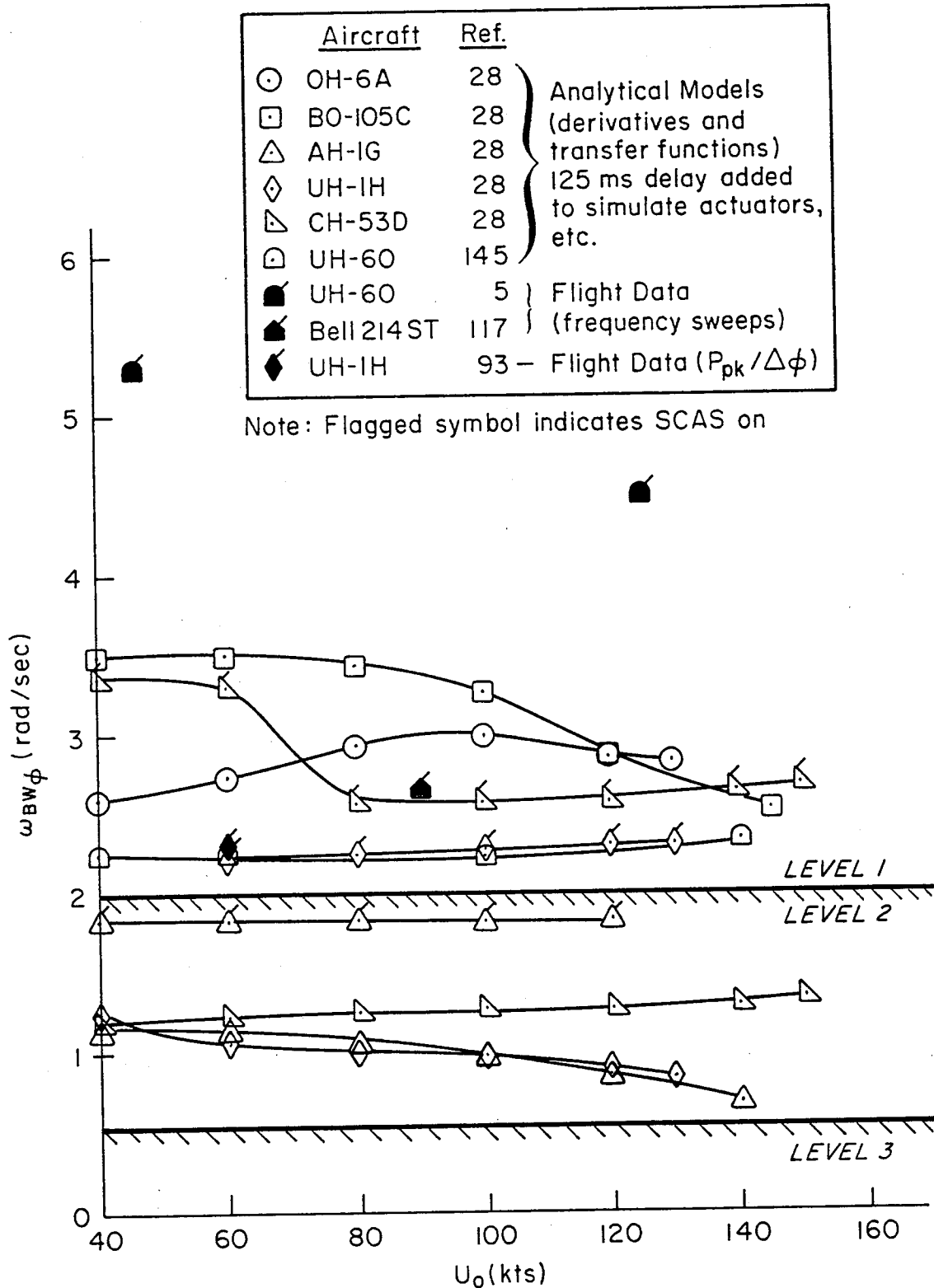


Figure 7(3.4.5.1). Roll Attitude Bandwidths of Operational Helicopters

SCAS-on as well. Flight test evaluation reports of the AH-1G (Reference 75) and the related YAH-1R (Reference 68) do not indicate any short-term roll response shortcomings in this helicopter, suggesting that the bandwidths plotted in Figure 7 for the AH-1G may be somewhat lower than the actual helicopter's bandwidth.

d. Guidance for Application

The discussion for Paragraph 3.3.2.1 should be consulted. Appendix A describes a method for obtaining the bandwidth parameters from flight data.

e. Related Previous Requirements

The helicopter specification MIL-H-8501A (Reference 31) does not have requirements on roll bandwidth. Instead, limits are placed on the angular velocity attained at the end of 1/2 sec following a 1-in. cyclic input, and on angular rate damping, at hover only (there is no forward-flight requirement). Both MIL-F-8785C (Reference 66) and MIL-F-83300 (Reference 29) specify roll damping in terms of the roll mode time constant  $T_R$ . Neglecting time delay, and assuming the Dutch roll mode is well-damped, for a conventional, unaugmented helicopter, the roll rate transfer function is approximated as

$$\frac{p}{\delta_A} = \frac{L_{\delta_A}}{(s + 1/T_R)}$$

for which  $\omega_{BW\phi} = 1/T_R$ . For a step  $\delta_A$  input, the time response of this system is given by

$$p(t) = T_R L_{\delta_A} (1 - e^{-t/T_R})$$

Limits on angular velocity are essentially limits on the term  $T_R L_{\delta_A}$ , which is control power in deg/sec (covered by Paragraph 3.4.5.2). In reality, however, time delays are very large, and SCAS is provided, so these approximations do not hold. Bandwidth and control power are more comprehensive than angular velocity in 1/2 sec and roll rate damping since they account directly for the effects of these time delays and SCAS elements.

a. Statement of Requirement

3.4.5.2 Moderate-Amplitude Attitude Changes (Attitude Quickness). The ratio of peak roll rate to change in bank angle,  $p_{pk}/\Delta\phi_{pk}$ , shall exceed the limits specified in Figure 3(3.4). The required attitude changes shall be made as rapidly as possible without significant reversals in the sign of the cockpit control input relative to the trim position. The initial attitudes, and attitude changes, required for compliance with this requirement, shall be representative of those encountered while performing the required Mission-Task-Elements (Paragraph 3.1.1). The parameters in Figure 3(3.4) are defined in Figure 4e(3.3).

b. Rationale for Requirement

This requirement is equivalent to the low-speed requirement, Paragraph 3.3.3. It allows a reduction in bandwidth for moderate attitude changes. Two sets of requirements are given, for Air Combat (Figure 3a(3.4)) and for all other MTEs (Figure 3b(3.4)).

The limits on  $p_{pk}/\Delta\phi_{pk}$  specified in Figure 3(3.4) are identical to the low-speed requirements of Paragraph 3.3.3, with one exception: the Level 1 limit for Air Combat, Figure 3a(3.4), is more stringent than the low-speed boundary in Figure 4a(3.3). This is a reflection of the higher required roll rate in forward flight (90 deg/sec, Table 3(3.4), as opposed to 50 deg/sec, Table 1(3.3)). Justification for this extension of the low-speed requirements is threefold: 1) the small-amplitude bandwidth requirements in roll, 3.3.2.1 and 3.4.5.1, are identical and are both well-supported by flight data; 2) there is support for the Figure 4(3.3) boundaries, as discussed in Supporting Data for Paragraph 3.3.3; 3) there is some evidence that similar limits are required in forward flight.

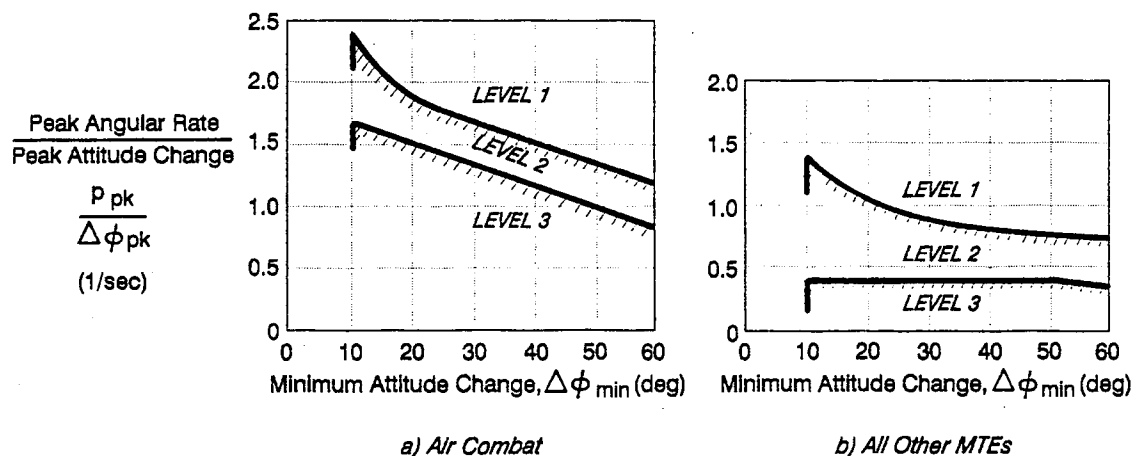


Figure 3(3.4). Roll Response Limits for Moderate-Amplitude Roll Attitude Changes -- Forward Flight

c. Supporting Data

The supporting data for Figure 3a(3.4) come from the roll-control study of Reference 93, described in detail in the Supporting Data discussions for Paragraphs 3.3.2.1 and 3.4.5.3. Maneuver performance data, in terms of peak roll rate,  $p_{pk}$ , vs. bank angle change,  $\Delta\phi$ , are given there for a variety of aircraft and tasks from both flight test and ground simulations. A review of this data shows that the most aggressive task, in terms of  $p_{pk}/\Delta\phi$ , is the HUD tracking task. Figure 1 shows a plot of  $p_{pk}/\Delta\phi$  vs.  $|\Delta\phi|$  for two configurations for this task (from simulation). Both sets of data indicate that the task did not require a high roll bandwidth at high bank angles.

d. Guidance for Application

See discussion for Paragraph 3.3.3.

e. Related Previous Requirement

None.



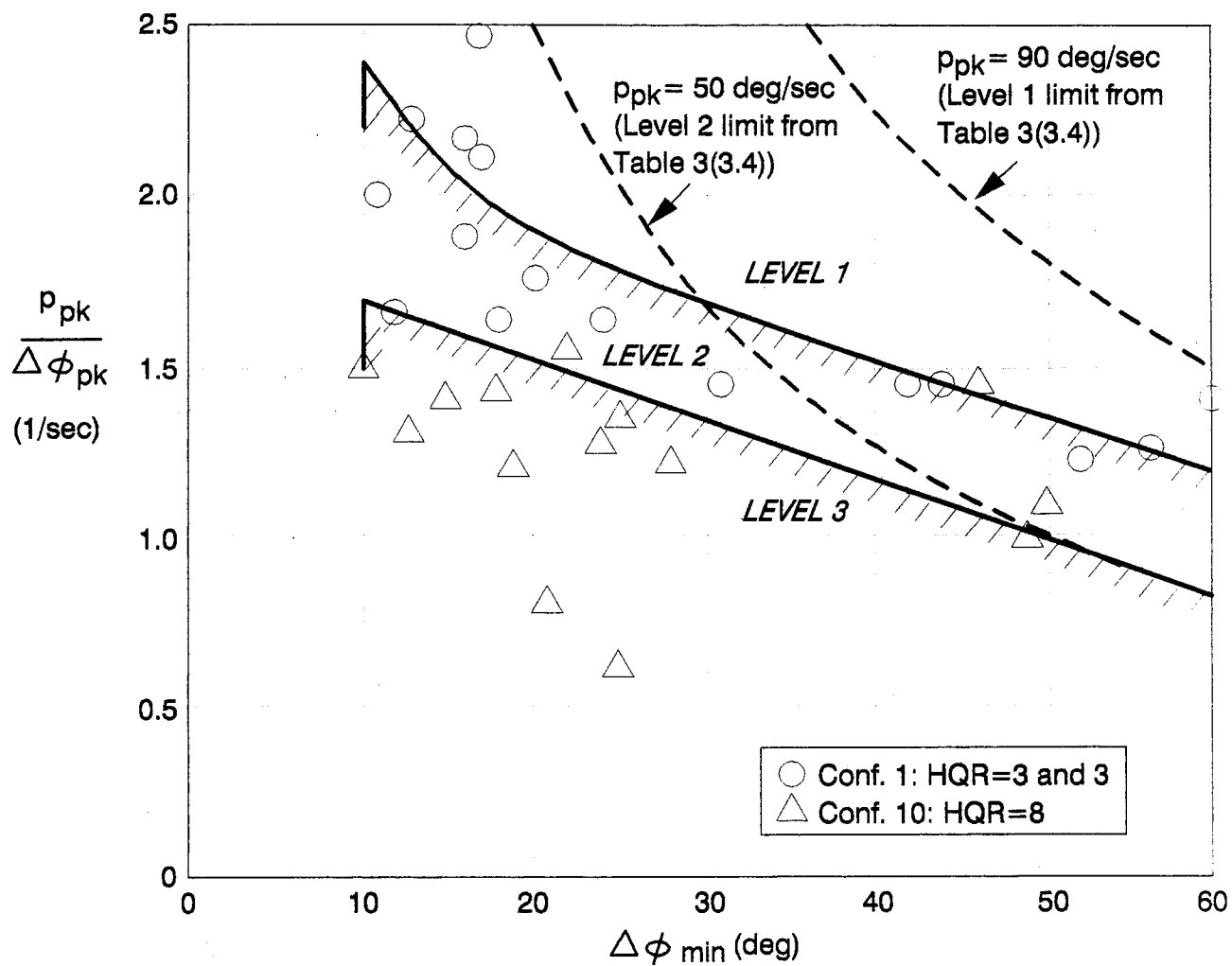


Figure 1(3.4.5.2). Roll Maneuver Performance Data for HUD Tracking (VMS Simulation, Reference 93)

a. Statement of Requirement

3.4.5.3 Large-Amplitude Roll Attitude Changes. The minimum achievable roll rate (for Rate Response-Types) or attitude change from trim (for Attitude Response-Types) shall be no less than the limits specified in Table 3(3.4). Yaw control may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate).

TABLE 3(3.4). REQUIREMENTS FOR LARGE-AMPLITUDE  
ROLL ATTITUDE CHANGES

MISSION-TASK-ELEMENT (MTE)	RATE RESPONSE-TYPES		ATTITUDE RESPONSE-TYPES	
	MINIMUM ACHIEVABLE ROLL RATE (deg/sec)		MINIMUM ACHIEVABLE BANK ANGLE (deg)	
	LEVEL 1	LEVEL 2	LEVEL 1	LEVEL 2
<u>Limited Maneuvering</u> All MTEs not otherwise specified	30	15	25	15
IMC MTEs	15	12	25	15
<u>Aggressive Maneuvering</u> Ground Attack Slalom Pull-Up/Push-Over	50	21	90	30
Assault Landing  Air Combat	90	50	Unlimited	60

b. Rationale for Requirement

This requirement, in combination with the bandwidth requirements of Paragraph 3.4.5.1 and the moderate-amplitude response requirements of Paragraph 3.4.5.2, sets the agility and maneuverability of rotorcraft. It is, therefore, a critically important requirement in the forward flight regime.

The various flying qualities specifications use differing measures for defining control power or effectiveness. For example, both MIL-F-83300 (Reference 29) and MIL-F-8785C (Reference 66) specify control effectiveness in terms of time to achieve a certain bank angle change, while the old helicopter specification MIL-H-8501A (Reference 31)

required a minimum bank angle change for a given time (1/2 second). The latter has been justly criticized (e.g., Reference 30) as being difficult to test because it is highly dependent on the pilot's ability to apply a rapid input.

In reviewing both the supporting data and previous requirements for this paragraph, it became clear that the most important response parameter from the pilot's standpoint is the ability to achieve large roll rates and bank angles in the minimum time. The time required to achieve a desired roll rate is dictated primarily by the lateral-directional dynamic characteristics, covered in Paragraphs 3.4.5.1 and 3.4.5.2. Those requirements assure that the response is suitably rapid so that the only remaining parameters of interest are the roll rate and roll angle achieved.

The requirement for a minimum bank angle at full lateral stick is aimed at insuring sufficient bank angle control with Attitude Response-Types. It is expected that, in some cases, the large bank angles required for certain Mission-Task-Elements may dictate the use of a Rate Response-Type because of the excessive control sensitivity resulting from an Attitude Response-Type.

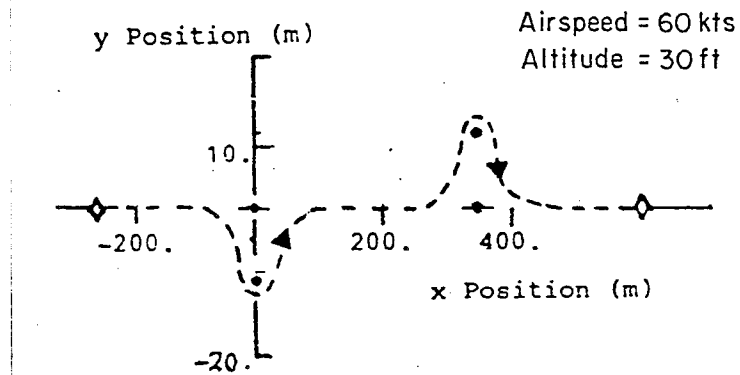
#### c. Supporting Data

The supporting data for each of the control power requirements in Table 3(3.4) are given in the following subsections.

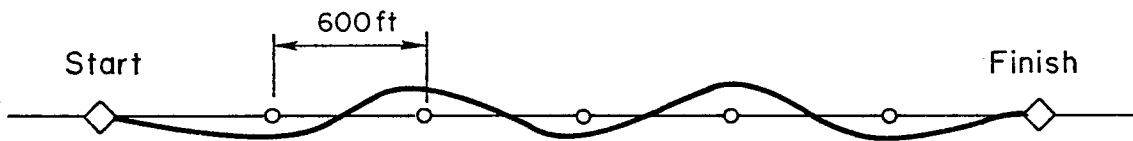
#### **NONSPECIFIED MISSION-TASK-ELEMENTS**

The data used to develop the bandwidth requirement (Paragraph 3.4.5.1) were also used here to develop the requirements for nonspecified MTEs. The slalom tests conducted in Reference 87 were flown at 60 kts airspeed and 100 ft altitude over ground markers (Figure 1c) and are therefore considered to be relatively non-aggressive.

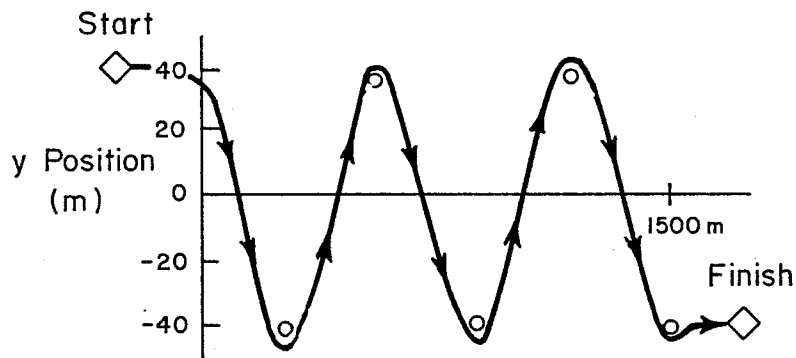
The Reference 87 data are plotted on Figure 2. Values for bandwidth are estimates using the derivatives given in Reference 87 and assuming a time delay of approximately 170 msec (approximating rotor lags, fuel system, etc.). Dashed lines are drawn to indicate tentative separation of Level 1 and Level 2 handling qualities (the Level 2 line is an extrapolation, since no Level 3 pilot ratings were given, and is therefore not supported by the data). Most of the data of Reference 87 are for the UH-1H V/STOLAND helicopter with a total lateral cyclic control throw of 12.6 in. (Reference 92), or a displacement from center of 6.3 in. The lines of constant steady-state roll rate drawn in Figure 2 are for this displacement, assuming a constant control response. For the two other data points, from the OH-6A and the AH-1G, pss is stated next to the symbols (using control throws given in Reference 28). It can be seen in this figure that the boundaries separating the Level 1 and 2 regions correspond approximately to constant levels of roll control effectiveness.



a) "German Slalom" (Jinking) Course Used for the DFVLR Evaluations (Repeated at Crows Landing by NASA/Army)



b) In-Line Slalom Flown at NASA Ames



c) U.S. Slalom Flown at Crows Landing (by NASA/Army) and at DFVLR

Figure 1(3.4.5.3). Definition of Slalom Courses

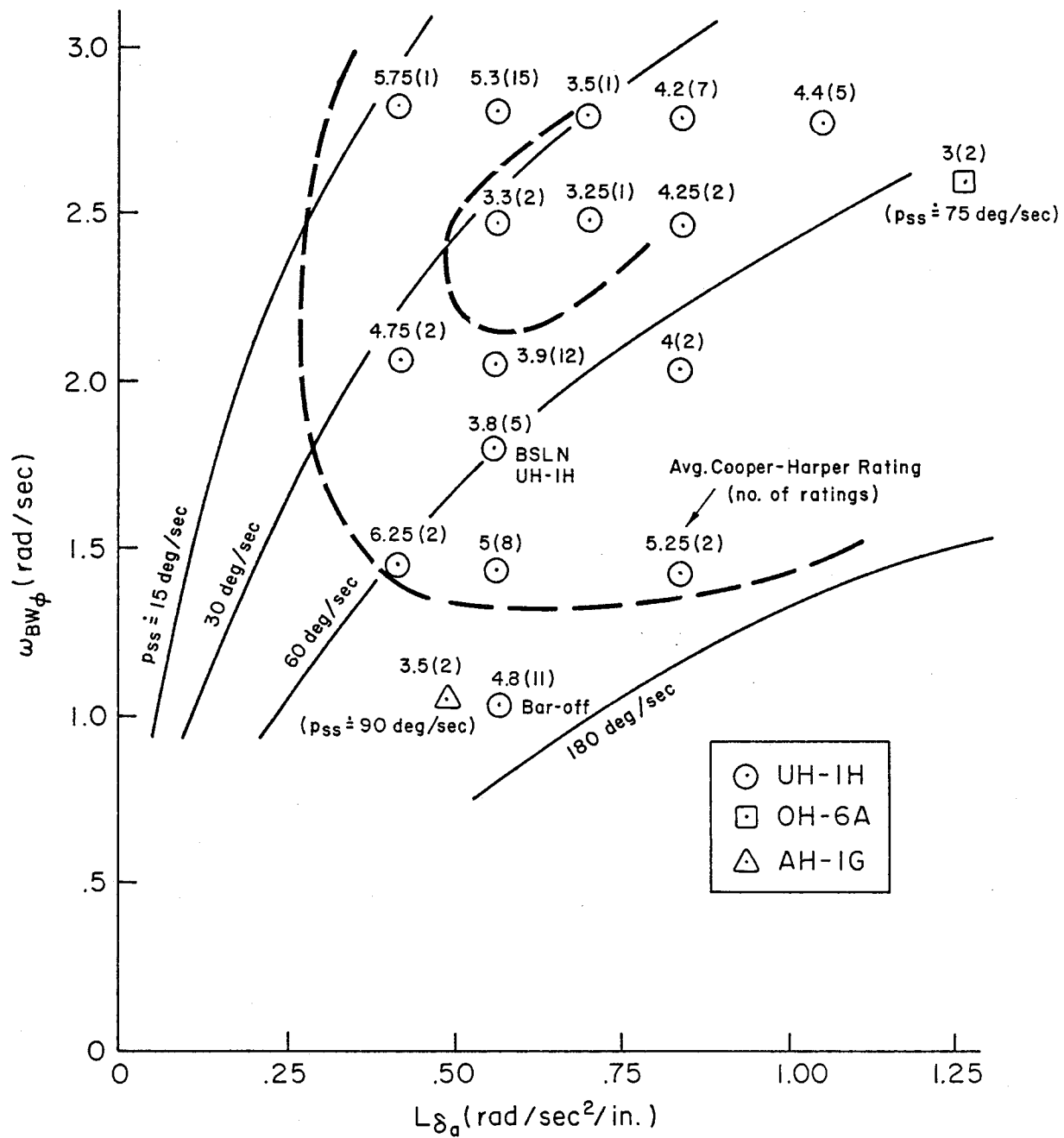


Figure 2(3.4.5.3). Steady-State Roll Rates Suggested by Level 1 and 2 Boundaries from Flight Data of Reference 87 (60-kt Slaloms at 100 ft Altitude, Around Ground Markers; Figure 1c)

A course identical to Figure 1c was flown at DFVLR in West Germany using the UH-1D and BO-105. The peak roll rates and bank angle changes observed during the DFVLR experiment are given in Figure 3 (taken from Reference 93). These data are for the basic UH-1D and BO-105 (average HQR = 3.8 and 2.8, respectively, for the task) and support the requirement for a minimum roll rate of 30 deg/sec.

The Figure 3 data also form the basis for setting the minimum achievable bank angle for Attitude Response-Types at 25 deg. Figure 3 shows bank angle change, not absolute bank angle; however, the slalom task involves attitude changes from a nominal level-flight bank angle (approximately zero degrees), and in the absence of detailed time histories for the Reference 93 maneuvers, we have assumed that bank angle achieved for the slalom task is one-half of total bank angle change.

The Level 2 roll control power limits are based on the ability to make a turn in IMC, as discussed below. Based on this data, a roll rate of at least 15 deg/sec and a bank angle of at least 15 deg are required for Level 2 in Table 3(3.4).

#### IMC MISSION-TASK-ELEMENTS

Only very limited roll control power is required for IMC operations. For example, Figure 4, taken from Reference 93, shows that a roll rate of 15 deg/sec and a bank angle change of 22 deg ( $\phi_{\max}$  of 11 deg) are adequate for an IMC turn. These values have been included in Table 3(3.4) as the Level 1 limits. The bank angle limit has been increased to 15 deg to allow an additional margin. Considering the lack of supporting data, the Level 2 limits are based on undocumented experience with fixed-wing aircraft control system development, which shows that roll rates less than 12 deg/sec are unacceptable. A relaxation in minimum bank angle below 15 deg is not felt to be warranted.

#### GROUND ATTACK, SLALOM, PULL-UP/PUSH-OVER, ASSAULT LANDING

The requirements for roll control power for these MTEs are based on several flight tests conducted at Ames Research Center and DFVLR.

A series of NASA/Army UH-1H flight tests was conducted in support of the Reference 93 roll control study. Several aggressive maneuvers were flown resulting in roll rates of up to 40 deg/sec and bank angles of 60 deg. Both of these values were achieved in the in-line slalom (Figure 1b) as shown in Figure 5.

The German Slalom (or lateral jinking, Figure 1a) was performed at Crows Landing, CA, by NASA/Army and at the DFVLR. Lateral jinking is characterized by a sudden lateral translation to avoid an obstacle and a rapid return to a defined course (see Figure 1a). The NASA and DFVLR data for lateral jinking, given in Figure 6, support a 40 deg/sec roll rate requirement; the UH-1D (Figure 6c) achieved close to 50 deg/sec in

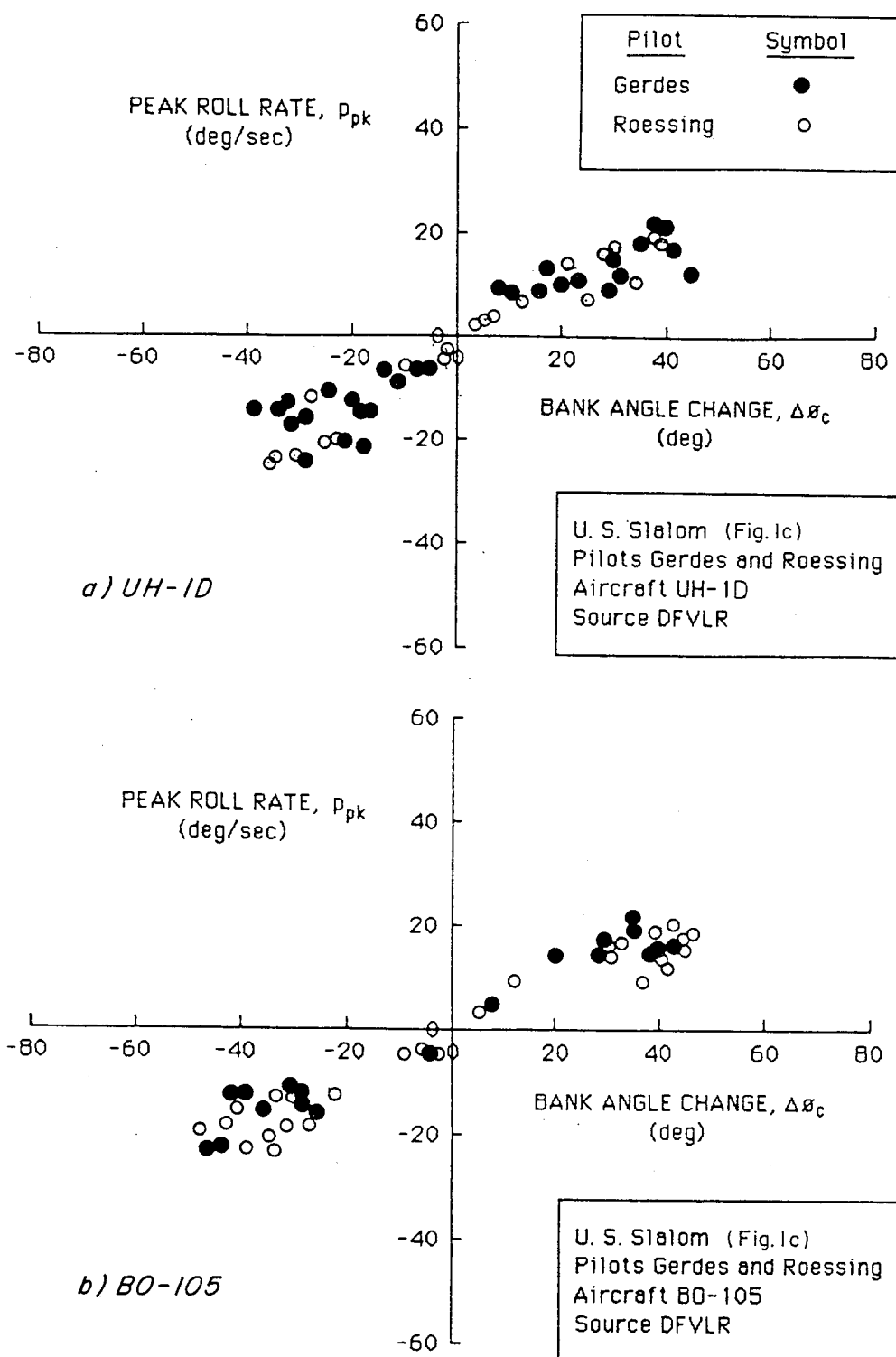


Figure 3(3.4.5.3). "U.S. Slalom" Maneuver Data  
(from Reference 93)

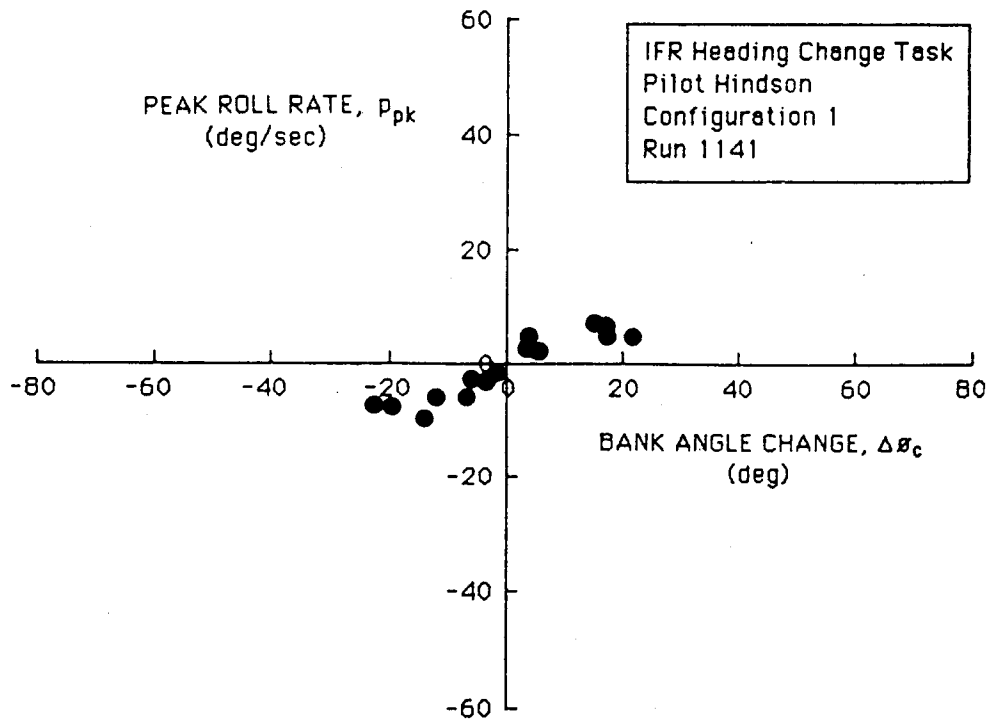


Figure 4(3.4.5.3). IMC Heading Change Performance Data for Basic Helicopter Type (from Reference 93)

two cases. The bank angle excursions for this MTE were less than those for the in-line slalom, with maximum values of slightly over 40 deg.

In an earlier test (1964), a UH-1B and an H-13 were flown through a series of aggressive NOE tasks (Reference 85). Reference 85 documents the peak rates and attitudes encountered in these tasks. These data showed a peak roll rate of 51 deg/sec for "evasive action" and 49 deg/sec in the "Fort Benning Confidence Course." The authors of Reference 85 recommend a roll rate capability of 70 deg/sec. Based on all of the available data, however, it would seem that 50 deg/sec is an adequate roll rate for NOE forward flight MTEs, and this is the value used in Table 3(3.4).

The DFVLR flew the "U.S. Slalom" course of Figure 1c at both 100 ft and 30 ft altitude. In the latter task, the pilots attempted to fly the shortest path between markers, rather than fly s-turns. The results of this test (Reference 88) were reviewed in the supporting data discussion for Paragraph 3.4.5.1. This task required higher peak roll rates to complete the turn. Pilot ratings are shown in Figure 7 on a plot of roll bandwidth (using derivatives given in References 88 and 28, and a time delay of 0.125 sec) versus control sensitivity. The lines of steady-state roll rate assume linear control response. These data show reasonable agreement with the 50 deg/sec roll rate requirement.



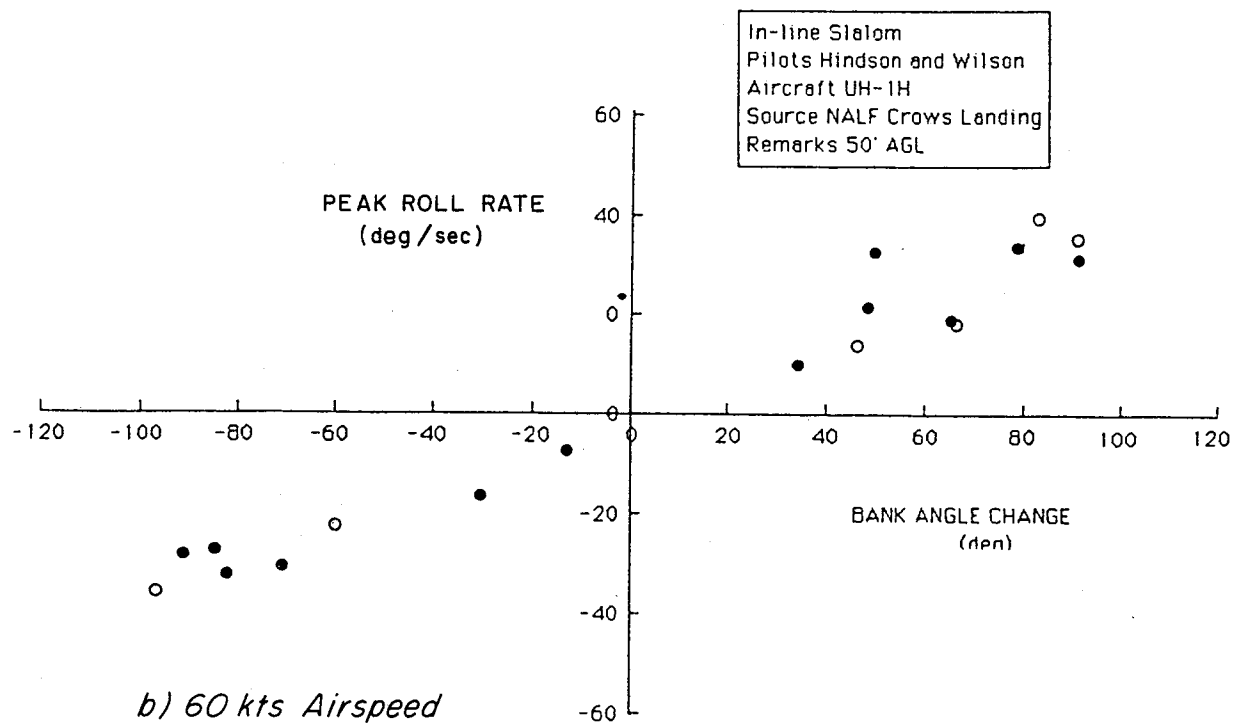
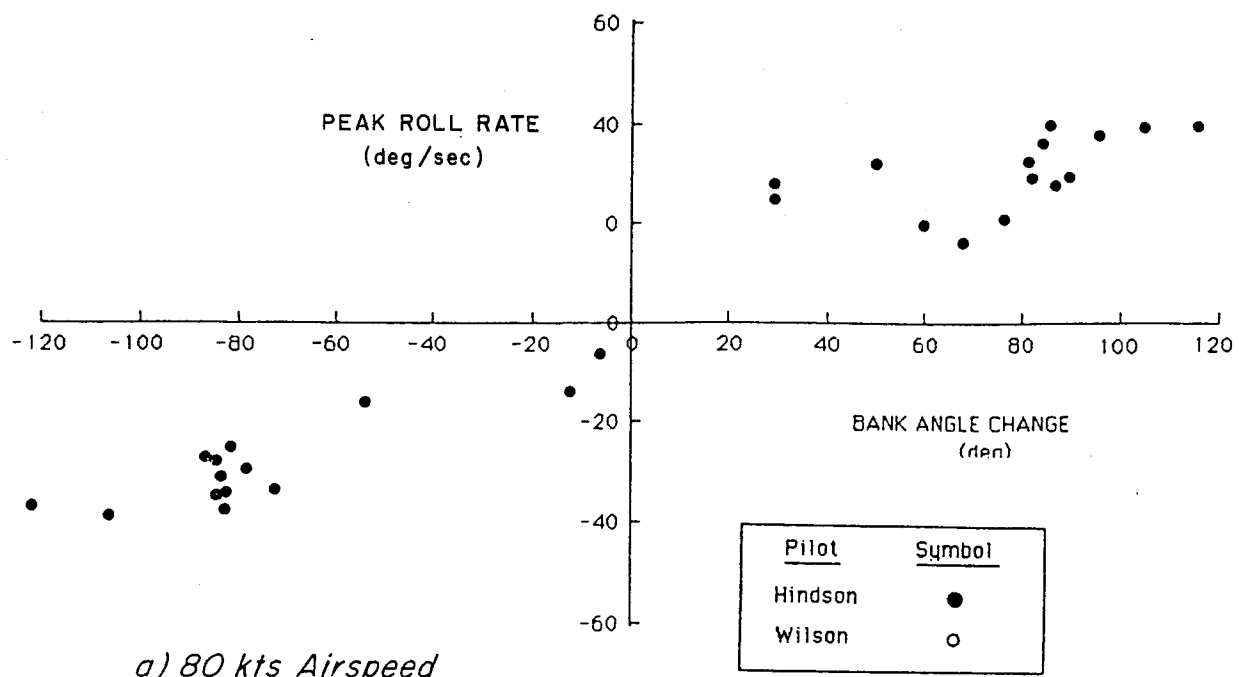


Figure 5(3.4.5.3). In-Line Slalom (Figure 1b) Maneuver Data  
(from Reference 93)

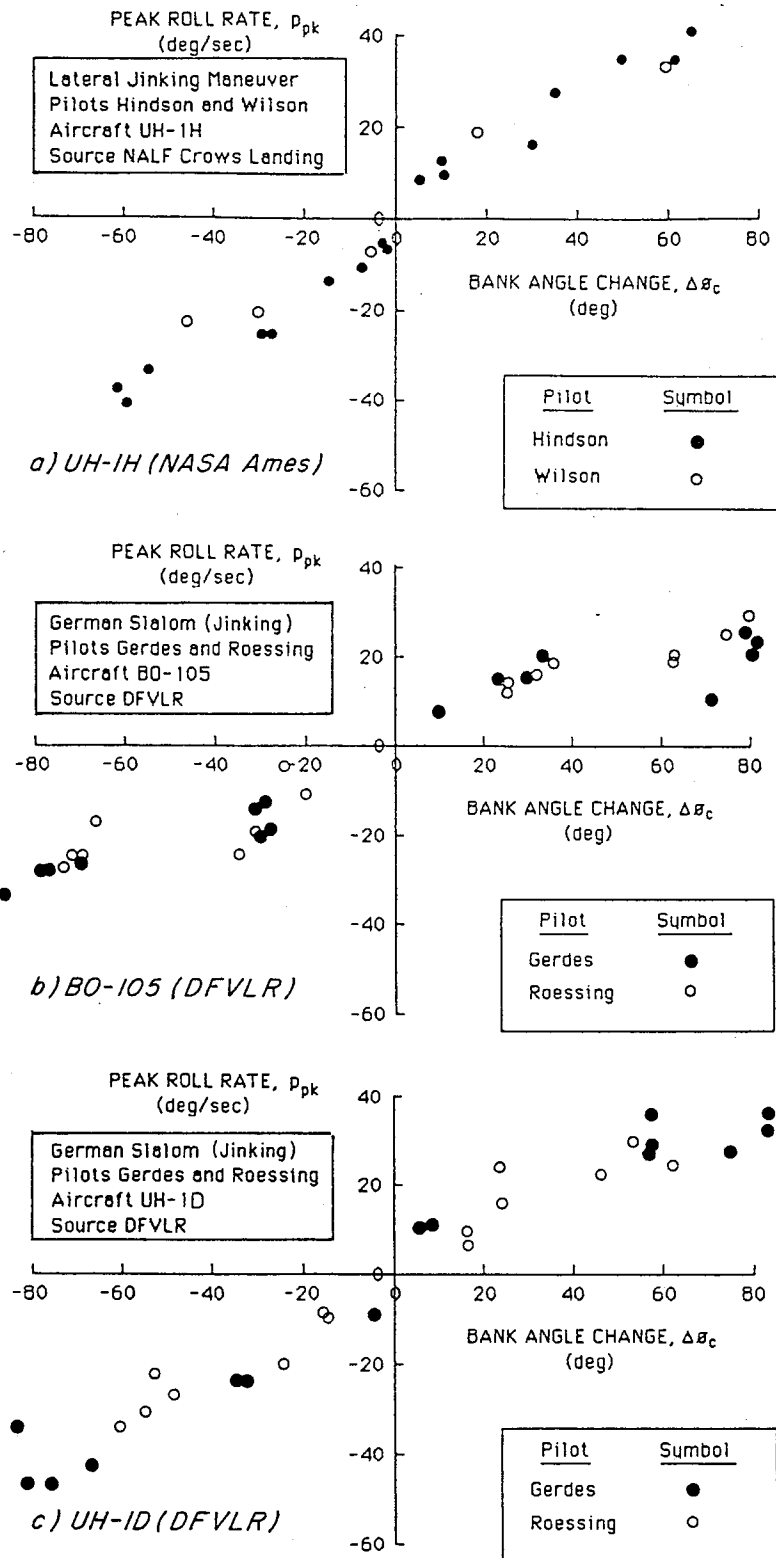


Figure 6(3.4.5.3). "German Slalom" (Jinking, Figure 1a)  
Maneuver Data (from Reference 93)

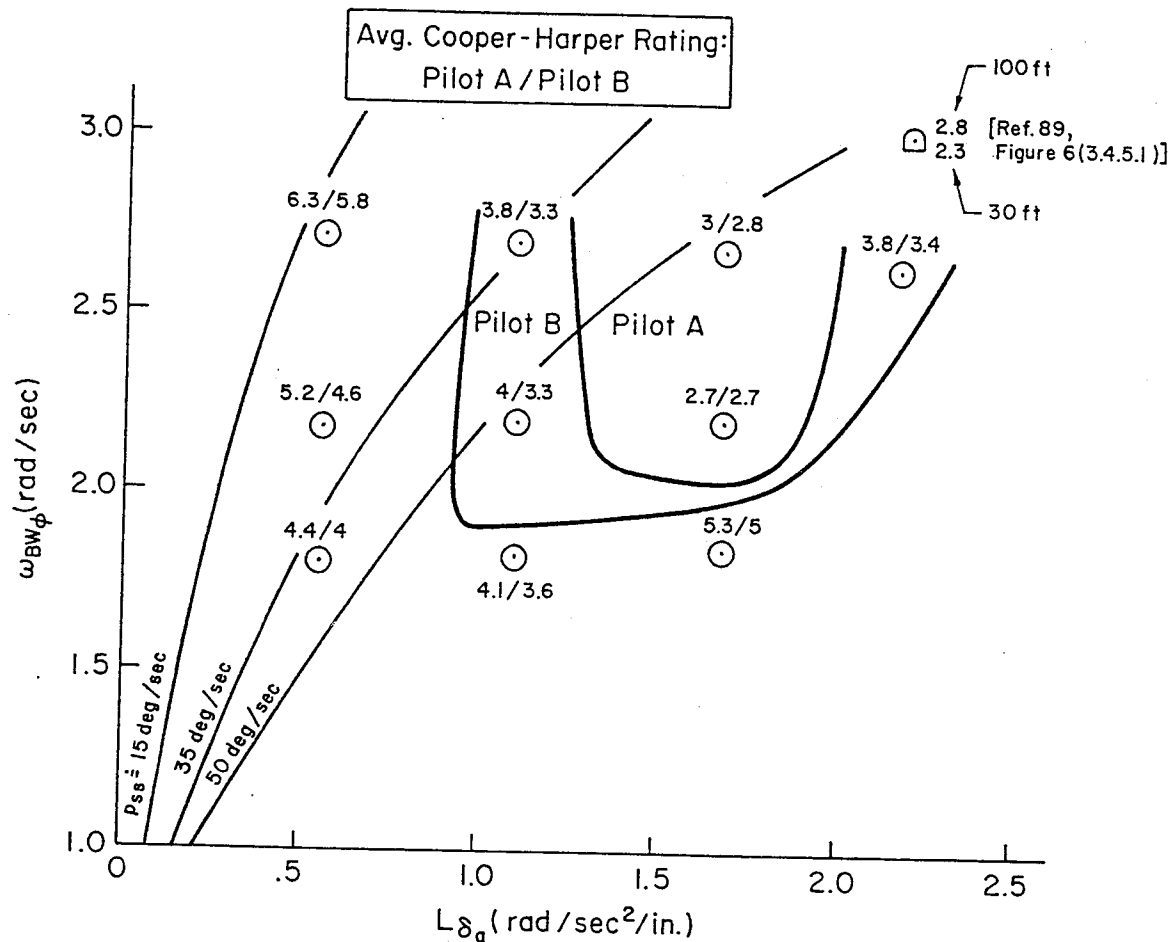


Figure 7(3.4.5.3). Steady-State Roll Rates Suggested by Results of Slalom Tests with B0-105 (Reference 88) (U.S. Slalom at 30 ft, Figure 1c)

A maximum bank angle capability of 90 deg, specified in Table 3(3.4), is somewhat larger than that supported by the Figure 5a slalom data. Several manufacturers and potential users of the specification have pointed out however, that a capability to achieve at least 90 deg of bank angle is desirable for an attack helicopter. Maneuvers such as "return to target," basically a wingover, utilize such large bank angles. The primary impact of the maximum bank angle requirement is to effectively eliminate Attitude Response-Types unless highly nonlinear lateral cyclic shaping is utilized. Another approach would be to have an Attitude Response-Type at low-to-moderate stick deflections and Rate Response-Type at larger deflections.

The Level 2 limit of 21 deg/sec roll rate is the same as the hover and low speed requirement at 45 kt. The pilot rating data in Figures 2

and 7 indicate that 15 deg/sec is adequate (e.g., HQR better than 6.5), and hence the use of the hover and low speed value is conservative. A Level 2 limit for the maximum bank angle is not apparent from the data. Thirty deg is specified in Table 3(3.4) on the basis that "adequate performance" [see Figure 1(2.8)] for these MTEs could not be achieved with lesser bank angles.

## AIR COMBAT

It was found in a U.S. Army helicopter air-to-air combat test (References 94 and 95) that roll agility was a key parameter in achieving the tactical advantage. For example, as noted in Reference 95, an OH-58 helicopter used in the tests was limited to 40 deg/sec roll rate, and was "at a significant disadvantage relative to the UH-60 and S-76 which recorded rates between 60 and 100 deg/sec during the encounters."

Experience with testing the AH-64 (Reference 98) has resulted in recommended maximum roll rates of at least 70 deg/sec at  $V_{mpr}$  (speed for minimum power in level flight) and 100 deg/sec at  $V_H$  (maximum level flight speed at intermediate rated power).

Maneuver flight data for the scissors maneuver for several helicopters is presented in Figure 8 (from Reference 93). These data show that a peak roll rate of 50 deg/sec would be adequate for all but one case (open square) where 70 deg/sec was achieved. These data were taken during the development of the test plan for air-to-air combat

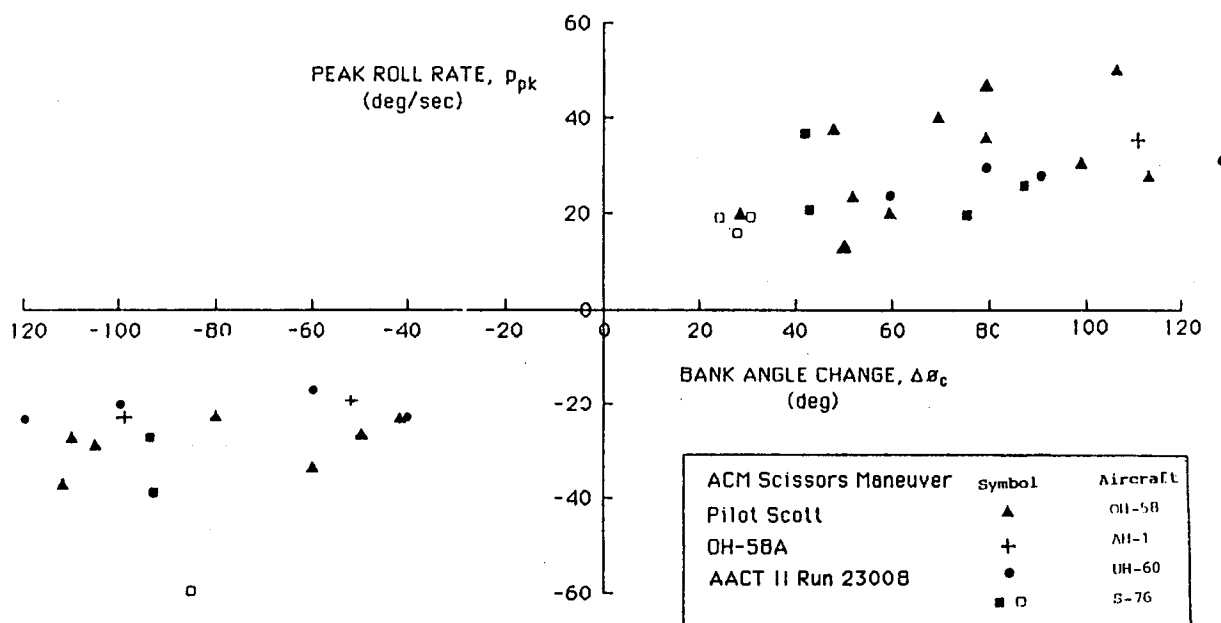


Figure 8(3.4.5.3). Scissors Maneuver Flight Data for a Variety of Helicopters (from Reference 93)

tests (Reference 96). It is interesting that somewhat higher roll rates were used during the actual tests (85 deg/sec was the maximum rate observed during the encounters in Reference 97).

A piloted simulation investigation of requirements for helicopter air combat (referred to as HAC II) was performed on the NASA Ames Vertical Motion Simulator (Reference 182). Variables included pitch and roll Response-Type, directional dynamics, and maneuver envelope limits. Roll-rate histograms from one-on-one air combat engagements for three pilots are shown in Figure 9. These plots show the percent of time the pilots exceeded a given roll rate on a given run (each line represents one run). These data show that the pilots rarely required more than 90 deg/sec to perform the task.

A control power of 90 deg/sec was specified on the basis that 1) 85 deg/sec was achieved with a modern-day helicopter in an actual air combat test; 2) the AH-64 is capable of 120 deg/sec; and 3) none of the manufacturers felt that 90 deg/sec was an excessive requirement. Since roll rate is not a strong function of airspeed, a relaxation to some lower roll rate at lower airspeeds is not warranted or needed. The most critical portion of the Operational Flight Envelopes for this requirement will be the upper limit on density altitude.

Aerobatic maneuvers in helicopters have been demonstrated on numerous occasions (for example, Reference 99 and film shown during presentations of Reference 98). It therefore seems reasonable to insist on the ability to perform 360 deg rolls for helicopters designed for air-to-air combat. This is reflected in Table 3(3.4) for Attitude-Response-Types.

There is no hard data upon which to base a Level 2 requirement for air-to-air combat. It seems reasonable, however, to insist on a roll rate of 50 deg/sec and a roll angle of 60 deg to stay reasonably effective following a control system failure or excursion into one of the Service Flight Envelopes.

d. Guidance for Application

None.

e. Related Previous Requirements

MIL-F-83300 sets limits on control response in terms of minimum attitude change in one second. The approach taken here is more directly related to the pilot's concerns in gross maneuvering. Attitude change per unit time is covered (indirectly) by the small and moderate-amplitude response requirements.

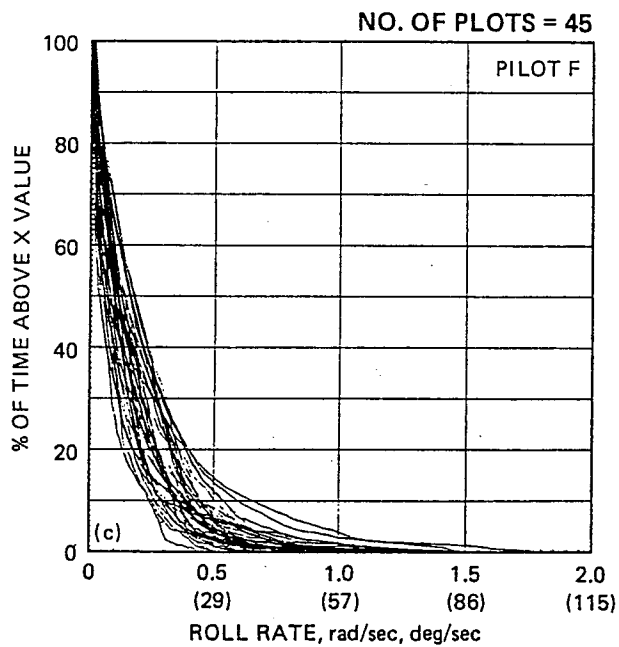
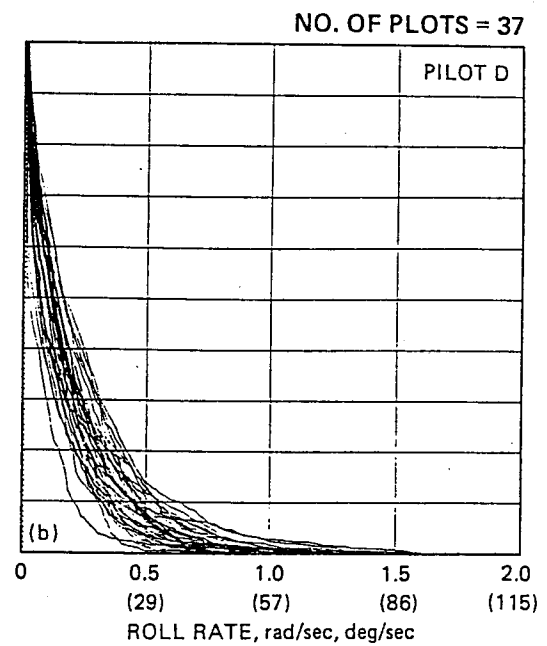
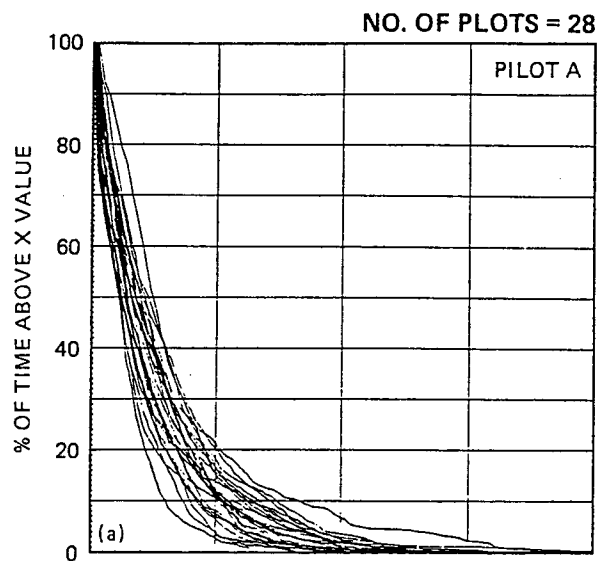


Figure 9(3.4.5.3). Time-Percentage Plots of Roll Rate Usage for Three Pilots from Helicopter Air Combat Simulation (from Reference 182)

a. Statement of Requirement

3.4.5.4 Linearity of Roll Response. There shall be no objectionable nonlinearities in the variation of rolling response with roll control deflection or force.

b. Rationale for Requirement

Ideally, some quantitative limit would be placed on the amount of allowable nonlinearity in roll response. There are no data for deriving such a limit, so this qualitative requirement from MIL-F-8785C is used instead.

c. Supporting Data

None.

d. Guidance for Application

None.

e. Related Previous Requirements

This statement comes from MIL-F-8785C (Reference 66, 3.3.4.4). A similar requirement is specified in MIL-F-83300 (Reference 29, 3.3.9.2).

a. Statement of Requirements

3.4.6 Roll-Sideslip Coupling. The requirements on roll-sideslip coupling apply for both right and left lateral control commands of all magnitudes up to the magnitude required to meet the roll performance requirements of 3.4.5.2. The cockpit yaw controller shall remain free. The parameters used are defined in Figure 4(3.4).

3.4.6.1 Bank Angle Oscillations. The value of the parameter  $\phi_{OSC}/\phi_{AV}$  following a pulse lateral control command for Rate Response-Types or step command for Attitude Response-Types shall be within the limits specified in Figure 5(3.4) for Levels 1 and 2. The input shall be as abrupt as practical. For Levels 1 and 2,  $\phi_{AV}$  shall always be in the direction of the lateral control command.

b. Rationale for Requirements

The roll-sideslip coupling requirements are based on the premise that helicopters are flown like fixed-wing airplanes in forward flight and that the need for steady roll response is just as great. The requirements have been taken from MIL-F-83300 (Reference 29).

Lateral oscillations generally result from a combination of two sources: low Dutch roll damping, and a large amount of yaw-due-to-lateral cyclic ( $N_{\delta_a}$ ) and yaw-due-to-roll ( $N_p$ ).

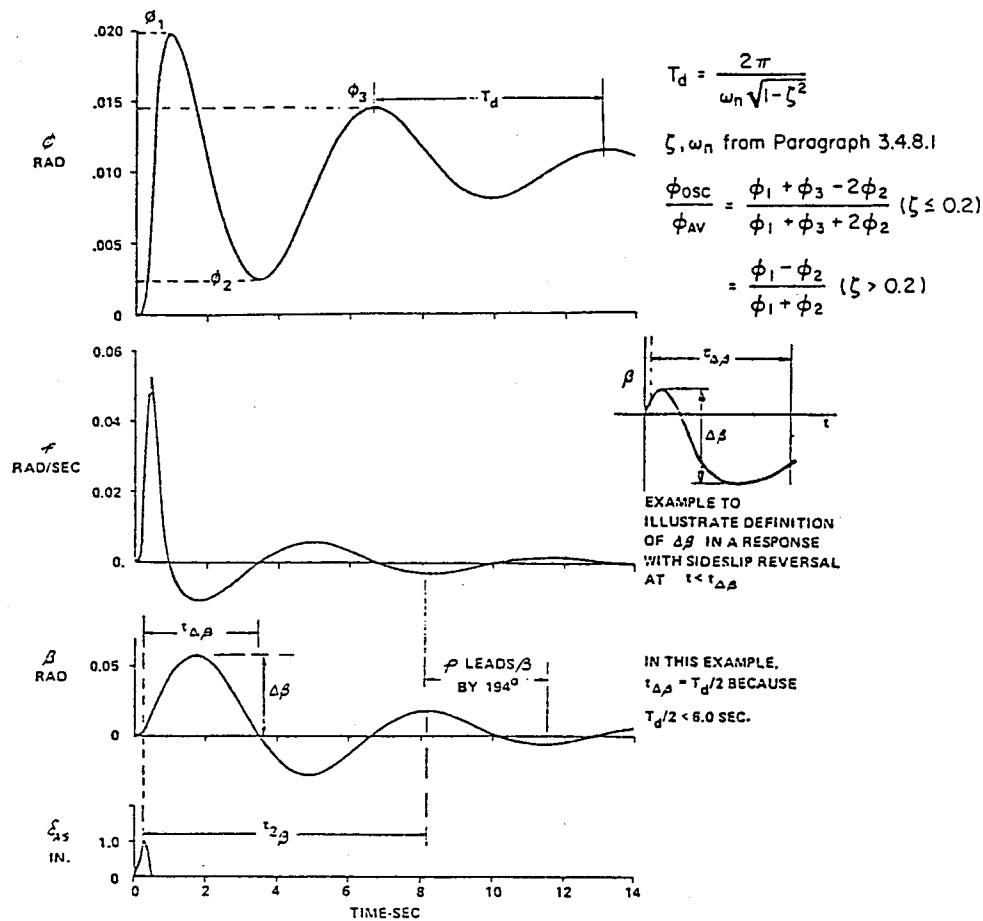
c. Supporting Data

The limits on  $\phi_{OSC}/\phi_{AV}$  vs.  $\psi_\beta$  in Figure 5(3.4) were developed for fixed-wing V/STOLs (Reference 30). There is only a limited amount of supporting data for rotorcraft, from Reference 115, discussed in Reference 30. This lack of data is not surprising, since it is a new requirement for rotorcraft. A review of flight test reports for operational helicopters (References 67-81) suggests that bank angle control problems are almost always related directly to inadequate Dutch roll damping.

In dynamic terms,  $\phi_{OSC}/\phi_{AV}$  results from the relationship between the roll numerator and Dutch roll denominator in the bank-angle-to-lateral-cyclic transfer function. For simple representation (neglecting higher-order dynamics, lags, etc.),

$$\frac{\phi}{F_{as}} = \frac{L_{F_{as}}[s^2 + 2\zeta_\phi\omega_\phi s + \omega_\phi^2]}{(s + 1/T_s)(s + 1/T_R)[s^2 + 2\zeta_d\omega_d s + \omega_d^2]}$$





$\phi, \beta, \delta_{AS}$  - change in roll attitude, sideslip, and lateral control position from trim

$\Delta\beta$  - the maximum change in sideslip following an abrupt roll control pulse command within time  $t_{\Delta\beta}$  where  $t_{\Delta\beta}$  is the lesser of 6 seconds or  $T_d/2$ .

$t_{n\beta}$  - time for the lateral-directional oscillation in the sideslip response to reach the  $n^{\text{th}}$  local maximum for a right command.

$\psi_\beta$  - phase angle expressed as a lag for a cosine representation of the lateral-directional oscillation in sideslip, where

$$\psi_\beta = \frac{-360}{T_d} t_{n\beta} + (n-1)360 \text{ (degrees)}$$

with  $n$  as in  $t_{n\beta}$  above.

$|\phi/\beta|_d$  - at any instant, the ratio of amplitudes of the bank angle and sideslip angle envelopes in the lateral-directional oscillatory mode.

Figure 4(3.4). Roll-Sideslip Coupling Parameters

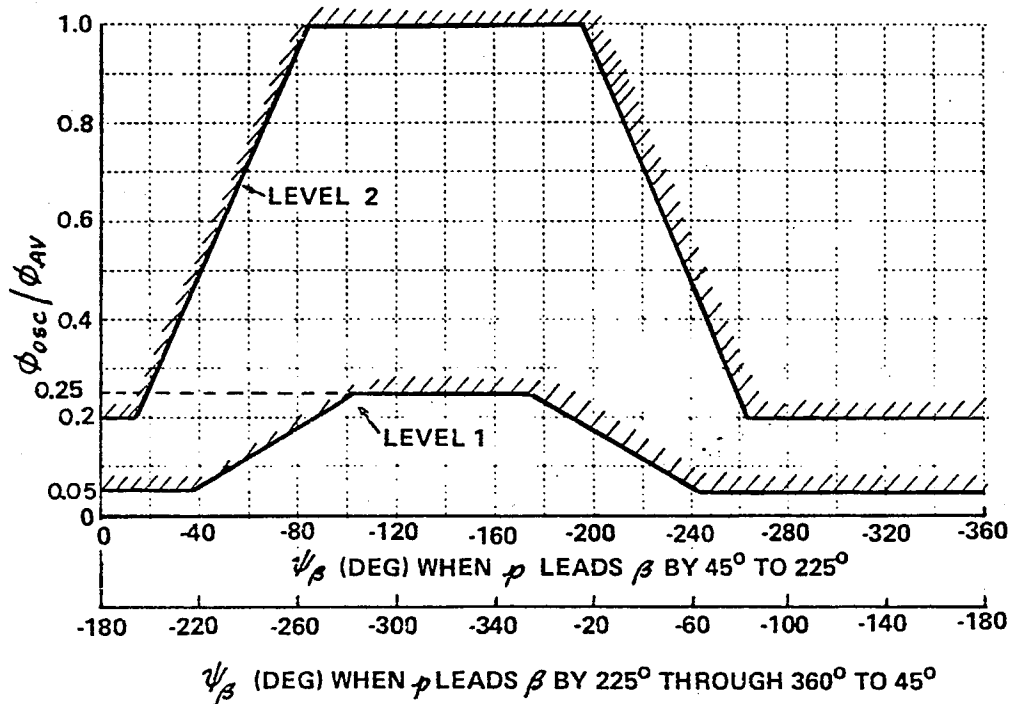


Figure 5(3.4). Bank Angle Oscillation Limitations

For ideal roll/sideslip decoupling,  $\zeta_\phi \doteq \zeta_d$ ,  $\omega_\phi \doteq \omega_d$ . Then the transfer function becomes simply

$$\frac{\phi}{F_{as}} = \frac{L_{Fas}}{(s + 1/T_s)(s + 1/T_R)}$$

The less perfect this cancellation, the greater the bank angle oscillation. The amount of oscillation, however, is also directly related to the Dutch roll frequency and damping ratio.

Computed  $\phi_{OSC}/\phi_{AV}$  for several helicopters, using data from Reference 28, were evaluated against the Figure 5(3.4) requirements. The UH-1H (teetering rotor), BO-105C (hingeless), and OH-6A (articulated) were chosen. A speed range from 60 to 130 kt (145 kt for the BO-105C) was used to assess the influence of airspeed on  $\phi_{OSC}/\phi_{AV}$ . The results are plotted in Figure 1. The UH-1H data are for the stabilizer bar off;  $\phi_{OSC}/\phi_{AV}$  is essentially zero for this helicopter with the bar off or on.

All of the helicopters easily meet the Level 1 limits. This is not too surprising when one looks at the Dutch roll and roll numerator characteristics, shown in Figure 2. The Dutch roll mode is relatively well-damped in all cases (only the BO-105C fails to meet the Level 1 lateral-directional oscillation limit). In addition, the roll zero is near the Dutch roll pole, indicating that yaw-due-to-lateral-cyclic and yaw-due-to-roll ( $N_{\delta_a}$  and  $N_p$ , respectively) are not excessively large.

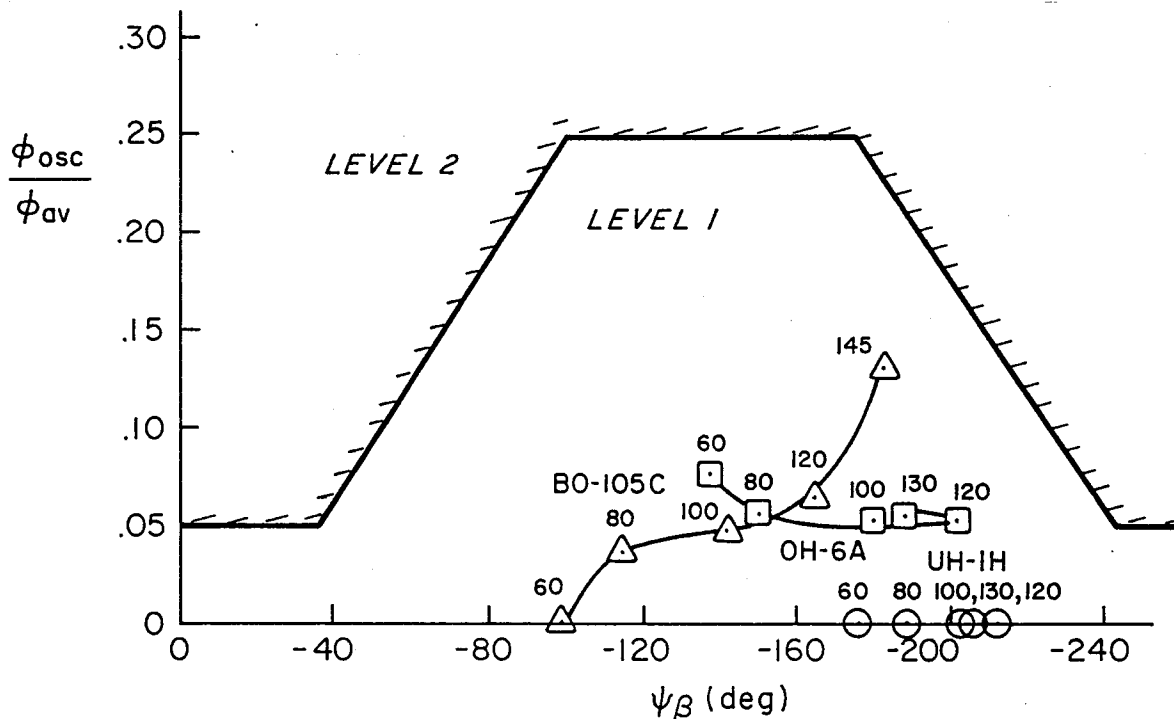


Figure 1(3.4.6.1). Bank Angle Oscillations for Helicopter Models from Reference 28

Thus, it is likely that actual rotorcraft will have little difficulty meeting this requirement, especially with some artificial stability augmentation.

#### d. Guidance for Application

Determination of  $\phi_{osc}/\phi_{av}$  and  $\psi_\beta$  requires measurement of several parameters from the bank angle, roll rate, and sideslip responses to a pulse lateral control input. The procedure is illustrated in Figure 4(3.4).

#### e. Related Previous Requirements

These requirements are based on 3.3.8 and 3.3.8.1 of MIL-F-83300 (Reference 29). A similar requirement, with identical Level 1 limits, is used in MIL-F-8785C (Reference 66).

There is no requirement in MIL-H-8501A corresponding to these requirements. Paragraph 3.6.1.2 of that specification limits lateral-directional oscillations in instrument flight, but this is more related to the denominator (i.e., Dutch roll) characteristics. Several paragraphs in MIL-H-8501A regulate against adverse yaw characteristics,

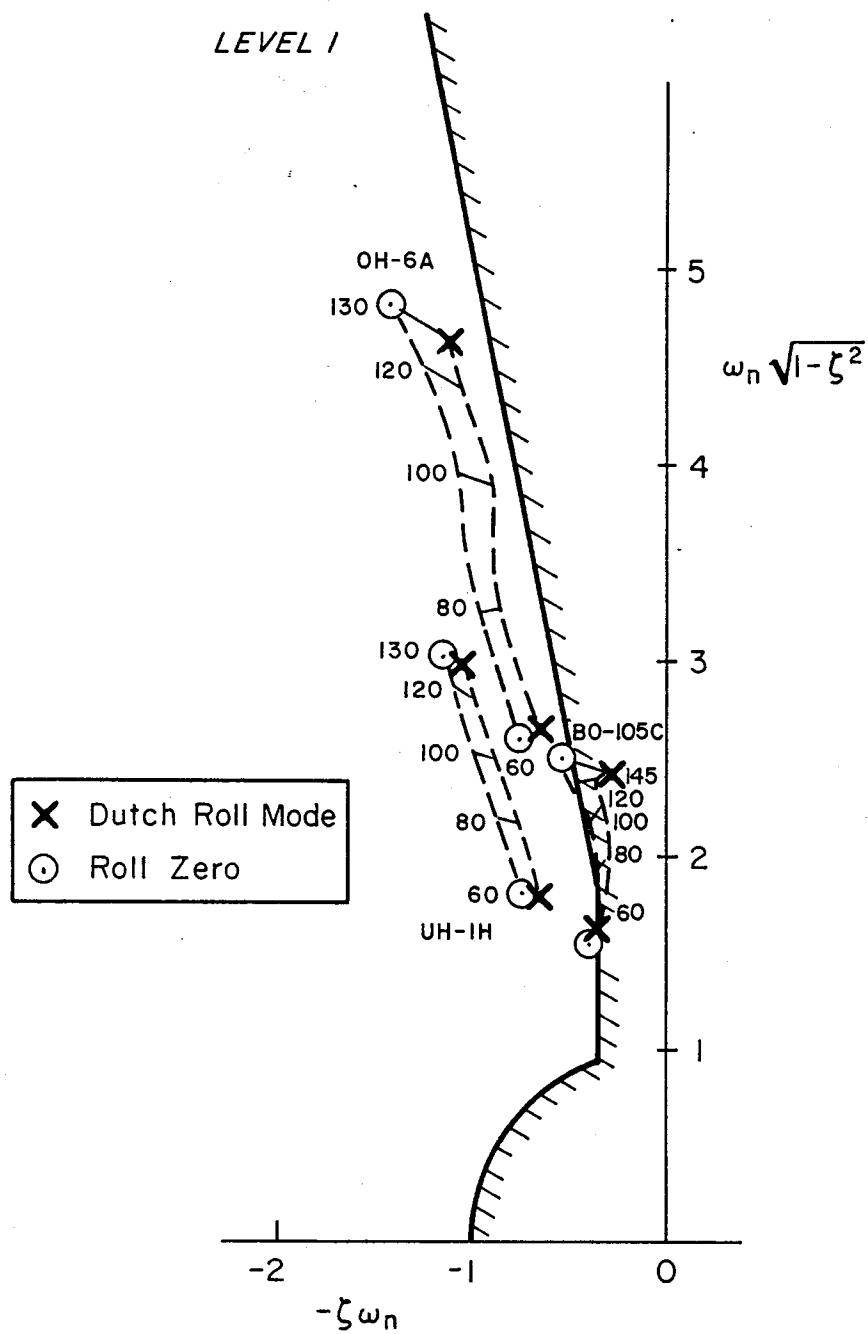


Figure 2(3.4.6.1). Roll Numerator and Dutch Roll Mode Characteristics for Helicopter Models of Figure 1

and hence indirectly relate to these requirements. For example, 3.3.9.1 requires that "no reversal in rolling velocity (i.e., return through zero) shall occur after a small lateral step displacement of the control stick is made with pedals fixed." This is akin to requiring  $\phi_2 > 0$  (Figure 4(3.4)), which is much more lenient than the current requirement -- e.g., suppose  $\phi_2 = 0$ ; then  $\phi_{OSC}/\phi_{AV} = 1.0$ , which is Level 2 (or Level 3) by Figure 5(3.4).

a. Statement of Requirement

3.4.6.2 Turn Coordination. The amount of sideslip resulting from abrupt lateral control commands shall not be excessive or require complicated or objectionable directional control coordination. The ratio of the maximum change in sideslip angle to the initial peak magnitude in roll response,  $|\Delta\beta/\phi_1|$ , for an abrupt lateral control pulse command for Rate Response-Types or step command for Attitude Response-Types, shall not exceed the limit specified on Figure 6(3.4). In addition,  $|\Delta\beta/\phi_1| \times |\phi/\beta|_d$  shall not exceed the limit specified on Figure 7(3.4).

b. Rationale for Requirement

This requirement is directed at precision of control, and in particular at limiting large sideslips during turns. For precise coordinated turns, it is desirable to have minimal sideslip and, for the sideslip that does occur, that it be correctable without undue pilot effort. This requirement, is taken from the V/STOL specification (Reference 29), and has been selected as a method which can be easily flight tested. The criterion suggested in Paragraph 3.2.10.1 of this BIUG should also be checked if turn coordination is marginal.

c. Supporting Data

The necessity to limit sideslip response to bank angle commands is long standing and numerous requirements have been developed for specifying reasonable limits. Both the fixed-wing V/STOL (Reference 29) and conventional airplane (Reference 66) specifications place limits on sideslip in turns. This is, however, a new requirement for rotorcraft, so there is little information available for rotary-wing aircraft. The only relevant flight data (Reference 115), shown in Figure 1, show reasonable agreement with the limits of Figures 6(3.4) and 7(3.4). As with Paragraph 3.4.6.1, we can compare analytical models of helicopters, using data from Reference 28, with the requirements of Figures 6(3.4) and 7(3.4). This is done for the UH-1H (stabilizer bar off), OH-6A, and BO-105C, for a range of airspeeds, in Figure 2. All aircraft meet the Level 1 requirement. The UH-1H with stabilizer bar on has even less sideslip, so it is not shown.

d. Guidance for Application

Figure 4(3.4) defines the parameters needed for applying Figures 6(3.4) and 7(3.4).

e. Related Previous Requirements

This requirement comes from 3.3.8.2 of MIL-F-83300 (Reference 29).

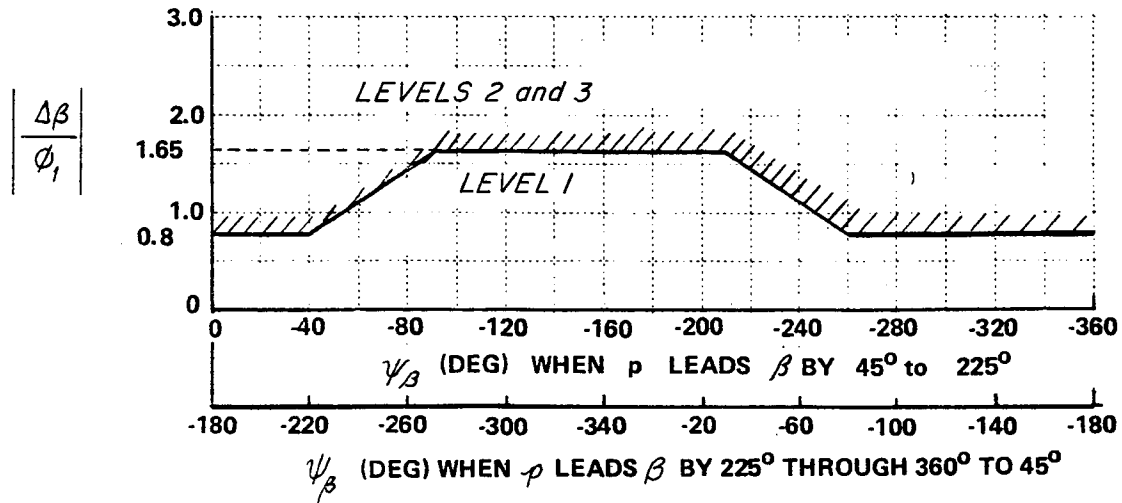


Figure 6(3.4). Sideslip Excursion Limitations  
(Boundary for  $\left| \frac{\Delta\beta}{\phi_1} \right|$ )

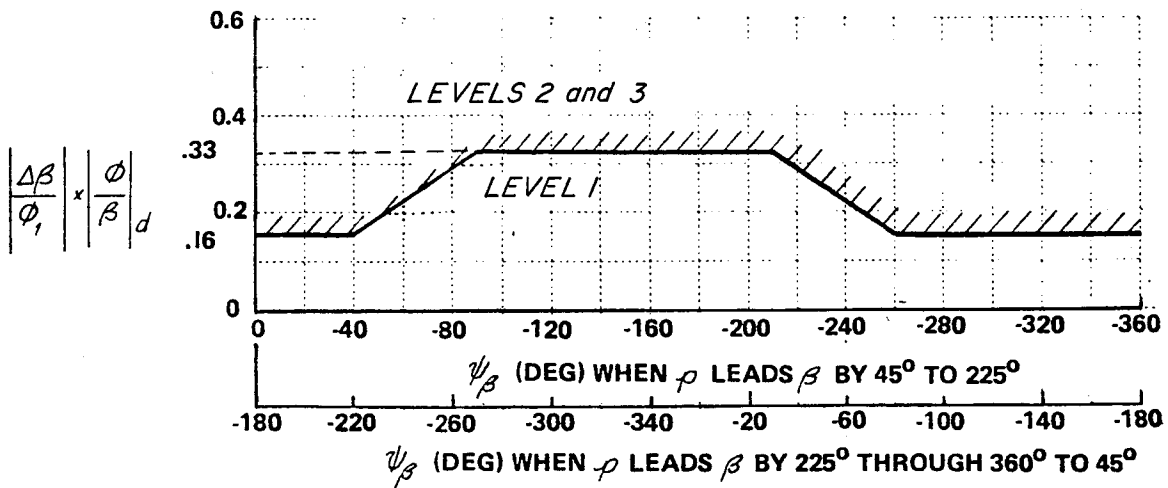


Figure 7(3.4). Sideslip Excursion Limitations  
(Boundary for  $\left| \frac{\Delta\beta}{\phi_1} \right| \times \left| \frac{\phi}{\beta} \right|_d$ )

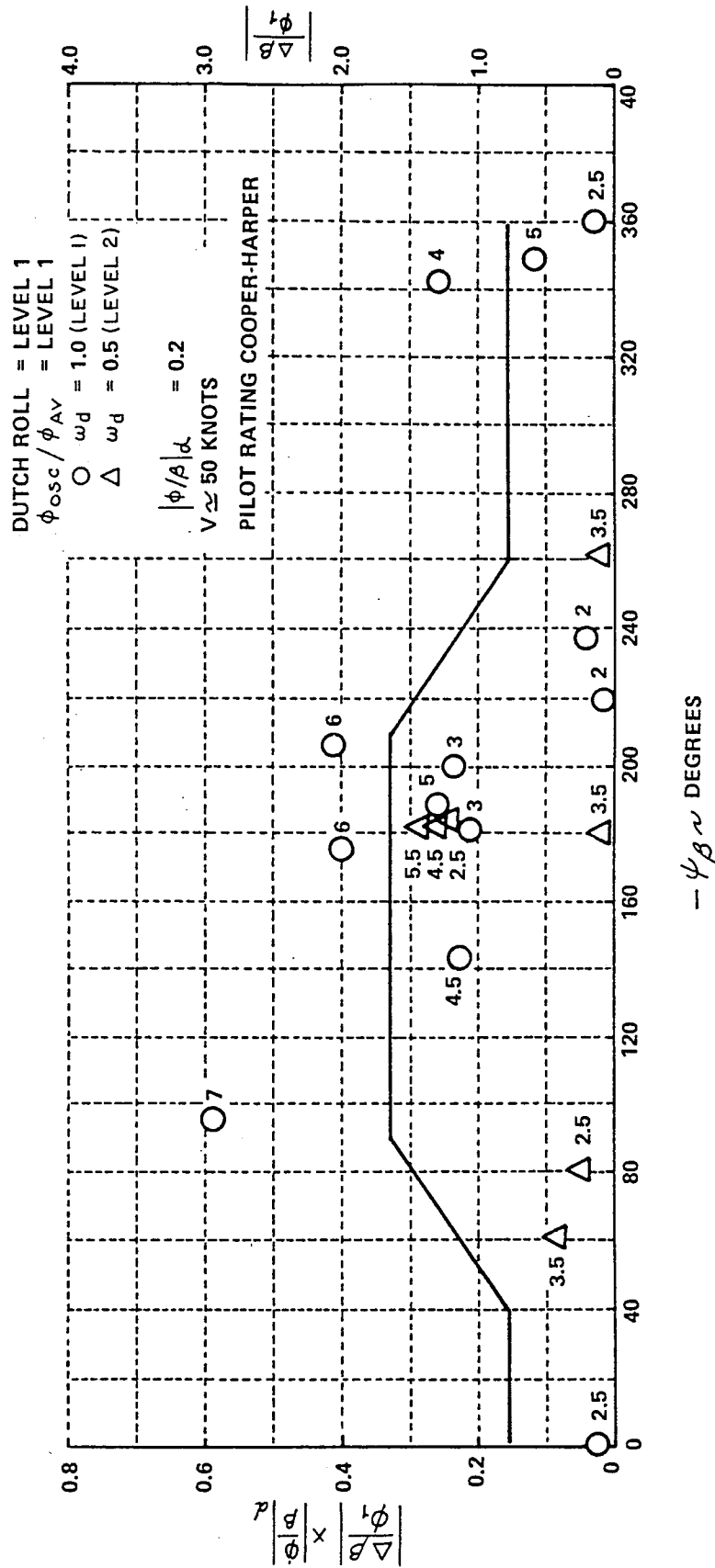
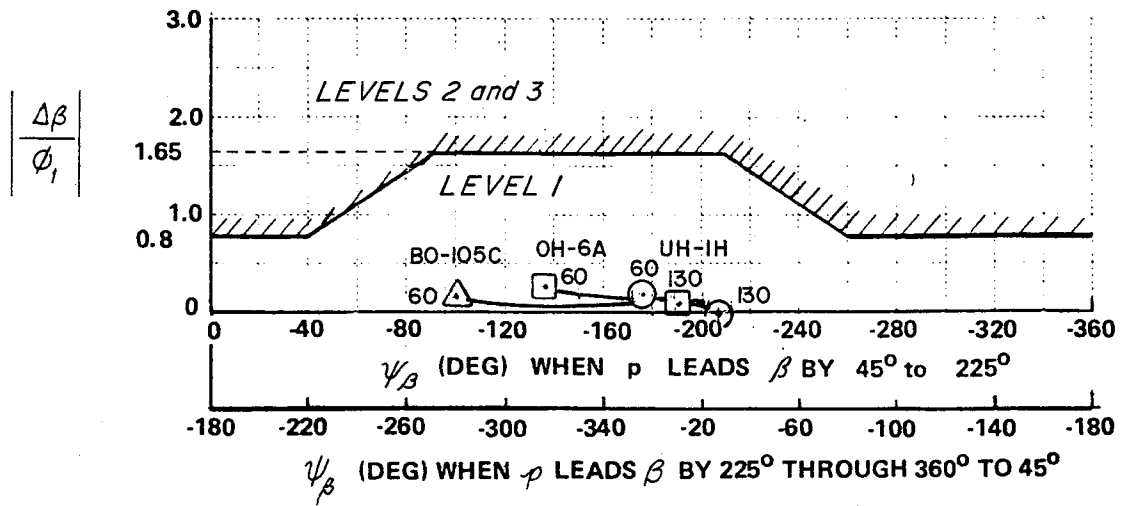
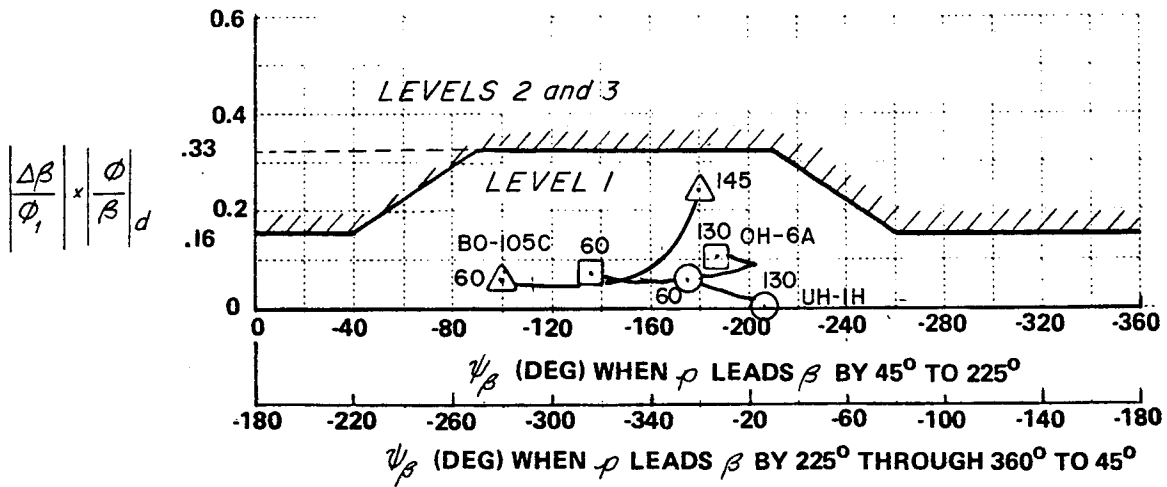


Figure 1(3.4.6.2). Comparison of Sideslip Excursion Requirement and Pilot Rating from NRC of Canada Flight Experiment (Reference 115; Figure Reproduced from Reference 30)





a) Boundary for  $|\Delta\beta/\phi_1|$



b) Boundary for  $|\Delta\beta/\phi_1| |\phi/\beta|_d$

Figure 2(3.4.6.2). Comparison of Analytical Models of Several Helicopters with Sideslip Excursion Requirements

a. Statement of Requirement

3.4.7 Yaw Response to Yaw Controller

3.4.7.1 Small-Amplitude Yaw Response for Air Combat (Bandwidth). The heading response to cockpit yaw control force or position inputs shall meet the limits specified in Figure 8(3.4). The bandwidth ( $\omega_{BW_\psi}$ ) and phase delay ( $\tau_{p_\psi}$ ) parameters are obtained from frequency responses as defined in Figure 2(3.3).

It is desirable to meet this criterion for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish.

b. Rationale for Requirement

The directional-axis requirements for forward flight have been developed by making two fundamental assumptions: 1) that most rotorcraft will be flown in coordinated flight for most of the required MTEs; and 2) that meeting the heading requirements in the Hover and Low Speed section will generally guarantee satisfactory yaw response at higher speeds. As a result, many of the requirements for forward-flight operations are qualitative in nature, and, for example, there is no heading bandwidth requirement for most MTEs.

Several simulations of helicopter air combat have demonstrated the strong need to have a high bandwidth in heading. It is on the basis of these simulations that this paragraph is included in the specification, and that it applies explicitly to the Air Combat MTE. There are no indications that the other forward-flight MTEs are not adequately addressed by the other requirements of the specification.

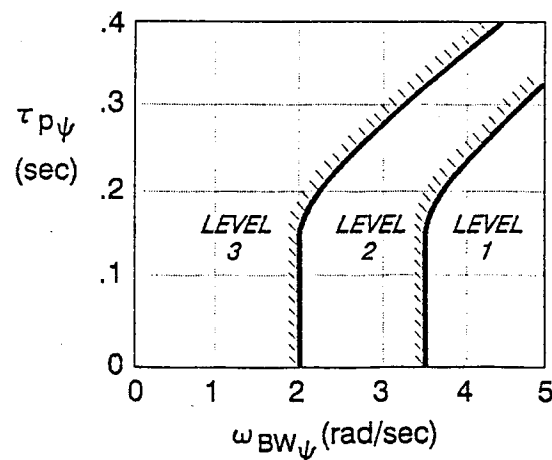


Figure 8(3.4). Requirement for Small-Amplitude Yaw Response for Air Combat -- Forward Flight

### c. Supporting Data

Three piloted simulations have been conducted by the U.S. Army to investigate the flying qualities requirements for one-on-one air combat. The simulations, dubbed HAC (for Helicopter Air Combat) I, II, and III, are documented in References 181, 182, and 183, respectively. The first two simulations were of an exploratory nature, leading to the third phase, from which the supporting data for this requirement are taken.

HAC I and II were performed on the NASA Ames Vertical Motion Simulator (VMS) using the full motion-base capability. For HAC III motion was not available, so the VMS was operated in a fixed-base mode. A Honeywell Integrated Helmet and Display Sight System (IHADSS) was employed for helmet sighting and to provide flight symbology. The helicopter model was representative of a modern attack aircraft with operational limits similar to those of the AH-64 Apache. The target aircraft was basically the same model, flown by a second pilot. Primary response variables were damping ratio and natural frequency of the heading response, fixed-forward vs. turreted guns, and size of the turret maneuver envelope. NOE operations were stressed and reinforced by employing a ground-to-air threat warning in the cockpit whenever altitude exceeded 150 ft above the ground. The pitch, roll and heave responses were all Level 1.

Six pilots participated in the study, although the limited-turret configurations were flown by only three of the pilots, and the fixed-gun configurations by two. Handling qualities ratings were given for a gross maneuvering subtask (covering the initial maneuver to gain a firing position), and a fine tracking subtask (weapon utilization). The ratings in Reference 183 indicate that the fine tracking subtask was generally the more demanding (HQRs were generally worse), so these data are used here.

Figure 1 summarizes the average HQRs and spreads in HQR from the HAC III simulation. For reference, the number of ratings for each data point is noted in parentheses next to the symbol for that point. Heading bandwidths were computed using derivatives and computational delays reported in Reference 183. The derivatives were devised so that the response dynamics were constant for all airspeeds above 50 kts.

With a fixed-forward gun (Figure 1a) precise attitude pointing was required to perform the task. The pilot ratings of Figure 1a suggest that a heading bandwidth of at least 3.5 rad/sec is required for Level 1 handling qualities, and a bandwidth of about 2.5 rad/sec for Level 2. The limited-motion turret was driven by the IHADSS sighting system with limits of  $\pm 40$  deg in azimuth, and 10 deg up and 60 deg down in elevation. The ratings of Figure 1b show some improvement with this turret, though the data trends are similar to the fixed-gun data of Figure 1a. Turret usage analysis in Reference 183 indicates that the up-elevation limit of 10 deg was the most constraining to the pilots and caused the most adverse comments. The full-traverse turret had an increased up-elevation range of 20 deg, and  $\pm 110$  deg in azimuth. The HQRs in Figure 1c show considerable improvement over the limited-motion data, requiring about 3 rad/sec for Level 1, and a Level 2 limit was not

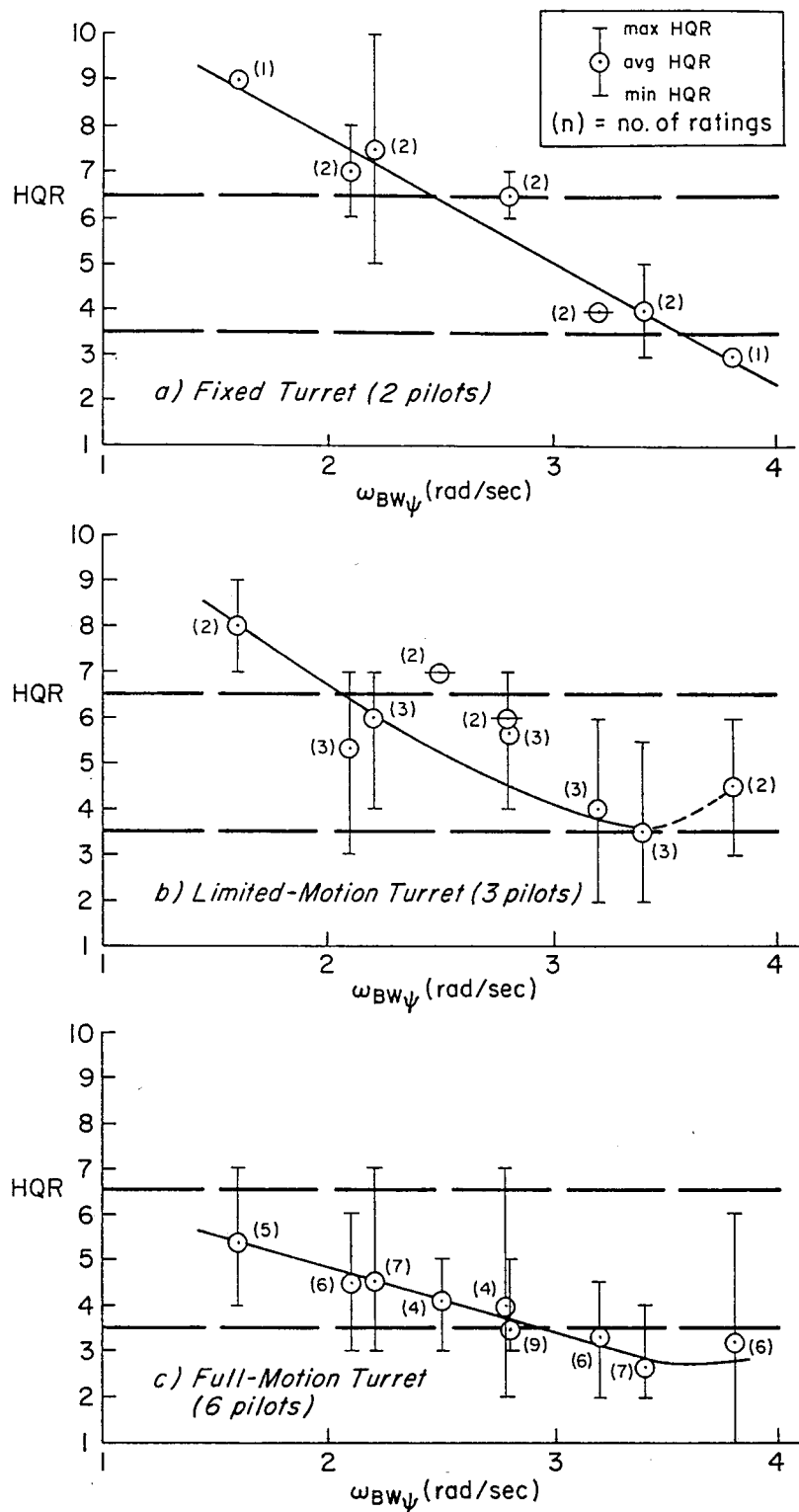


Figure 1(3.4.7.1). Summary of Handling Qualities Ratings from HAC III Simulation (Reference 183)

reached. The turret-usage analysis in Reference 183 shows that while the pilots did not require the additional range in azimuth, the increased up-elevation limit provided a significant improvement in tracking capability.

The Level 1 and 2 limits on heading bandwidth (3.5 and 2 rad/sec, respectively) in Figure 8(3.4) have been specified on the basis of the fixed and limited-motion turret results of Figures 1a and 1b for several reasons. Firstly, the turret model used in the simulation had an extremely high response bandwidth and represented an idealized gun system. Secondly, since most gun turrets are, and will continue to be, mounted on the bottom of the fuselage, the up-elevation capability will be restricted. And thirdly, while the improvement in handling qualities (especially for Level 2 operations) is quite strong when a wide-motion turret is used, incorporating this into the specification would demand that additional requirements be placed on the response of the gun system itself, and this is neither appropriate nor feasible for a handling qualities specification.

d. Guidance for Application

Guidance for obtaining the required frequency response (Figure 2(3.3)) is given in Appendix A. Since application of Figure 8(3.4) requires frequency responses of heading ( $\psi$ ) to yaw control inputs, the most straightforward method to obtain the heading response will be to record the yaw rate signal since, for small pitch and roll attitudes,  $\dot{\psi} \approx r/s$ .

e. Related Previous Requirements

There are no requirements in other specifications relating specifically to the short-term bandwidth.

a. Statement of Requirement

3.4.7.2 Large-Amplitude Heading Changes. The heading change following an abrupt step displacement of the yaw control, with all other cockpit controls fixed, shall not be less than:

- Level 1: the lesser of 16 degrees or  $\beta_L$
- Level 2: the lesser of 8 degrees or  $1/2 \beta_L$
- Level 3: the lesser of 4 degrees or  $1/4 \beta_L$

where  $\beta_L$  is the sideslip limit of the Operational Flight Envelope in degrees.

b. Rationale for Requirement

For most Mission-Task-Elements, yaw control power at low forward flight speeds will most likely be set by crosswind performance and heading control during rapid decelerations and at high speeds by torque limits. (Important exceptions are air-to-air combat and ground attack.) This paragraph places a limit on directional control response as well as control power for maneuvering flight.

The format of this requirement is based on MIL-F-83300 (Reference 29, 3.3.10.1). Other forms for specifying control power, such as maximum yaw rate per unit time, were investigated briefly. There is little data for developing any new requirement, however, and it appears that specifying yaw attitude change in one second is sufficient at this time. The actual attitude change required is not well-supported by data, but, as the Supporting Data discussion shows, it is adequate.

It is expected that, for most helicopters, meeting the low-speed heading requirements of 3.3.5 will assure sufficient response to meet this requirement as well.

c. Supporting Data

With the exception of the Helicopter Air Combat (HAC) simulations of References 181, 182, and 183, there have been no systematic investigations of yaw response required for helicopters in forward flight. The attitude requirements specified here are based on a review of the yaw characteristics of current helicopters from Reference 28. The values for attitude change in one second are well above those specified for V/STOLs in MIL-F-83300, but appear to be attainable by most helicopters. As an example, Figure 1 shows the range of yaw attitude changes measured in flight for a one-inch rudder pedal displacement on an AH-1G Huey Cobra, SCAS on (Reference 68). The requirements of this paragraph are easily met, especially at high speeds where the sideslip limit decreases. The Huey Cobra was considered to have good directional controllability characteristics in forward flight.

The Helicopter Air Combat (HAC II) simulation of Reference 182 investigated the effects of the variations in sideslip maneuvering

- NOTES: 1. SCAS ON  
2. SHADED ENVELOPE CONTAINS ALL SCAS ON FLIGHT TEST DATA

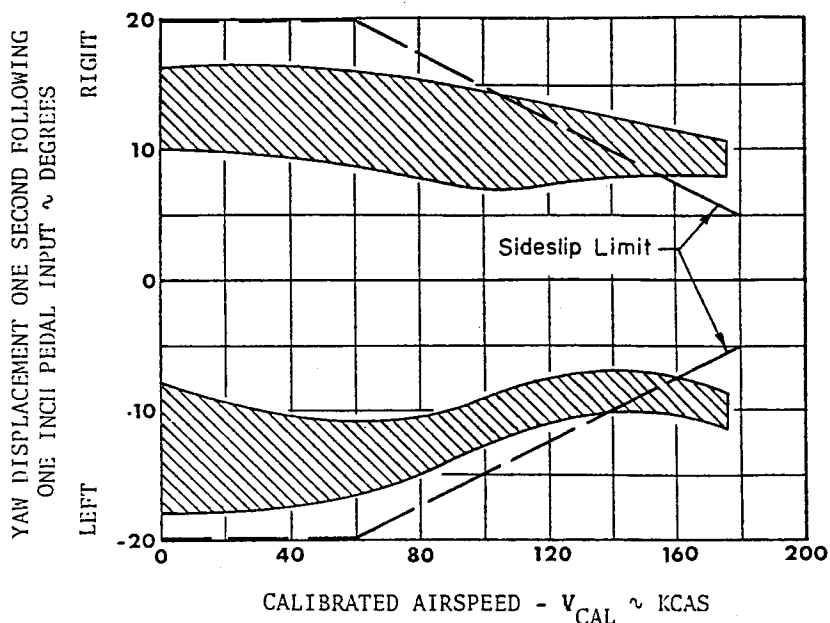


Figure 1(3.4.7.2). Yaw Attitude Response at 1 sec for a 1-inch Pedal Input. AH-1G Huey Cobra (from Reference 68)

envelope for performing one-on-one air combat engagements. Three maneuvering envelopes (covering steady and transient load factors, rate of climb, and sideslip) were evaluated: a low baseline, corresponding to an early model attack aircraft; a high baseline, typical of a modern utility aircraft; and an advanced model representative of future scout/attack aircraft. The sideslip envelopes for these models are shown in Figure 2. Since these envelopes were set by aircraft limits, rather than mission requirements, they can be interpreted as the limits of the Service Flight Envelope (SFE) rather than the Operational Flight Envelope (OFE) (Paragraphs 2.9.1 and 2.9.2).

Figure 3 shows pilot usage data in the form of crossplots of sideslip vs. airspeed for the three envelopes. Each trace in these plots represents one run. The traces for the low-baseline envelope are included on the other plots, and the high-baseline traces are included on the advanced-aircraft plot. Excursions outside the limits of the envelopes resulted from a combination of large yaw and roll inputs. These plots show that the pilots take advantage of the expanded operating envelope provided by the high baseline, but seldom required the full range of sideslips provided for the advanced aircraft. The authors of Reference 182 conclude that,

Though the degree of sideslip used by individual pilots varied, the most successful pilots used aircraft sideslip performance to significant advantage. For these pilots, the sideslip envelope

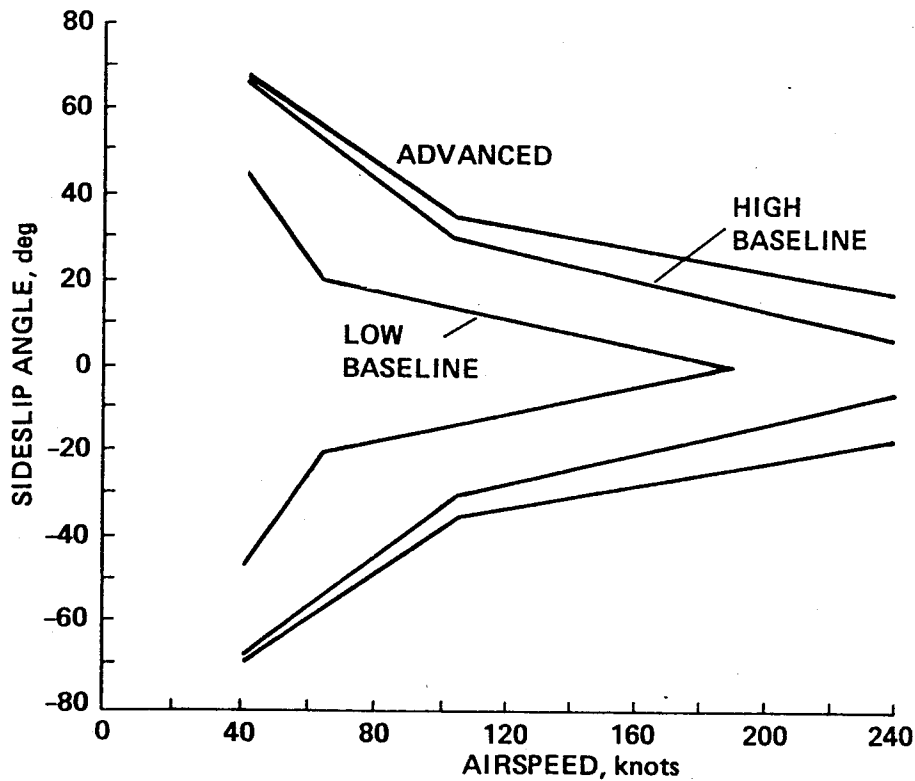


Figure 2(3.4.7.2). Sideslip Envelopes from HAC II Simulation  
(from Reference 182)

typical of early attack helicopters is clearly not sufficiently large. The envelope afforded by modern utility aircraft is close to adequate if the entire envelope can be exploited without consequence.

For the specification, this has been interpreted as requiring that it must be "possible to develop steady sideslip angles throughout the Operational Flight Envelope," since the OFE represents the envelope specified by mission requirements.

#### d. Guidance for Application

While the intent of this paragraph is to assure adequate control power, it can, if applied independently of the lateral-directional modal requirements of 3.4.5, be met by devising only marginal control power,  $N_{\delta_r}$ , with low yaw damping,  $N_r$ . Low yaw damping effectively increases the initial control sensitivity but at the expense of Dutch roll oscillations. Conversely, extremely large levels of yaw damping, such as that obtained by using a high-gain yaw damper, can result in a loss of initial control response, and an overly sluggish aircraft.



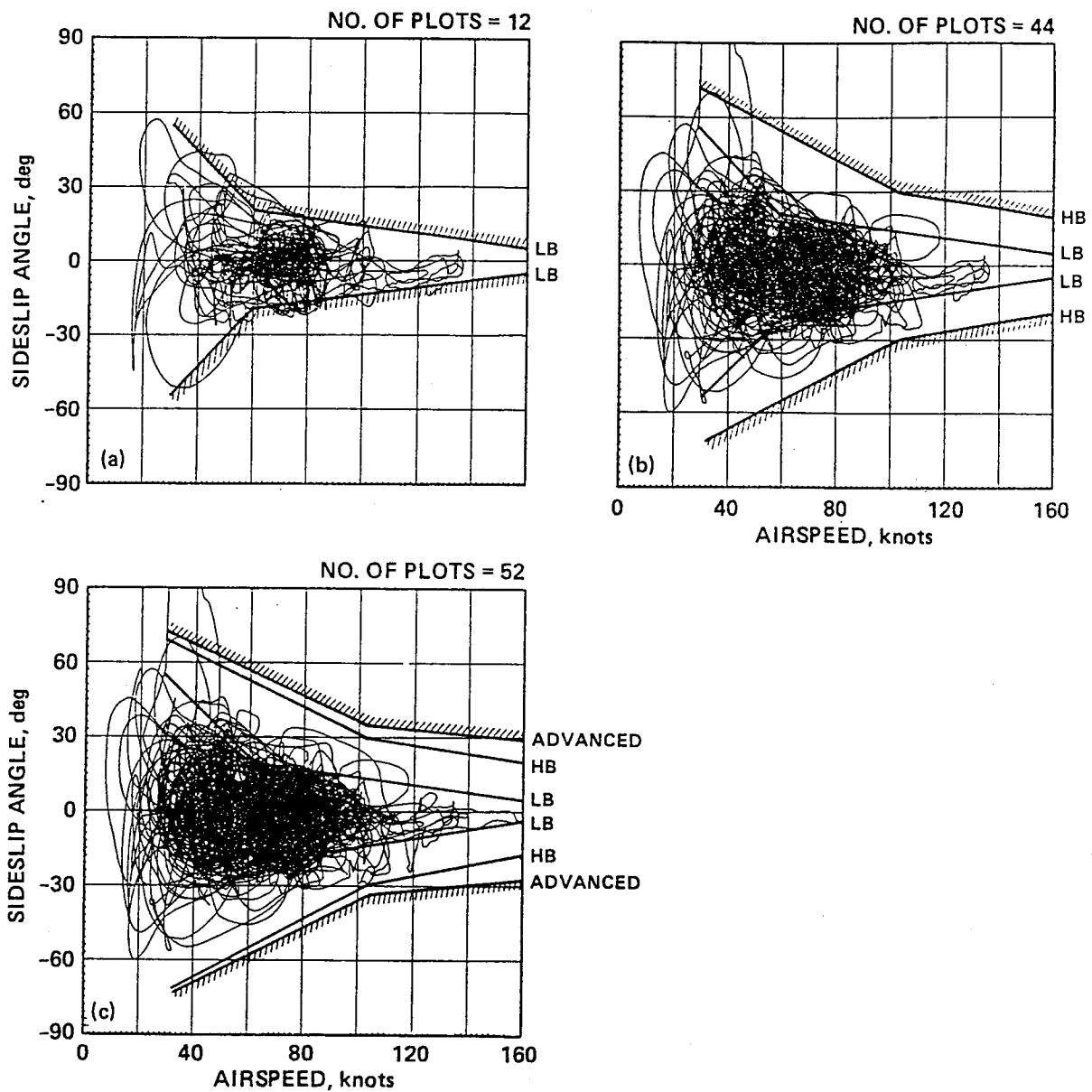


Figure 3(3.4.7.2). Pilot Usage of Sideslip Envelope from HAC II Simulation (from Reference 182) (a) Low Baseline (b) High Baseline (c) Advanced

e. Related Previous Requirements

The wording of this requirement is taken from MIL-F-83300, Reference 29, 3.3.10.1. Directional control power is specified in 3.3.5, 3.3.6, and 3.6.1.1 of MIL-H-8501A, Reference 31. This reference relates required yaw attitude in one second to aircraft weight, i.e.,

$$\psi(1) = \frac{K}{(W + 1000)^{1/3}}$$

where K = 110 for a one-inch input and 330 for full rudder deflection. The following table shows how the peak yaw attitude limit requirements compare for this paragraph, MIL-F-83300, and MIL-H-8501A, the latter assuming a helicopter weighing 8000 lb:

Source	Yaw attitude at 1 sec following pedal input
Current Spec	16.0 deg (Level 1)
MIL-F-83300	6.0 deg (Level 1)
MIL-H-8501A	15.9 deg (No Level Specified)

The current specification limit is essentially the same as the requirement from MIL-H-8501A, at low speeds, for an 8000 lb helicopter; obviously, the MIL-H-8501A requirement will differ with weight. Both are considerably more stringent than the MIL-F-83300 limit.

a. Statement of Requirement

3.4.7.3 Linearity of Response. There shall be no objectionable nonlinearities in the variation of directional response with yaw control deflection or force.

b. Rationale for Requirement

This is a qualitative requirement to guard against nonlinear yawing responses that might result in degraded handling qualities. It is not intended to prevent any nonlinearities, since some degree of nonlinear control response may be desirable (e.g., to warn the pilot of impending limits in controllability). Since it is not possible to quantitatively determine how much nonlinearity is "objectionable," the requirement is purely qualitative.

c. Supporting Data

None.

d. Guidance for Application

The nature of this paragraph -- relying on the pilot's assessment of "objectionable nonlinearities" -- is such that piloted simulation at least, and certainly flight testing, will be required for compliance. Compliance with this paragraph will be determined during normal testing for the other requirements in the specification, and hence no special guidance is needed for application of the requirement.

e. Related Previous Requirements

This paragraph is similar to 3.3.10.2 of MIL-F-83300 (Reference 29).

a. Statement of Requirement

3.4.7.4 Yaw Control with Speed Change. With the rotorcraft initially trimmed directionally, it shall be possible to maintain heading constant with the yaw controller, with no perceptible change in bank angle, over a speed range of  $\pm 30$  percent of the trim speed or  $\pm 20$  knots ( $\pm 10$  m/s), whichever is less (except where limited by the boundaries of the Service Flight Envelope). For pedal controllers, the yaw control forces shall not be greater than one-half those of Table 4(3.6). For other yaw control types, the forces required shall not be objectionable to the pilot. These requirements must be satisfied without retrimming, and should be accomplished at constant power (altitude varies), and at constant altitude (power varies).

b. Rationale for Requirement

With the inherent shortcomings resulting from control power limitations and from cross-coupling, heading control is a common problem for helicopters during rapid accelerations or decelerations. For example, the decelerating-approach data reviewed in the Supporting Data for Paragraph 3.2.2 indicate that heading control almost always contributed to increased pilot workload. While this is more of a control power than a control force issue, this paragraph is intended to assure that small speed changes from trim do not result in a requirement for excessive forces. A qualitative requirement covers any unconventional controller, for example a multi-axis sidestick.

c. Supporting Data

There is no supporting data for this paragraph. The speed ranges over which the paragraph is applicable should be evaluated and revised if necessary.

d. Guidance for Application

None.

e. Related Previous Requirements

This requirement, with the exception of the statement concerning unconventional yaw controls, is taken directly from 3.3.10.3 of MIL-F-83300 (Reference 29). A similar requirement is given in 3.3.5.1 of MIL-F-8785C (Reference 66).

a. Statement of Requirement

3.4.8 Lateral-Directional Stability

3.4.8.1 Lateral-Directional Oscillations. The frequency,  $\omega_n$ , and damping ratio,  $\zeta$ , of the lateral-directional oscillations following a yaw control doublet, shall exceed the minimums specified on Figure 9(3.4). The requirements shall be met with controls fixed and with them free for oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation.

b. Rationale for Requirement

The limits given on Figure 9(3.4) are based primarily on the Dutch roll mode requirements for conventional airplanes in MIL-F-8785C (Reference 66). Any reference to "Dutch roll" per se has been removed from this specification, because it is not always appropriate to refer specifically to a "Dutch roll" oscillation for rotorcraft that corresponds exactly to the conventional airplane Dutch roll. Instead, these requirements are to be applied to any observable oscillation in the lateral or directional axis.

It is felt that the lateral-directional Dutch roll requirements of MIL-F-8785C are applicable to rotorcraft in forward flight, as long as the appropriate interpretations are made as summarized below.

- The Level 1 boundary for slalom, ground attack and air combat MTEs ( $\zeta \geq 0.35$ ) is based on the low-speed requirement for yaw oscillations in divided attention operations. There are no quantitative data to support this limit in forward flight. The intent is the same, however, since excessive lateral-directional oscillations in a high-workload environment will result in degraded handling qualities at any speed.
- The boundary defined by  $\zeta \geq 0.19$  and  $\zeta\omega_n \geq 0.35$  is the MIL-F-8785C Dutch roll limit for most Category A Flight Phases, e.g., rapid maneuvering and precision tracking. There is no reason to believe that rotorcraft have any unique damping requirements in forward flight that are distinct from fixed wing aircraft, and hence the boundary has been retained. The representation of this boundary as the Level 2 limit for slalom, ground attack, and air combat is less supportable, and is based on convenience of format.

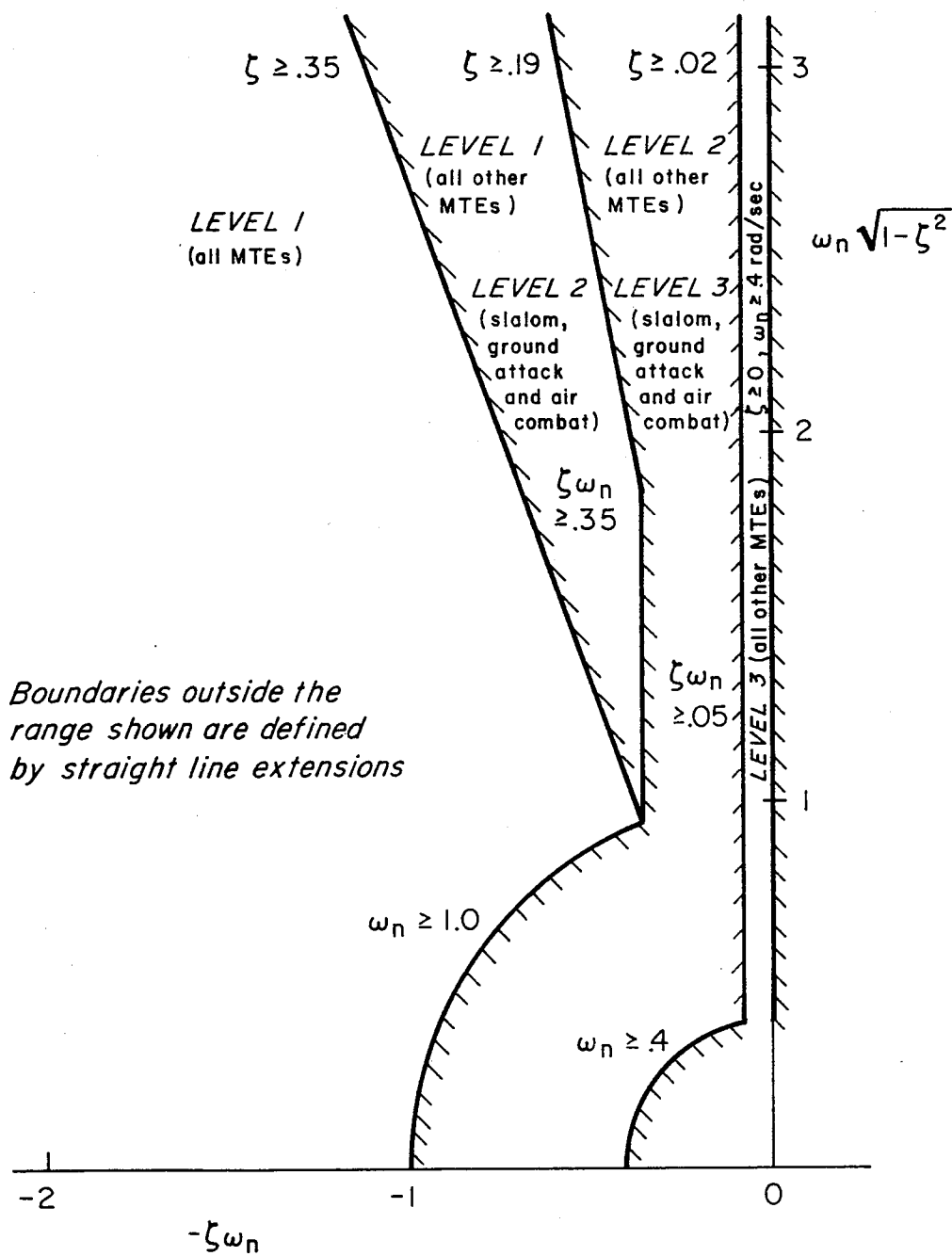


Figure 9(3.4). Lateral-Directional Oscillatory Requirements

- MIL-F-8785C allows  $\zeta \geq 0.08$  for cruise and terminal flight phases. Based on the available data (see Supporting Data), this level of damping is too low, even for undemanding tasks.
- There is no evidence that a unique Level 3 boundary is required for helicopters as distinct from fixed-wing aircraft and hence the MIL-F-8785C limit of at least zero damping and  $\omega_n \geq 0.4$  rad/sec is applied directly.

### c. Supporting Data

There has been considerable interest in the lateral-directional characteristics of helicopters in terrain flight in recent years. For the most part, this interest has resulted in studies addressing the roll mode and roll control power and sensitivity, with little or no investigation of the effects of variations in lateral-directional oscillatory characteristics. As a result, there is almost no data available from which the requirements for oscillatory frequency and damping may be drawn.

The flight experiment of Reference 115 provides some data. This experiment, conducted by the National Research Council of Canada, used a variable-stability Bell 47 helicopter to investigate a variety of lateral-directional characteristics in support of the development of MIL-F-83300 (see Reference 30). Included were several configurations with perfect roll control (i.e., the roll numerator defined by  $\zeta_\phi$ ,  $\omega_\phi$  was devised to cancel the Dutch roll mode) and various combinations of Dutch roll frequency and damping. Five pilots flew a series of 50-kt maneuvers, including visual approaches ending in a 300-ft lateral offset maneuver. The modal response ratio  $|\phi/\beta|_d$  which describes the ratio of rolling and sideslip responses due to the Dutch roll, was set at values of 0.2, 0.75, and 1.5. Results of this experiment are shown in Figure 1.

None of the cases shown in Figure 1 lies in the Level 1 region of Figure 9(3.4). There is a significant effect of  $|\phi/\beta|_d$  on pilot rating, with the mid- to high- $|\phi/\beta|_d$  ratings generally agreeing with the boundaries. This is reasonable support for the Figure 9(3.4) limits, since  $|\phi/\beta|_d = 0.2$  is somewhat low for most helicopters. For example, the following helicopters have the following range of  $|\phi/\beta|_d$  over the forward flight speed range (based on derivatives in Reference 28): UH-1H, 0.37-0.39; OH-6A, 0.33-0.67; BO-105C, 0.45-1.25.

Some supporting information can also be obtained from the flight test reports of References 67 through 80. Most, however, either do not contain sufficient data to determine the lateral-directional oscillatory characteristics or do not have comments regarding these characteristics. Three reports have both: Reference 67 for the Bell Model 209 (AH-1G), Reference 68 for the AH-1G Huey Cobra, and Reference 74 for the OH-6A. In all cases, the flight tasks described apply to low-level or up-and-

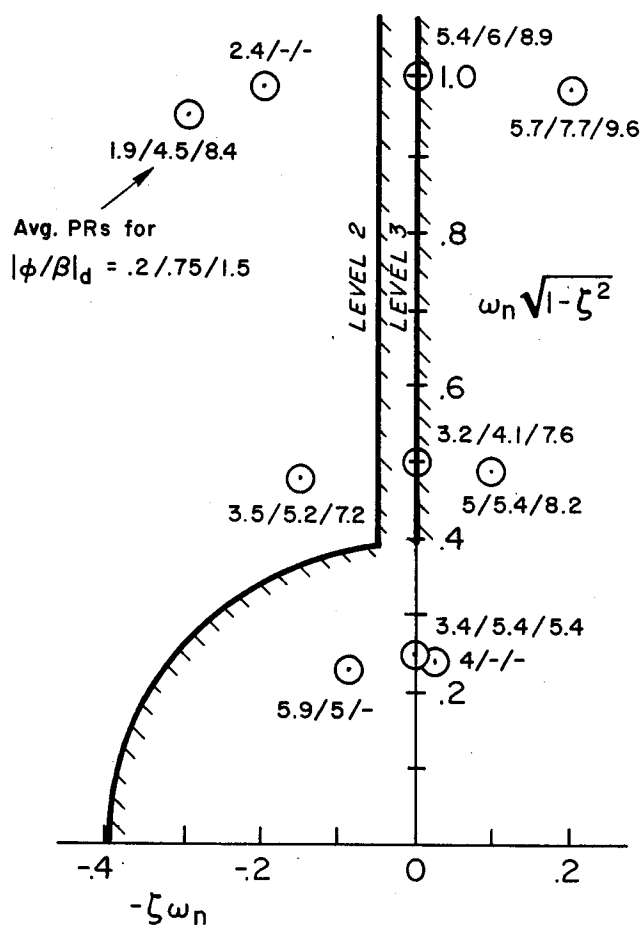


Figure 1(3.4.8.1). Pilot Ratings from Flight Experiment of Reference 115 for Cases with  $\zeta_\phi = \zeta_d$ ,  $\omega_\phi = \omega_d$

away flight (IFR flight for the OH-6A), and thus do not cover the more stringent MTEs.

Figure 2 shows data from References 67 and 68 for the AH-1G, SCAS off. The lateral SCAS for the AH-1G consists of roll rate feedback and results in a heavily damped or over-damped mode. Thus, the SCAS-off data are of more interest here.

For level flight, the AH-1G has somewhat low lateral-directional damping (damping ratio approximately 0.05-0.10). This was considered to be objectionably low in both References 67 and 68. From Reference 67, "at speeds greater than 120 KCAS, the low level of natural damping caused the helicopter to exhibit lateral directional oscillations that were easily excited and could not be damped by the pilot.... Deceleration to a speed where the characteristics were satisfactory (120 KCAS) was uncomfortable and was accompanied by roll excursions of 10 degrees but could be accomplished without unusual skill or corrective action by the pilot." This description indicates at least Level 2 flying qualities, although there is no specific task associated with the evaluation. Based on the flagged symbols in Figure 2, 120 KCAS would correspond to  $\zeta \omega_n = 0.2$ . The proposed requirement specifies a minimum  $\zeta \omega_n$  of 0.35, in



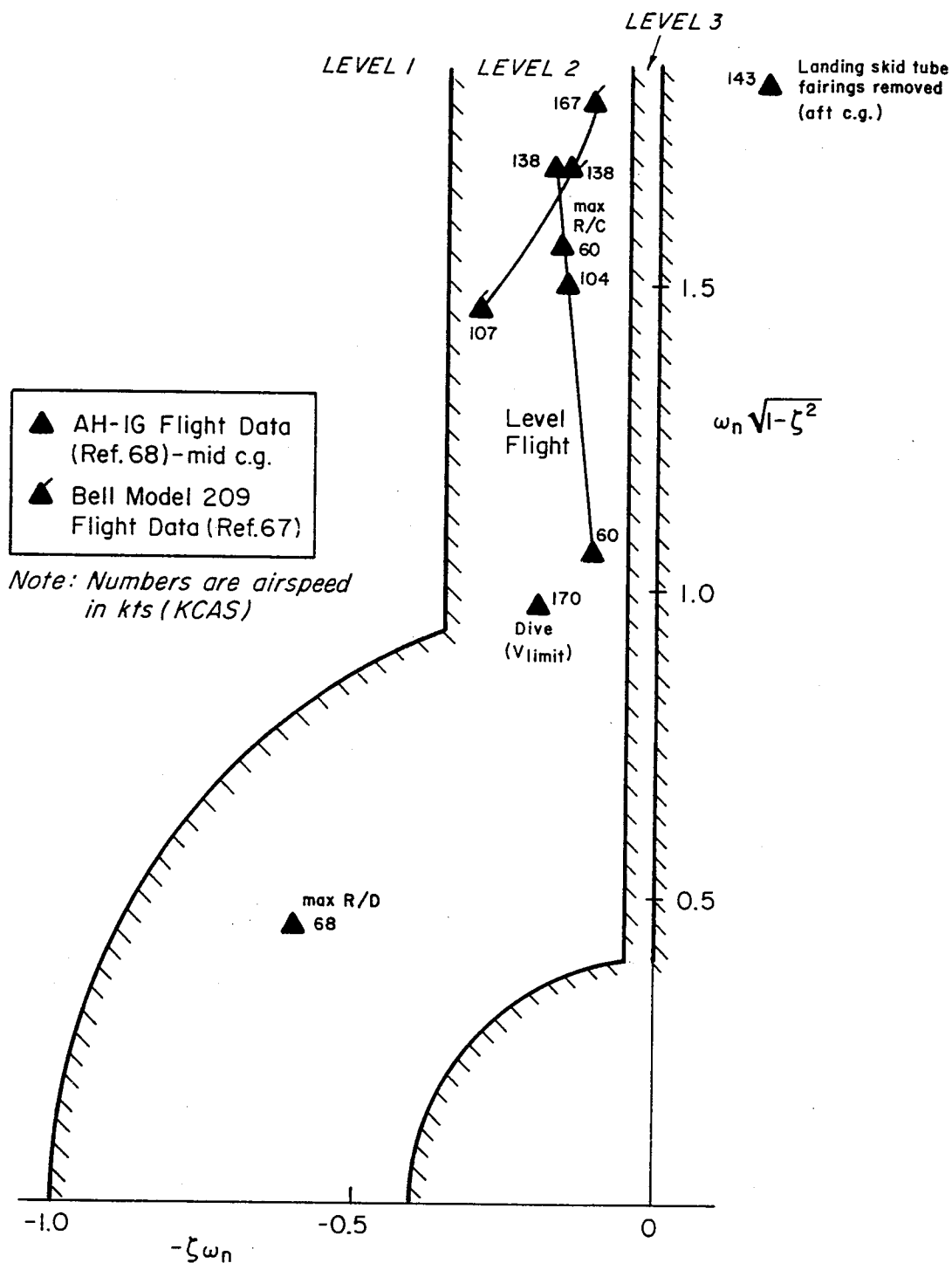


Figure 2(3.4.8.1). Comparison of References 67 and 68 Flight Data for AH-1G with Lateral-Directional Oscillatory Requirements: (Up-and-Away). SCAS Off.

agreement with the flight results for the AH-1G.

Additional AH-1G data are available from Reference 68, including Dutch roll measurements during maximum-rate climbs (max R/C on Figure 2), maximum-rate descents (max R/D), and dives at  $V_{LIMIT}$ . In level flight, the Dutch roll mode was lightly damped ( $\zeta = 0.1$ ). According to Reference 68, "This low damping of the lateral-directional motions with SCAS OFF resulted in objectionable, uncomfortable aircraft motions, particularly at higher airspeeds. In addition, this characteristic, aggravated by the excessively high lateral breakout forces resulted in a pilot induced oscillation in the roll axis at high speeds. This characteristic seriously detracts from mission suitability and makes satisfactory, effective completion of most missions questionable during SCAS OFF operations. The aircraft can be safely returned to base and landed. However, precise flight tasks are very difficult to perform (HQRS 6)." This supports the Level 2 region of Figure 9(3.4), as drawn in Figure 2. For one series of aft-center-of-gravity measurements the landing gear cross-tube fairings were removed, producing divergent oscillations. Reference 68 states that "flights during periods of restricted visibility, such as at night or during instrument conditions with the fairings removed and with SCAS OFF, is not recommended (HQRS 8)." This is clearly Level 3; Figure 9(3.4) specifies that the oscillatory mode must be at least neutrally stable ( $\zeta = 0$ ) for Level 3 operations.

The OH-6A was evaluated for suitability during instrument flight in Reference 74, and a small amount of data can be obtained from this reference. Figure 3 shows the oscillatory characteristics from the flight data. The low damping was considered unsatisfactory for IFR flight in Reference 74: "The lateral-directional gust response was simulated by rapidly displacing the lateral cyclic or directional control either left or right from trim a distance of 1 inch, holding for approximately 0.5 second, and returning to trim. Additionally, flight in light-to-moderate turbulence was conducted.... Flight in turbulent air is characterized by a continual lateral-directional oscillation resulting in a 3- to 4-degree sideslip oscillation, which required moderate pilot compensation to maintain precise heading control (HQRS 4).... The lightly damped lateral-directional gust response characteristics are a shortcoming." For the relatively high frequencies exhibited (Figure 3), the minimum damping ratio is 0.19 for Level 1. This data supports a minimum at least above the OH-6A value of 0.13.

#### d. Guidance for Application

While the requirements of this paragraph were developed to limit oscillations of the lateral-directional Dutch roll mode, they are intended here to cover any oscillation that is encountered in forward flight in response to lateral or directional control or gust inputs. The primary oscillation of interest is the lateral-directional mode corresponding to the "Dutch roll" in fixed-wing airplanes.

For determining compliance, it is standard practice to apply a yaw control pulse or doublet input, and measure the oscillations in the sideslip response. Lateral control inputs do not normally excite the

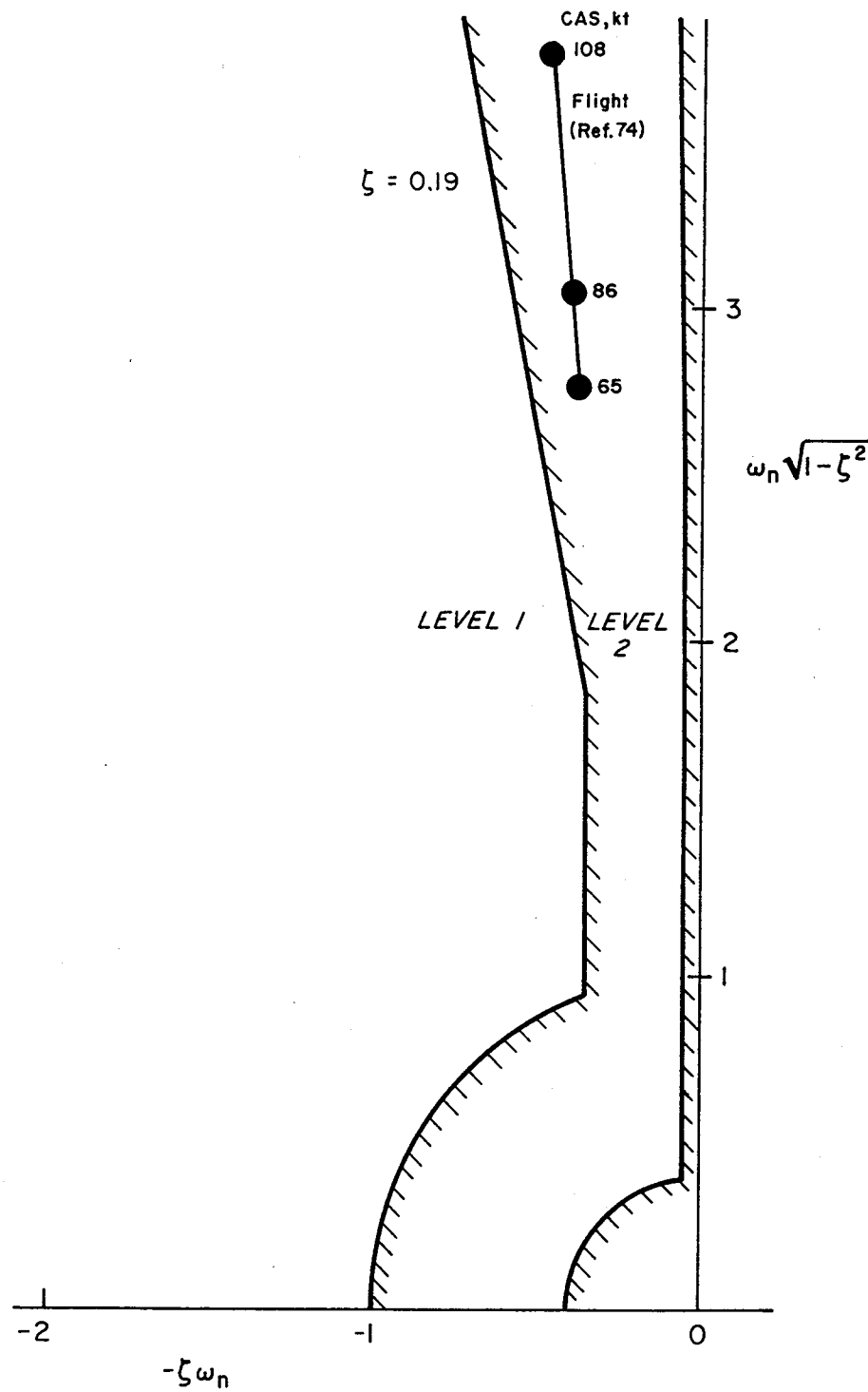


Figure 3(3.4.8.1). Comparison of Flight Data for OH-1A with Lateral-Directional Oscillatory Requirements (Up-and-Away). SCAS Off

lateral-directional oscillation sufficiently to determine its frequency and damping ratio. When the damping ratio of an oscillation in sideslip is greater than 0.35 (the minimum for Level 1 slalom, ground attack and air combat operations), there will be little observable oscillatory response in any trace, and this can be considered to be proof of compliance.

Discussions with members of the rotorcraft industry have reflected some concern over whether it is appropriate to specify limits on "lateral-directional oscillations" at low forward flight speeds -- e.g., at 45 kts. This concern stems from the fact that such oscillations at low speeds tend to be highly coupled between the pitch, roll, and yaw degrees of freedom, and hence are more than just lateral-directional oscillations. While this is true, a review of the dynamics of several helicopters (Reference 28) shows that the sideslip (or heading) response to a yaw control pulse is dominated by a mode that is equivalent to the Dutch roll. As an example, Figure 4 shows the heading responses to an impulse rudder pedal input for the OH-6A, BO-105C, and AH-1G at 40 kts. The solid lines represent the full, uncoupled, six-degree-of-freedom responses, while the dotted lines are the responses of the "Dutch roll" mode alone. It is clear in all cases that the short-term heading response is dominated by the Dutch roll mode. The present wording of the paragraph makes it applicable to this mode, as well as to any other oscillations that may be encountered in the heading response.

Because the various boundaries and labels in Figure 9(3.4) can sometimes be confusing to interpret, Table 1 lists the minimum limits from Figure 9(3.4) in a more readable format. It is felt that Figure 9(3.4) is more directly usable, since it presents the requirements in a graphical form, but the user may wish to consult Table 1 if there are any questions about the Levels in Figure 9(3.4).

#### e. Related Previous Requirements

The limits on lateral-directional oscillations in Figure 9(3.4) are taken directly from Dutch roll requirements in MIL-F-8785C (Reference 66), for Category A operations, i.e., nonterminal tasks requiring rapid maneuvering, precision tracking, or precise flight path control. The helicopter specification, MIL-H-8501A, does not place any limits on lateral-directional oscillations for VFR forward flight, but in IFR flight, the longitudinal and lateral-directional requirements are identical (Paragraph 3.6.1.2, Reference 31). These requirements are quite lenient, especially when compared with the Level 1 limits of Figure 9(3.4), as shown in Figure 5. In fact, the MIL-H-8501A limits lie within the Level 2 boundary in Figure 9(3.4).

The V/STOL specification, MIL-F-83300 (Reference 29), has even more lenient requirements at high frequencies, as indicated on Figure 5, although the limits for low-frequency oscillations are more stringent than those of MIL-H-8501A. It is clear from the data discussed in Supporting Data that both MIL-H-8501A and the Level 1 limits of MIL-F-83300 allow Level 2 responses.

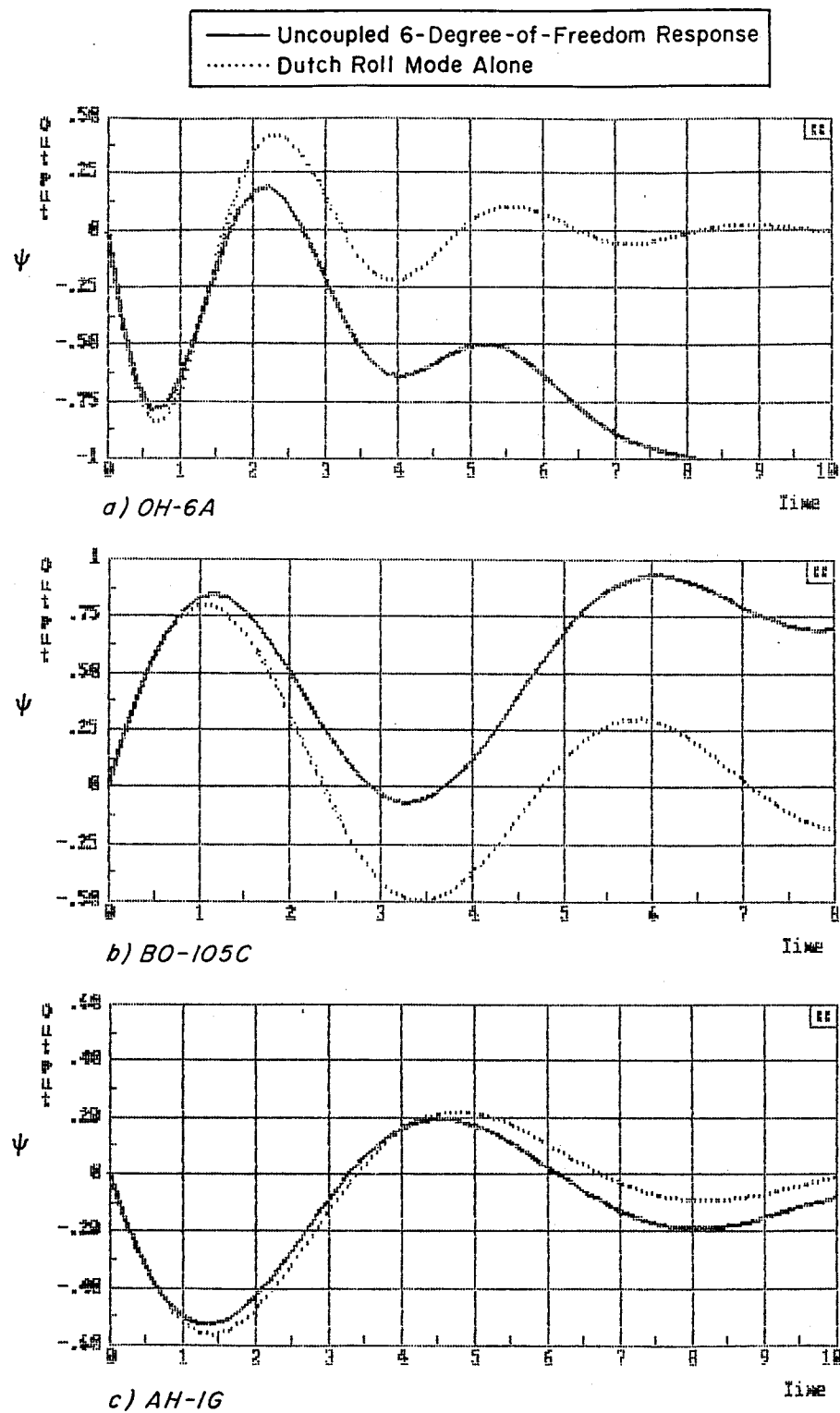


Figure 4(3.4.8.1). Response of Representative Helicopters to Rudder Pedal Impulse Input (40 kt)

TABLE 1(3.4.8.1). LATERAL-DIRECTIONAL OSCILLATION  
REQUIREMENTS OF FIGURE 9(3.4)

LEVEL	MISSION-TASK ELEMENT	MIN $\zeta^*$	MIN $\zeta\omega_n^*$ (rad/sec)	MIN $\omega_n$ (rad/sec)
1	Slalom, Ground Attack, and Air Combat	0.35	--	1.0
	All other MTEs	0.19	0.35	1.0
2	Slalom, Ground Attack and Air Combat	0.19	0.35	1.0
	All other MTEs	0.02	0.05	0.4
3	Slalom, Ground Attack and Air Combat	0.02	0.05	0.4
	All other MTEs	0	--	0.4

\*The governing damping requirement is that yielding  
the larger value of  $\zeta$ .

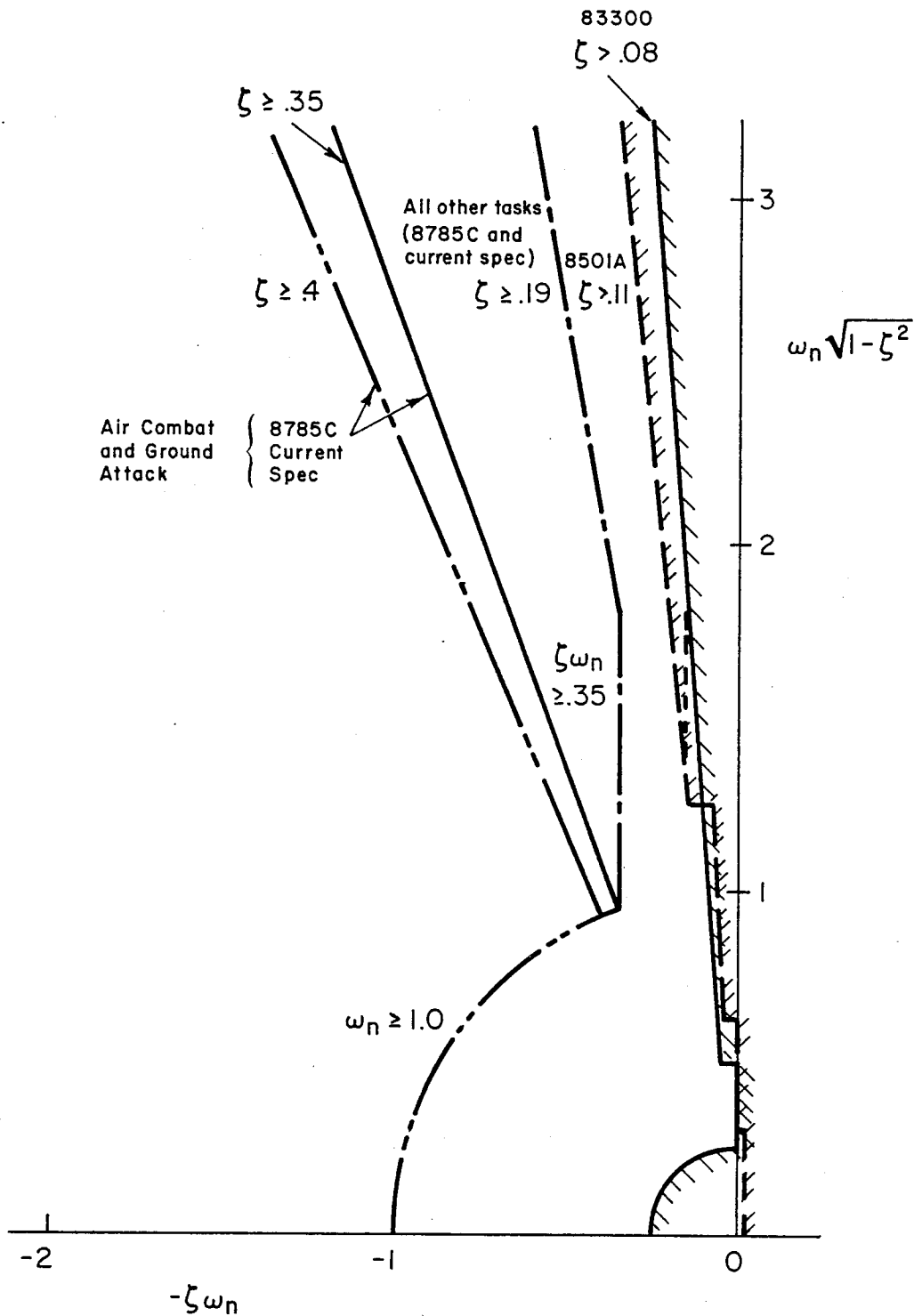


Figure 5(3.4.8.1). Comparison of Lateral-Directional Oscillation Requirements of MIL-H-8501A (IFR) with Level 1 Limits of MIL-F-83300, MIL-F-8785C, and Current Specification

a. Statement of Requirement

3.4.8.2 Spiral Stability. Following a lateral pulse control input, the time for the bank angle to double amplitude shall be greater than the following:

Level 1:	20.0 seconds
Level 2:	12.0 seconds
Level 3:	4.0 seconds

These requirements shall be met with the cockpit controls free and the aircraft trimmed for straight and level flight. The values specified apply to an exponential divergence and should not depend on the size of the control input. If the variation of roll angle with time is linear following the pulse control input, this requirement is satisfied.

b. Rationale for Requirement

Spiral instability is normally manifested as a "nuisance" mode during divided attention operations. For those Response-Types where Attitude Hold is required by Paragraph 3.2.2, this requirement will easily be met. The requirement is intended to assure that spiral instability does not result in an unacceptable increase in pilot workload when flying without benefit of Attitude Hold.

No unique distinction can be identified between the spiral mode requirement for helicopters in forward flight and fixed-wing aircraft. Therefore, the fixed wing requirements are used from the V/STOL specification, MIL-F-83300.

c. Supporting Data

Since this requirement was taken directly from MIL-F-83300, the supporting data comes from the BIUG for that specification (Reference 30).

A study of airworthiness requirements for STOL aircraft (Reference 165) recommends that the spiral stability, as measured by time to double amplitude of bank angle, should not be less than 20 seconds for satisfactory operation, nor less than 5 seconds for safe operation. The data used in Reference 165 are presented on Figure 1, and compared with the requirements of this Paragraph.

Reference 166 presents the results of an in-flight investigation primarily directed at determining the effect of the spiral mode on response of an aircraft in cruising flight. It also investigated the relationships between acceptable spiral stability limits and lateral-directional dynamics and control characteristics. Although the configurations evaluated do not meet the Level 1 lateral-directional requirements of the specification, the results do indicate that pilot rating can be acceptable for  $T_2 > 20$  seconds and degrades significantly as  $T_2$



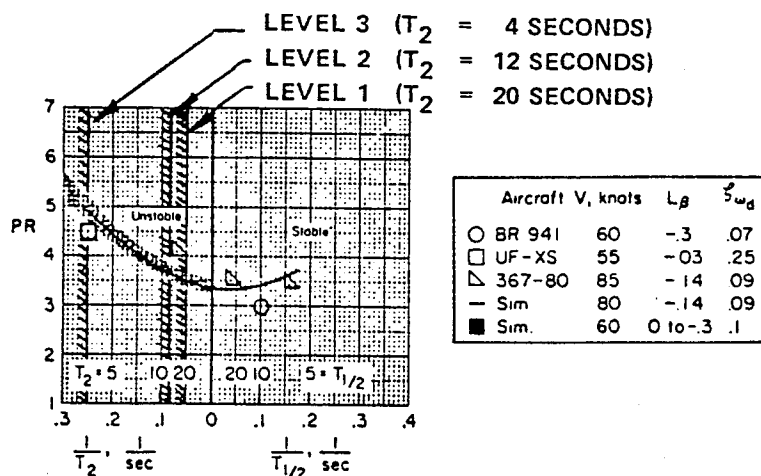


Figure 1(3.4.8.2). Variation of Pilot Rating with Spiral Stability (from Reference 165)

decreases. The results of this experiment are in general agreement with the spiral stability requirements.

A review of the analytical models for five helicopters in Reference 28 suggests that spiral instability (due to aerodynamics alone) is not a problem for these helicopters. It is possible, of course, that non-linearities in the aerodynamics, adverse feedbacks in the SCAS, or excessive friction and freeplay in the cyclic, could cause a divergence in bank angle. By the wording of this requirement, a divergence caused by any of these sources would be considered to be a spiral instability.

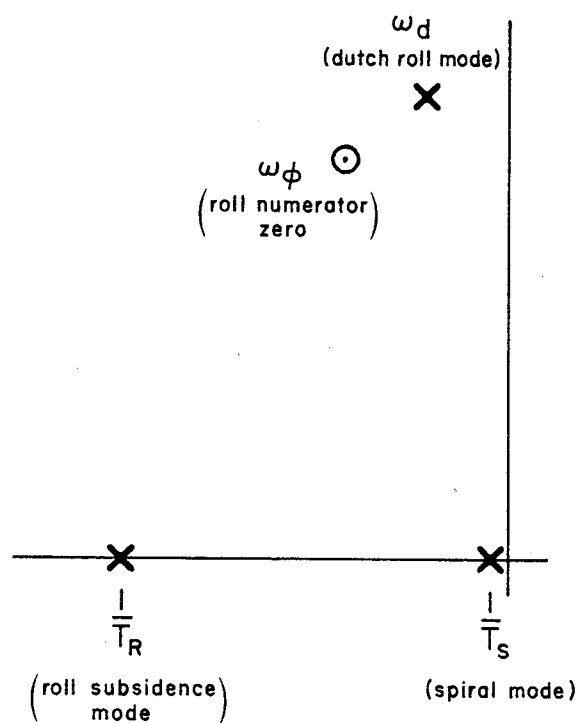
#### d. Guidance for Application

Proper testing and interpretation of test results for spiral stability requires some fundamental considerations which are discussed below.

Spiral stability refers to the location of one root in the lateral-directional equation.\* The bank angle transfer function is shown in Figure 2, where it can be seen that the spiral mode consists of a real root near the origin. The spiral mode is stable when this root resides in the left-hand-plane, neutral if it's at the origin, and unstable when it's in the right half plane. The bank angle time responses to ideal step and pulse lateral control inputs for each of these spiral mode locations are shown in Figure 3, where it can be observed that:

---

\*The characteristic equation is the denominator of all the lateral and directional responses. Each of these responses (yaw rate, bank angle, lateral acceleration, and sideslip) has a different numerator, but the denominator remains the same. Therefore, the poles of the denominator are termed the characteristic modes: spiral, roll subsidence, and dutch roll.



$$\frac{\phi}{\delta_a} = \frac{L_{\delta_a} (s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2)}{\underbrace{(s + 1/T_S)}_{\text{spiral}} \underbrace{(s + 1/T_R)}_{\text{roll subsidence}} \underbrace{(s^2 + 2\zeta_d \omega_d s + \omega_d^2)}_{\text{dutch roll}}}$$

Figure 2(3.4.8.2). Lateral-Directional Roots and Bank Angle Transfer Function

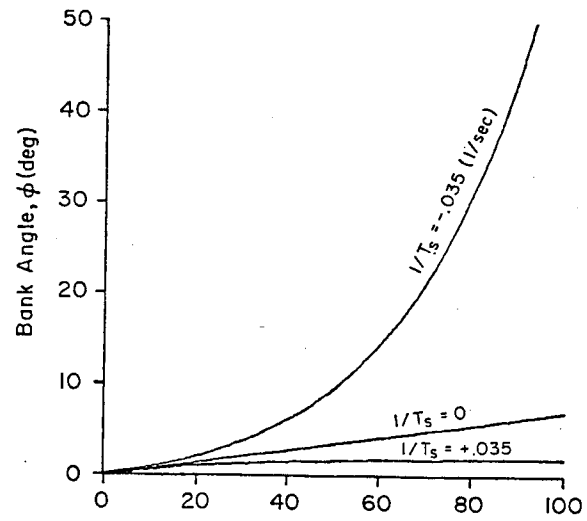
- For all inputs, the response is concave up when the spiral mode is in the right half plane. This is defined as unstable.
- For a pulse input (Figure 3b):
  - 1) The bank angle response is concave up when the spiral mode is in the right-half-plane. This is defined as unstable, and is characterized by the fact that the time to double amplitude is constant for all input magnitudes, and times.
  - 2) The bank angle response is constant when the spiral mode is at the origin. This is defined as neutral.
  - 3) The bank angle returns to zero when the spiral mode is in the left-half-plane. This is defined as stable.
- For a step input (Figure 3a):
  - 1) The bank angle response is concave up when the spiral mode is in the right-half-plane. As with the pure pulse, this is defined as unstable, but the time to double amplitude is not constant, and therefore is not a measure of the instability. This is a consequence of the fact that the response is not pure, but is the sum of the spiral and roll subsidence modes. This is discussed in greater detail below.
  - 2) The bank angle response is in linear divergence when the spiral mode is neutral. This observation is highly pertinent to the proper interpretation of the data taken in flight test.
  - 3) The bank angle is constant when the spiral mode is stable.

When the spiral is unstable (response is concave up), the time required to double amplitude ( $T_2$ ) is constant, providing the input is removed. This is illustrated in Figure 3c, where the time to double is seen to be 20 seconds regardless of the size of the pulse input, or at what time the measurement is taken. On this basis, the time to double amplitude is a standard measure of the degree of instability of the spiral mode, and in fact the spiral root ( $1/T_s$ ) is well approximated as,

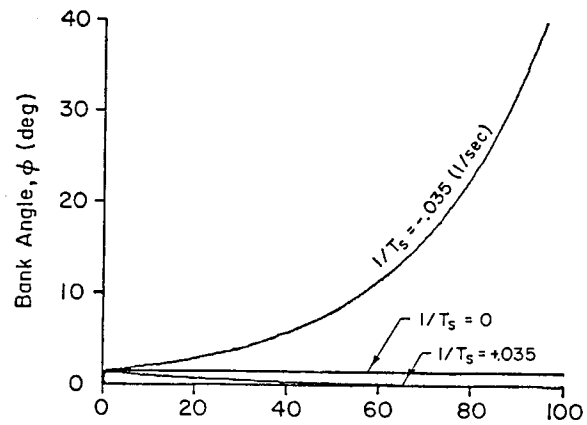
$$1/T_s = -.694/T_2$$

However,  $T_2$  is only valid as a measure of the spiral instability if 1) the response is concave up, and 2) the input is a pure lateral cockpit control pulse with zero residual offset due to friction. It is a common error to attempt measurement of time to double amplitude for cases where both these conditions are not met. One test of the validity of the

a) Step Input



b) Pulse Input



c) Pulses of Increasing Magnitude

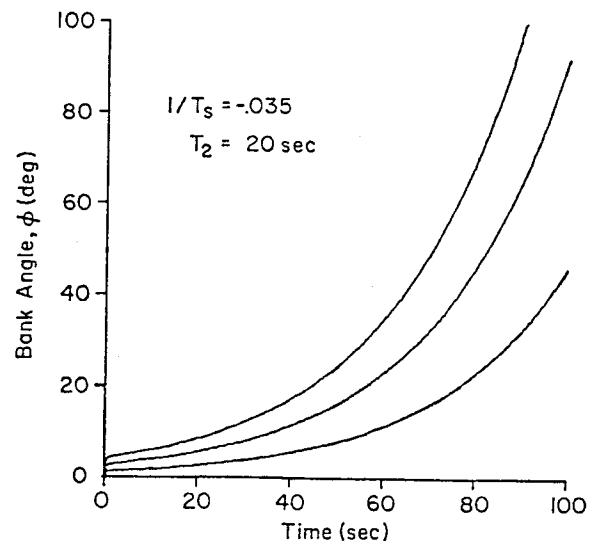


Figure 3(3.4.8.2). Generic Spiral Response Characteristics

measurement is to make it at two or more points in the response; the answer should be the same. A common problem that occurs when making flight test determinations of spiral stability is that a pulse cockpit control input results in a small step at the control actuator due to control system friction. As a consequence, the bank angle time history is the result of a step input, not the intended pure pulse. On this basis, it is appropriate to consider the generic characteristics of the bank angle response to a small lateral cockpit control step, as shown in Figure 3a, when analyzing the flight test data.

e. Related Previous Requirements

This requirement is taken from MIL-F-83300 (Reference 29, Paragraph 3.3.7.3). MIL-F-8785C (Reference 66) contains a similar requirement (3.3.1.3).

It should be noted that the helicopter requirements on spiral stability are equal to the most lenient of the spiral stability requirements of MIL-F-8785C. That is, the helicopter specification requirement on spiral stability is the same as the Class II and Class III requirement for all Flight Phases and Class I and Class IV requirement for Flight Phase Categories B and C.

a. Statement of Requirement

3.4.9 Lateral-Directional Characteristics in Steady Sideslips.  
The requirements of 3.4.9.1 through 3.4.9.3.1 are expressed in terms of characteristics in yaw-control-induced, steady zero-yaw-rate sideslips with the aircraft trimmed for straight and level flight. Sideslip angles to be demonstrated shall be to the limits of the Operational Flight Envelope.

b. Rationale for Requirement

The requirements of 3.4.9.1 through 3.4.9.3.1 determine the sideslip characteristics for the rotorcraft. This paragraph states the angles over which these requirements must be applied. The wording is based on 3.3.11 of MIL-F-83300.

c. Supporting Data

None.

d. Guidance for Application

None.

e. Related Previous Requirements

Similar requirements exist in the CTOL, VSTOL, and helicopter specifications. The major differences are the sideslip angles required for compliance. For example, MIL-F-8785C (Reference 66) states the following (3.3.6):

Requirements... apply at sideslip angles up to those produced or limited by:

- a. Full yaw-control-pedal deflection, or
- b. 250 pounds of yaw-control-pedal force, or
- c. Maximum roll control or surface deflection...

MIL-F-83300 (Reference 29, 3.3.11) requires that

sideslip angles to be demonstrated shall be the lesser of 25 degrees or  $\sin^{-1}$  (30/airspeed in knots), or those limited by structural limitations, or the yaw control and roll control force limits of Table 4(3.6). In any event, the minimum sideslip to be demonstrated shall be the lesser of 15 degrees or  $\sin^{-1}$  (30/airspeed in knots).

From MIL-H-8501A (Reference 31, 3.3.9):

The helicopter shall possess positive, control fixed, directional stability, and effective dihedral in both powered and autorotative flight at all forward speeds above 50 knots,  $0.5 V_{\max}$ , or the speed for maximum rate of climb, whichever is the lowest. At these flight conditions with zero yawing and rolling velocity, the variations of pedal displacement... with steady sideslip shall be stable... up to full pedal displacement in both directions, but not necessarily beyond a sideslip angle of 15 degrees at  $V_{\max}$ , 45 degrees at the low speed determined above, or beyond a sideslip angle determined by a linear variation with speed between these two angles.

In the current requirement, reference to pedal displacement or other yaw control input has been removed; the required sideslip for demonstration should not be artificially limited by control limits.

a. Statement of Requirement

3.4.9.1 Yaw Control in Steady Sideslips. For the sideslips specified in 3.4.9, right yaw control deflection and force shall be required in left sideslips and left yaw control deflection and force shall be required in right sideslips. For Levels 1 and 2, the following requirements apply. Between sideslip angles of  $\pm 15$  degrees, or the limits of the Operational Flight Envelopes, whichever is less, the variation of yaw controller deflection and force shall be essentially linear with sideslip. For larger sideslip angles, an increase in yaw control deflection shall always be required for an increase in sideslip, and the following requirements shall apply:

Level 1: The gradient of sideslip angle with yaw control force shall not reverse slope.

Level 2: The gradient of sideslip angle with yaw control force is permitted to reverse slope provided the sign of the yaw control force does not reverse.

The term gradient does not include that portion of the yaw control force versus sideslip-angle curve within the preloaded breakout force or friction band.

b. Rationale for Requirement

This requirement is intended to assure good control/response characteristics. Both conventional pedal and sidestick yaw controllers are recognized to the extent possible. Some differences may be justified in the specifics of the requirements for these two very different controller configurations, but there is no information on which to base the different requirements, and the fundamental purpose of this paragraph should be met no matter what the controller type.

c. Supporting Data

None.

d. Guidance for Application

None.

e. Related Previous Requirements

This paragraph is a rewording of 3.3.11.1 from MIL-F-83300 (Reference 29). It is closely related to 3.3.9 of MIL-H-8501A (Reference 31), and MIL-F-8785C (Reference 66), 3.3.6.1, is similar.



a. Statement of Requirements

3.4.9.2 Bank Angle in Steady Sideslips. For the sideslips specified in 3.4.9, an increase in right bank angle shall accompany an increase in right sideslip, and an increase in left bank angle shall accompany an increase in left sideslip.

3.4.9.3 Lateral Control in Steady Sideslips. For the sideslips specified in 3.4.9, left lateral control deflection and force shall be required in left sideslips and right control deflection and force shall be required in right sideslips. For Levels 1 and 2, the variation of lateral control deflection and force with sideslip angle shall be essentially linear.

3.4.9.3.1 Positive Effective Dihedral Limit. For Level 1, positive effective dihedral (right roll control for right sideslip and left roll control for left sideslip) shall never be so great that more than 75 percent of the roll control power available to the pilot and no more than 49 N (11 pounds) of roll control force (for centerstick controllers) are required for sideslip angles which might be experienced in performing the required Mission-Task-Elements. The corresponding limits for Level 2 shall be 90 percent and 60 N (13.5 pounds).

b. Rationale for Requirements

This set of requirements, taken essentially verbatim from MIL-F-83300, specify the bank angle and rolling moment characteristics to be encountered in steady sideslips. The control power and force limits of 3.4.9.3.1 have been increased from the similar MIL-F-83300 requirement because of the wider sideslip range specified in 3.4.9.

c. Supporting Data

The Helicopter Air Combat (HAC II) simulation of Reference 182 demonstrated the value of some degree of positive effective dihedral by enhancing the rolling capability of the simulated helicopter with sideslip.

d. Guidance for Application

None.

e. Related Previous Requirements

These three requirements come from MIL-F-83300 (Reference 29), 3.3.11.2, 3.3.11.3, and 3.3.11.3.1. Very similar statements are specified in MIL-F-8785C (Reference 66), 3.3.6.2, 3.3.6.3, and 3.3.6.3.2. In addition, MIL-H-8501A (Reference 31) incorporates all of these requirements into a single paragraph, 3.3.9, quoted in the Related Previous Requirements discussion for 3.4.9.

a. Statement of Requirement

3.4.10 Pitch, Roll, and Yaw Responses to Disturbance Inputs.  
Pitch, roll, and yaw responses to inputs directly into the control surface actuator shall meet the bandwidth limits based on cockpit controller inputs specified in Paragraphs 3.4.1.1 (pitch), 3.4.5.1 (roll), and 3.4.7.1 (yaw). If the bandwidth and phase delay parameters based on inputs to the control surface actuator can be shown by analysis to meet the cockpit control input requirements, no testing is required. This requirement shall be met for Level 1, and relaxed according to Paragraph 3.2.12 for Levels 2 and 3.

b. Rationale for Requirement

This requirement is identical to Paragraph 3.3.2.3 and 3.3.7, except that it applies to forward flight.

c. Supporting Data

See Paragraph 3.3.2.3.

d. Guidance for Application

See Paragraph 3.3.2.3.

e. Related Previous Requirement

None.

a. Statement of Recommended Requirement\*

3.5 TRANSITION OF A VARIABLE CONFIGURATION ROTORCRAFT  
BETWEEN ROTOR-BORNE AND WING-BORNE FLIGHT

The transition is defined herein as the region in which a variable configuration rotorcraft is reconfigured between wing-borne and rotor-borne flight by means of changing the rotor mast angle between essentially horizontal and vertical orientations with respect to the longitudinal axis of the aircraft. A nominal transition corridor defined by a schedule of mast angle shall be specified by the manufacturer. The handling qualities specified herein must be satisfied over an airspeed range of  $\pm 25$  knots about this nominal profile. Unless otherwise noted, the requirements for Hover and Low Speed (Paragraph 3.3) and Forward Flight (Paragraph 3.4) must be satisfied during the transition, in addition to the special requirements contained in this section. The requirements for "UCE>1 and/or divided attention operations" shall apply for Hover and Low Speed, and the requirement for "IMC operations" shall apply for Forward Flight. The speed which separates the Hover and Low Speed and Forward Flight requirements is revised from 45 knots groundspeed to 45 knots airspeed for this section only.

b. Rationale for Recommended Requirement

The basic handling qualities requirements during transition are no different than any other flight condition except for those aircraft responses directly related to variations in the rotor mast angle. Therefore, the requirements in Sections 3.3 and 3.4 apply during transition, and the special requirements in this section are directed toward possible handling deficiencies that would arise directly from mast angle variations.

The division between the Hover and Low Speed requirements (3.3) and the Forward Flight requirements (3.4) is based on 45 knots of ground speed. The rationale for using groundspeed is that the piloting tasks for Hover and Low Speed consist almost exclusively of maneuvers with respect to outside objects. However, the transition between wing-borne and rotor-borne flight is conducted with respect to airspeed, and hence the appropriate selection of requirements between Paragraphs 3.3 and 3.4 should be based on airspeed for the transition. That is, at airspeeds below 45 knots, the Hover and Low Speed requirements of 3.3 apply, whereas the Forward Flight requirements of 3.4 apply above an airspeed of 45 knots. However, if operations are to be conducted at fixed intermediate mast angles, the rotorcraft is no longer defined to be in transition, and the original definitions based on groundspeed apply (see Paragraph 2.6). This leads to a necessity for considering flight at low

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\*Requirements for transition are not given in the specification. Instead, recommended requirements are included to allow this background document to be utilized for variable configuration rotorcraft (e.g., tilt-rotor).

and zero airspeeds (flight in tailwinds) which will have the effect of limiting Low Speed flight to mast angles very near 90 degrees. It is felt that a necessity to convert mast angle to avoid a stall while operating at Low Speed, and with respect to outside objects, represents excessive pilot workload.

c. Supporting Data

A minimum transition corridor width of plus or minus 25 knots of airspeed is specified on the basis of recommendations from the Reference 193 moving-base piloted simulation study conducted at NASA Ames. That study investigated the effect of a narrowing conversion corridor on pilot opinion. The results of the experiment are somewhat clouded by the fact that the baseline XV-15 model was, for various reasons, Level 2. The pilot ratings do, however, degrade significantly when the conversion corridor is narrowed. As stated in Reference 193, the baseline XV-15 conversion corridor, which had a minimum width of approximately plus or minus 25 kts, was deemed acceptable. A reduction of this minimum corridor width to approximately plus or minus 12 kts caused a significant increase in the pilot workload during the decelerating approach evaluation task.

On the basis of the results from the Reference 193 experiment, the conversion corridor should be at least 50 kts wide at its narrowest point. In addition, the Bell/Army/NASA XV-15 tilt rotor has been shown to successfully operate with reportedly low pilot workload with a plus or minus 25 knot transition corridor.

d. Guidance for Application

Compliance with the transition criteria, as well as the criteria for Hover and Low Speed (3.3) and Forward Flight (3.4), should be demonstrated over a range of airspeeds of at least plus or minus 25 knots about the nominal transition schedule of mast angle vs. airspeed. If continuous operations are expected at intermediate mast angles in the vicinity of 45 knots airspeed, the requirements of Sections 3.3 and 3.4 should be met independent of this section.

e. Related Previous Requirements

This requirement is similar to that in MIL-F-83300. The current requirement offers additional guidelines on its application.

There is no related requirement in MIL-H-8501A.

a. Statement of Recommended Requirement

3.5.1 Allowable Transition Controller-to-Pitch Attitude Coupling.  
The peak pitch attitude excursion following the most rapid possible change in the transition controller, with controls free, shall be less than the limits specified in Table 1(3.5) during transition. These limits shall apply for transition controller inputs of at least 10 degrees throughout the specified transition profile.

TABLE 1(3.5). ALLOWABLE TRANSITION  
CONTROLLER-TO-PITCH ATTITUDE COUPLING

LEVEL	$\frac{\theta_{\text{peak}}}{\delta_m}$ (deg/deg)
1	$\pm 0.25$
2,3	$\pm 0.50$

b. Rationale for Recommended Requirement

The primary objectives of the transition controller are to:

1. Reconfigure the rotorcraft.
2. Change speed rapidly.

Rapid changes in the transition controller to achieve the above objectives should not result in undesirable pitch attitude excursions. Such excursions would be undesirable particularly in a high workload task, such as a decelerating approach to IMC conditions.

Pitch attitude coupling is limited in terms of the peak attitude excursions per unit deflection of the transition controller. This metric was found to be a reasonable parameter for defining transition controller coupling in a V/STOL aircraft transition study (Reference 3).

c. Supporting Data

Transition controller-to-pitch attitude coupling is characterized by the ratio of peak pitch attitude change to a step transition controller input ( $\theta_{\text{peak}}/\delta_m$ ). The specification limits on this ratio are drawn from the results of the moving-base piloted simulation of V/STOL transition (conducted on the NASA Ames VMS) detailed in Reference 3. Two types of transition were simulated -- a transition to a STOL configuration with a 66 degree thrust angle (equivalent to a 66 degree mast angle), to allow a deceleration from 250 to 90 knots for a constant speed instrument approach, and a deceleration to hover after breakout using thrust vectoring. The transition to STOL was conducted during the

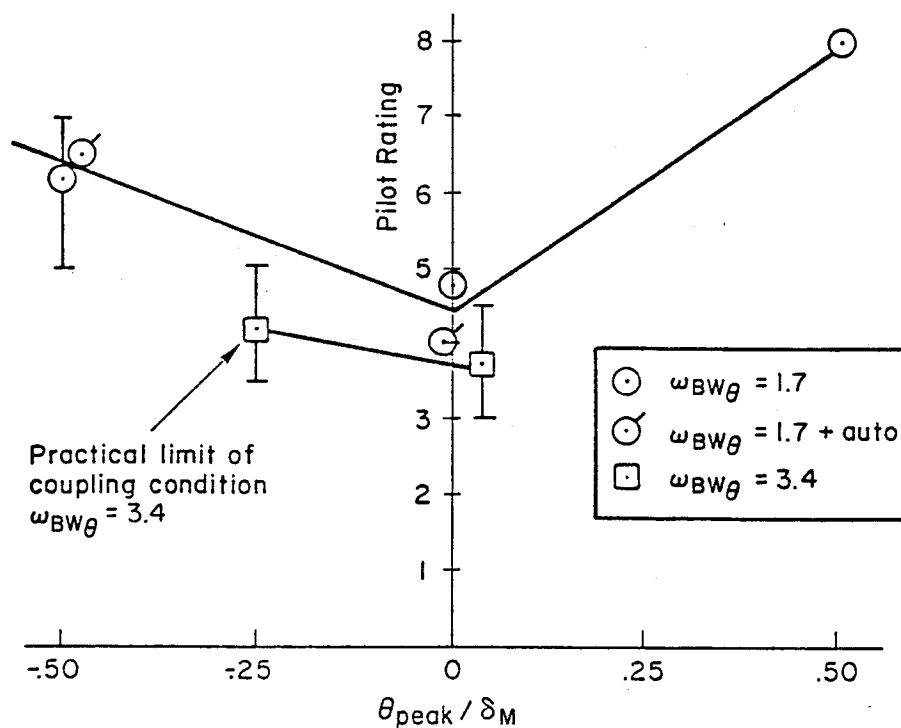
turn-on to the ILS final approach course, which was a high workload task (see Reference 3 for details). Several levels of transition controller-to-pitch-attitude coupling and pitch attitude bandwidths were evaluated under simulated VMC and IMC conditions. The magnitude of the coupling was maintained at an approximately constant level during all variations of airspeed and trim transition controller angle to ensure that the HQRs related to a given level of coupling. The Cooper-Harper pilot rating results are shown plotted as a function of the transition controller-to-pitch-attitude coupling in Figure 1. Only one level of coupling was simulated with the high pitch attitude bandwidth configurations ( $\omega_{BW\theta} = 3.4$  and  $3.8$  rad/sec) because it was not possible to increase the coupling beyond only a moderate level for such high values of attitude stiffness. The most significant degradations in pilot ratings due to coupling all exist for marginal pitch attitude bandwidths (Figure 1). These degradations evoked pilot commentary that indicated that the primary effect of the coupling was perceived as loose attitude control, and that the coupling itself was not an excessively objectionable feature. These results indicate that the incorporation of attitude bandwidths somewhat above the Level 1 limits has the important side effect of reducing the negative impact of interaxis coupling, as well as the magnitude of the coupling itself.

The data trends in Figure 1 indicate that levels of coupling above 0.50 cause a distinct degradation in pilot ratings, and this is taken as the Level 1 limit. There is some evidence in Figure 1a that positive values of coupling may be more of a problem than negative values, but this effect is not included in the requirement because it is supported by only one data point.

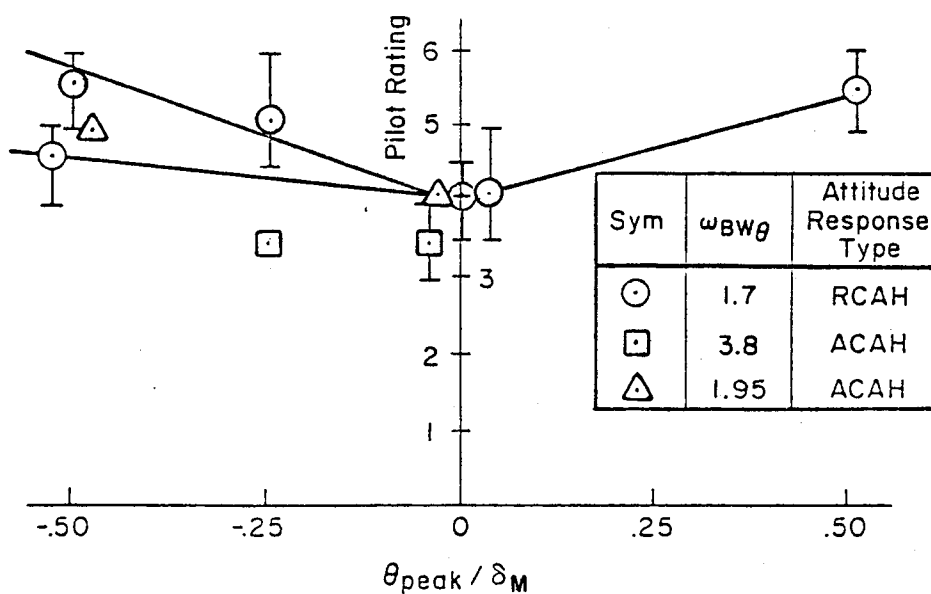
The maximum transition controller-to-pitch attitude coupling levels simulated in the Reference 3 experiment are representative of fixed-wing V/STOL aircraft. Much larger values of coupling could be encountered in variable configuration rotorcraft, such as a tilt-rotor, where variations in flow over the tail can induce very large values of coupling between rotor mast angle and pitch attitude. A moving-base piloted simulation investigation (also on the NASA Ames VMS) of a tilt-rotor aircraft (XV-15) in a decelerating approach under IFR conditions is reported in Reference 193. Three levels of transition controller-to-pitch attitude coupling were evaluated, but unlike the Reference 3 study, the level of coupling was allowed to vary with airspeed and trim transition controller angle (as appropriate for the XV-15). The ratio of  $\theta_{\text{peak}}/\delta_m$  derived from the transfer functions given in Reference 193 for the three levels of coupling and three representative flight conditions are shown in Table 2. The transition controller is the nacelle angle (i.e., rotor mast tilt angle). Unfortunately, the data given in Reference 193 are for the unaugmented aircraft, and consequently the coupling values in Table 2 do not reflect the effect of the SCAS.

The pitch attitude bandwidth was approximately 2 rad/sec for the intermediate flight condition of 60 deg/110 kts, assuming a 130 msec time delay to represent simulator visual and motion system delays.

Pilot ratings for the high workload task where complete conversion from wing-borne flight to rotor-borne flight was accomplished on the



a) Transition to STOL (IMC conditions) (RCAH)



b) Transition to Hover (VMC conditions)

Figure 1(3.5.1). Summary of Thrust Vector Angle to Pitch Attitude Coupling (from Reference 3)



TABLE 2(3.5).  $\theta_{\text{peak}}/\delta_m$  COUPLING (DEG/DEG) FROM  
REFERENCE 193 (UNAugMENTED AIRCRAFT)

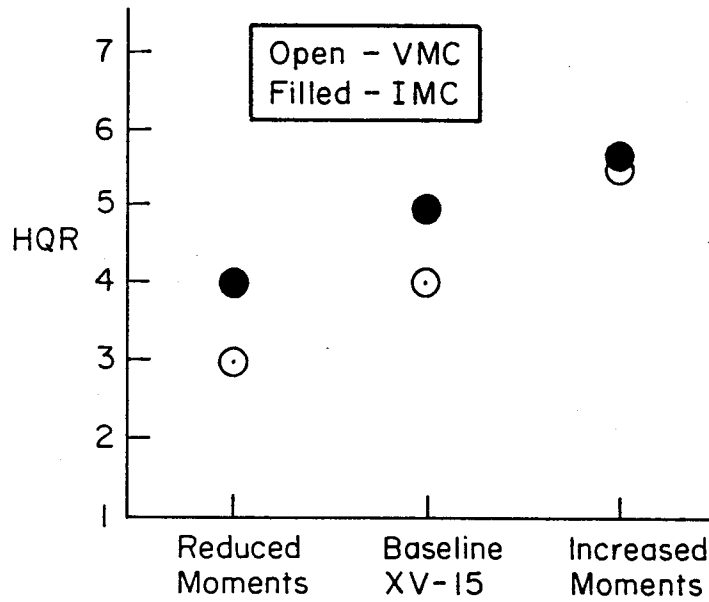
	FLIGHT CONDITION		
	NACELLE ANGLE (DEG)/AIRSPEED (KTS)		
	0/150	60/110	90/60
Increased Coupling	1.41	0.468	2.41
Baseline Coupling	-0.874	0.257	1.77
Decreased Coupling	0.320	0.033	1.10

glide-slope under IFR conditions were exclusively Level 2 regardless of the level of coupling. Pilot ratings improved with the attitude command SCAS. As can be seen in Table 2, the variation of the coupling magnitude with flight condition is large when compared to the vectored thrust cases in Figure 1. The pilot rating results from Reference 193 are not as adverse as might be expected as shown in Figure 2 (note that the task in Figure 2a is similar to that of Figure 1a). Considering that the values in Table 2 are for the unaugmented XV-15, it is likely that the coupling was significantly less for the rate and attitude SCASs evaluated by the pilots.

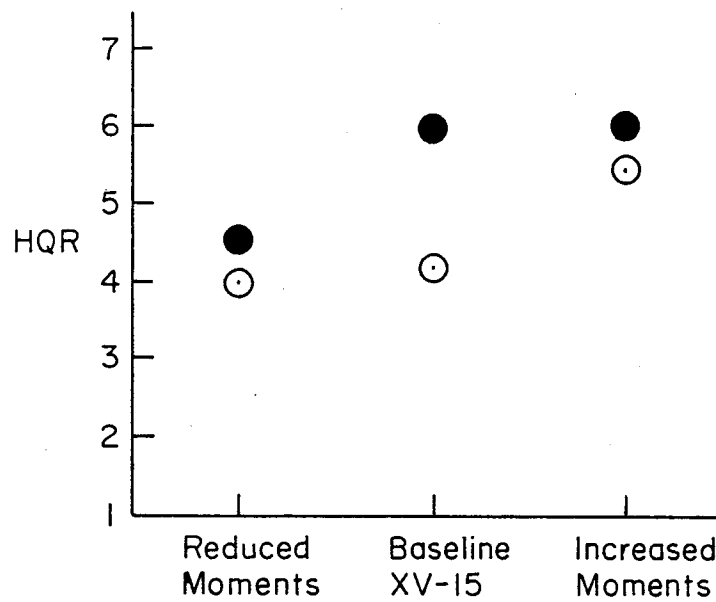
The pilots in the Reference 193 (XV-15) experiment also commented on significant altitude excursions with transition controller variations. The Reference 3 experiment indicated that good pitch attitude dynamics (bandwidth) alleviates this situation by allowing the pilot to effectively maintain altitude through pitch attitude control. Reference 193 supports this conclusion in that the inclusion of an ACAH Response-Type significantly improved the transition with increased coupling. Because of these results, it was decided not to include a requirement on "ballooning," but to insure adequate pitch attitude bandwidth. This is accomplished by invoking the requirements for UCE>1 and IMC in Paragraphs 3.3 and 3.4, respectively, during the transition.

#### d. Guidance for Application

The specification requires the measurement of the peak pitch attitude response to a rapid change of the transition controller with all other controls free. The transition controller input should be as step-like as possible. In most cases the transition controller (rotor mast angle) will move at a predetermined rate significantly slower than the cockpit step inputs. In fact, the magnitude of the coupling can be reduced by decreasing the angular rate of the rotor mast angle. Practically speaking, the requirement is not dependent on the shape of the cockpit controller input, and hence the details of the shape of the input will typically not have an effect on proof of compliance.



*a) Partial Conversion Before Glideslope and Final Conversion on Glideslope (pilot B)*



*b) All Conversion on Glideslope*

Figure 2(3.5.1). Pilot Ratings for Varied Levels of Coupling from Reference 193

e. Related Previous Requirements

There are no requirements on transition in MIL-H-8501A.

The transition requirements in MIL-F-83300 address the topic of transition controller to pitch attitude coupling by limiting the rate of movement of the pitch controller necessary to maintain trim in response to change in forward speed. The current requirement limits the pitch attitude response, and hence, the required pitch controller input necessary to maintain trim. The current requirement is also substantiated by a limited data base which was non-existent at the time of writing of MIL-F-83300.

a. Statement of Recommended Requirement

3.5.2 Allowable Degradations in Pitch Attitude Control Over Limited Speed Ranges During Transition. For Level 1, the pitch attitude control requirements of Section 3.4 shall be met at all speeds in the transition regardless of the rate of deceleration or acceleration.

For Levels 2 and 3, the pitch attitude response may be unstable over limited speed ranges in the transition. In no case shall such instabilities result in a time to double pitch attitude of less than 4.5 seconds for Level 2 and 2.2 seconds for Level 3 following a pulse pitch control input. In addition, the degradation must not exist over a speed range greater than 40 kts, nor shall it exist at speeds below 100 kts.

b. Rationale for Recommended Requirement

The requirement exists largely in recognition of the fact that limited instabilities during transition may be unavoidable in variable configuration rotorcraft due to highly nonlinear flow phenomena. This is especially true for operation with failures in the primary flight control system. It seems reasonable to allow regions of instability during the transition, as long as they are controllable by the pilot. Such a relaxation would greatly simplify a backup flight control system. For example, it might allow the use of equalization in the input path, i.e., without feedbacks for the backup system, thereby drastically reducing the number of required sensors. Pilots would, of course, be trained to avoid steady operation in such unstable regions when operating in the failed state.

c. Supporting Data

Supporting data for this recommended requirement was derived from the experiment described in Reference 3. These data show that the pilots assigned ratings between 5 and 6 for a time to double amplitude ( $T_2$ ) of 4.5 seconds, and between 7 and 8 for a  $T_2$  of 2.2 seconds. The instability occurred between 140 and 100 knots. Since no data exists for other speed ranges, the relaxation is restricted to a 40 knot range above 100 knots. The Reference 3 data indicates that increased coupling between the transition controller and pitch attitude tends to aggravate the instability, but not sufficiently to change flying qualities Levels.

d. Guidance for Application

None.

e. Related Previous Requirements

There are no related previous requirements in MIL-H-8501A or MIL-F-83300.

a. Statement of Recommended Requirement

3.5.3 Acceleration/Deceleration Characteristics. From every possible fixed operating point within the rotor-borne flight envelope, with the rotorcraft trimmed at the operating point, it shall be possible to accelerate rapidly and safely to the wing-borne configuration at approximately constant altitude and also on any other flight path as required by the MTEs of 3.1.1. From trimmed steady, level, unaccelerated flight in the wing-borne configuration it shall be possible to decelerate rapidly at approximately constant altitude and also on any other flight path as required by the MTEs, to all possible operating points within the rotor-borne flight envelope. The time taken for these maneuvers and the altitudes flown shall be those designated by the mission requirements. All controls required to effect a transition shall be easily operated by one pilot. At any time during a transition, it shall be possible for the pilot to quickly and safely stop the transition maneuver and reverse its direction.

b. Rationale for Recommended Requirement

This requirement has been taken directly from requirements in MIL-F-83300.

The intent of this requirement is to ensure that the initiation of a transition maneuver is not irreversible, and can be performed safely and quickly without excessive pilot attention to aircraft attitude, airspeed, trim, or any other factor(s) which would reduce the pilot's ability to control the aircraft about a chosen flight path. This requirement is based on the philosophy that the pilot should be able to operate the transition controller to accelerate and decelerate in a reasonably aggressive manner without excessive workload.

The requirement to accomplish the maneuver at constant altitude is intended to disallow excessive ballooning which cannot be easily compensated for with pitch attitude.

The acceleration/deceleration rate is a function of kinematics, and does not need to be specified. However, a time is specified to insure that other factors (e.g., very slow rotor-mast angular rate) do not cause the transition to limit the ability to accomplish the required MTEs.

c. Supporting Data

This requirement is qualitative in nature, and is not supported by data, other than comments by reviewers that the ultimate goal is to be able to "maneuver the helicopter with abandon."

d. Guidance for Application

None.

e. Related Previous Requirements

This requirement is taken directly from MIL-F-83300. There is no related requirement in MIL-H-8501A.

a. Statement of Recommended Requirement

3.5.4 Control Margin. It shall be possible to achieve the angular rates defined by the Limited Maneuvering Mission-Task-Elements of Table 1(3.3) at airspeeds below 45 knots during the transition. For airspeeds at and above 45 knots, during the transition the requirements of Paragraph 3.4.2 shall apply in the pitch axis, and for roll control, the Limited Maneuvering requirements of Table 3(3.4) shall apply.

b. Rationale for Recommended Requirement

This recommended requirement is necessary to specify which control power requirements of Paragraphs 3.3 and 3.4 apply for the transition. For Hover and Low Speed, the requirements for Limited Maneuvering MTEs are judged to be adequate since it is unlikely that more aggressive MTEs will be accomplished during a transition. The pitch control power requirement for Forward Flight (Paragraph 3.4.2) is based on the ability to achieve the limit load factor defined by the Operational Flight Envelope which seems a reasonable requirement for transition. The requirements for "Limited Maneuvering" in Table 3(3.4) were felt to be adequate for roll control during transitions as they allow reasonable agility, without requiring that the more aggressive MTEs be accomplished during a transition (see Table 3(3.4)).

c. Supporting Data

The supporting data is given in the appropriate paragraphs covering the backup material for Sections 3.3 and 3.4.

d. Guidance for Application

As discussed in the backup material for Paragraph 3.5.1, an airspeed of 45 knots is used as the metric to define where the criteria of Paragraph 3.3 and 3.4 are to be applied during a transition maneuver. This is in contrast to all other MTEs where ground speed is the appropriate metric. The minimum roll rates for Forward Flight are to be taken from Table 3(3.4), and are based on "all MTEs not otherwise specified," e.g., the minimum achievable roll rate for Level 1 is 30 deg/sec.

e. Related Previous Requirement

MIL-F-83300 required at least 50% of the nominal pitch, roll, and yaw control moments available. It was felt that the requirement should be more related to the required tasks (MTEs) than some percent of the available moment. Such a requirement is now possible because of data that was not available during the development of MIL-F-83300.

a. Statement of Requirement

3.6 CONTROLLER CHARACTERISTICS

3.6.1 Conventional Controllers

3.6.1.1 Centering and Breakout Forces. Pitch, roll, and yaw controls shall exhibit positive centering in flight at any normal trim setting. The combined effects of centering, breakout force, stability, and force gradient shall not produce objectionable flight characteristics or permit noticeable departures from trim conditions with controls free. Breakout forces, including friction, preload, etc., refer to the cockpit control force required to start movement of the control surface in flight. The requirements for breakout force are given in Tables 1(3.6) and 2(3.6). Table 1(3.6) applies for Hover and Low Speed whereas the values in Table 2(3.6) apply for Forward Flight (Paragraph 2.6). The change in breakout force with speed shall not be objectionable.

The minimum collective-control breakout force may be measured with adjustable friction set. Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided qualitative agreement between ground measurement and flight observation can be established.

TABLE 1(3.6). ALLOWABLE BREAKOUT FORCES, NEWTONS (POUNDS) --  
HOVER AND LOW SPEED

COCKPIT CONTROL	LEVEL 1	LEVEL 2	LEVEL 3
	min/max	min/max	max
Pitch (Centerstick)	2.2/6.7 (0.5/1.5)	2.2/8.9 (0.5/3.0)	26.7 (6.0)
Roll (Centerstick)	2.2/6.7 (0.5/1.5)	2.2/8.9 (0.5/2.0)	17.8 (4.0)
Yaw (Pedals)	8.9/31.1 (2.0/7.0)	8.9/31.1 (2.0/7.0)	62.3 (14.0)
Collective	4.4/13.3 (1.0/3.0)	4.4/13.3 (1.0/3.0)	26.7 (6.0)



TABLE 2(3.6). ALLOWABLE BREAKOUT FORCES, NEWTONS (POUNDS) --  
FORWARD FLIGHT

CONTROL	SCOUT/ATTACK		UTILITY	
	min	max	min	max
Pitch (Centerstick)	2.2 (0.5)	13.3 (3.0)	2.2 (0.5)	22.2 (5.0)
Roll (Centerstick)	2.2 (0.5)	8.9 (2.0)	2.2 (0.5)	17.8 (4.0)
Yaw (Pedals)	8.9 (2.0)	31.1 (7.0)	8.9 (2.0)	62.3 (14.0)

NOTE: The values in Table 2(3.6) are for Levels 1 and 2. For Level 3 the maximum values may be doubled.

b. Rationale for Requirement

The requirements stated in Section 3.6 of the specification cover the mechanical characteristics of the cockpit controls. Paragraphs 3.6.1.1 through 3.6.1.3 are intended specifically for conventional control arrangements, i.e., lefthand collective, pedals for yaw, and centerstick for pitch and roll. There is simply not enough quantitative data at this time to set similar limits for sidestick control arrangements, so the discussion for Paragraph 3.6.2 is used to summarize some of the information that has been gathered on the dynamics of sidesticks.

Design of cockpit controller mechanical characteristics involves a relatively complex interrelationship among several factors, including breakout, friction, force/deflection gradients, etc. A review of helicopters currently in operational use by the Army (see Supporting Data) suggests that as long as the other factors are in harmony, breakout forces in excess of those stated in Tables 1(3.6) and 2(3.6) are acceptable. The forces specified here were taken from MIL-F-83300, and are intended to assure good controller characteristics regardless of the force/deflection gradient (provided, of course, that the gradient meet the requirements of Paragraph 3.6.1.2).

c. Supporting Data

Since this paragraph covers "breakout forces, including friction," it is of value to consider both the breakout and friction force bands for controllers. The following discussion, excerpted from Reference 30,

provides some background on the influence of friction and preload on breakout and centering.

"Figure 1a illustrates the cockpit control static force-deflection characteristic obtained with a simple linear spring force-feel system... This system has absolute centering since, upon removal of the applied force, the control returns to its initial position.

"The more usual situation in an aircraft control system is to have some residual friction which results in a force-feel characteristic as illustrated in Figure 1b. This control system possesses positive but not absolute centering. If the control is displaced by a force from the initial position to point A and released, the control tends to return, but due to friction stops at B, short of the trim position. If displaced to position C and released, the control will not return but remain deflected at D with zero force. In fact, the control may occupy deflected positions from E to B with zero force. The greater the magnitude of this zero force displacement band relative to the total control deflection, the less positive the centering characteristic. Over this band, the lack of control force provides no cue to the pilot as to the magnitude of the control deflection....

"It is possible, however, to further reduce the trim control displacement band through the introduction of preloaded springs to the force-feel system. The improvement of the centering characteristic with preload is indicated in Figure 1c.... The cockpit control breakout force is taken to be the force which must be overcome to start movement of the cockpit control, starting from a control position midway through the trim displacement band."

A review of flight test reports for Army helicopters was conducted to gather data for comparison with the requirements of this Paragraph. Values of breakout and friction force were read from control force/deflection plots, and any relevant comments concerning these characteristics were noted. Tables 1 and 2 summarize the results. Requirements from both MIL-H-8501A and MIL-F-83300 are included for comparison.

It is clear from a review of Tables 1 and 2 that breakout forces that exceed the limits of this paragraph are frequently considered to be satisfactory. Two factors influence this conclusion, however: 1) in most instances, the other mechanical characteristics (force/deflection gradients, etc.) are in harmony with the higher breakout forces; and 2) increased breakout forces are acceptable for forward-flight operations. The latter conclusion supports the MIL-F-83300 format of specifying breakouts as a function of speed range, as opposed to MIL-H-8501A, which specifies relatively low values for all speeds. The former conclusion suggests that higher breakouts are also acceptable as long as the force/deflection gradient is increased accordingly, but it is felt that the values in Tables 1(3.6) and 2(3.6) assure acceptable characteristics regardless of the details of the force/deflection gradient (as long as that gradient meets the requirements of Paragraph 3.6.1.2).

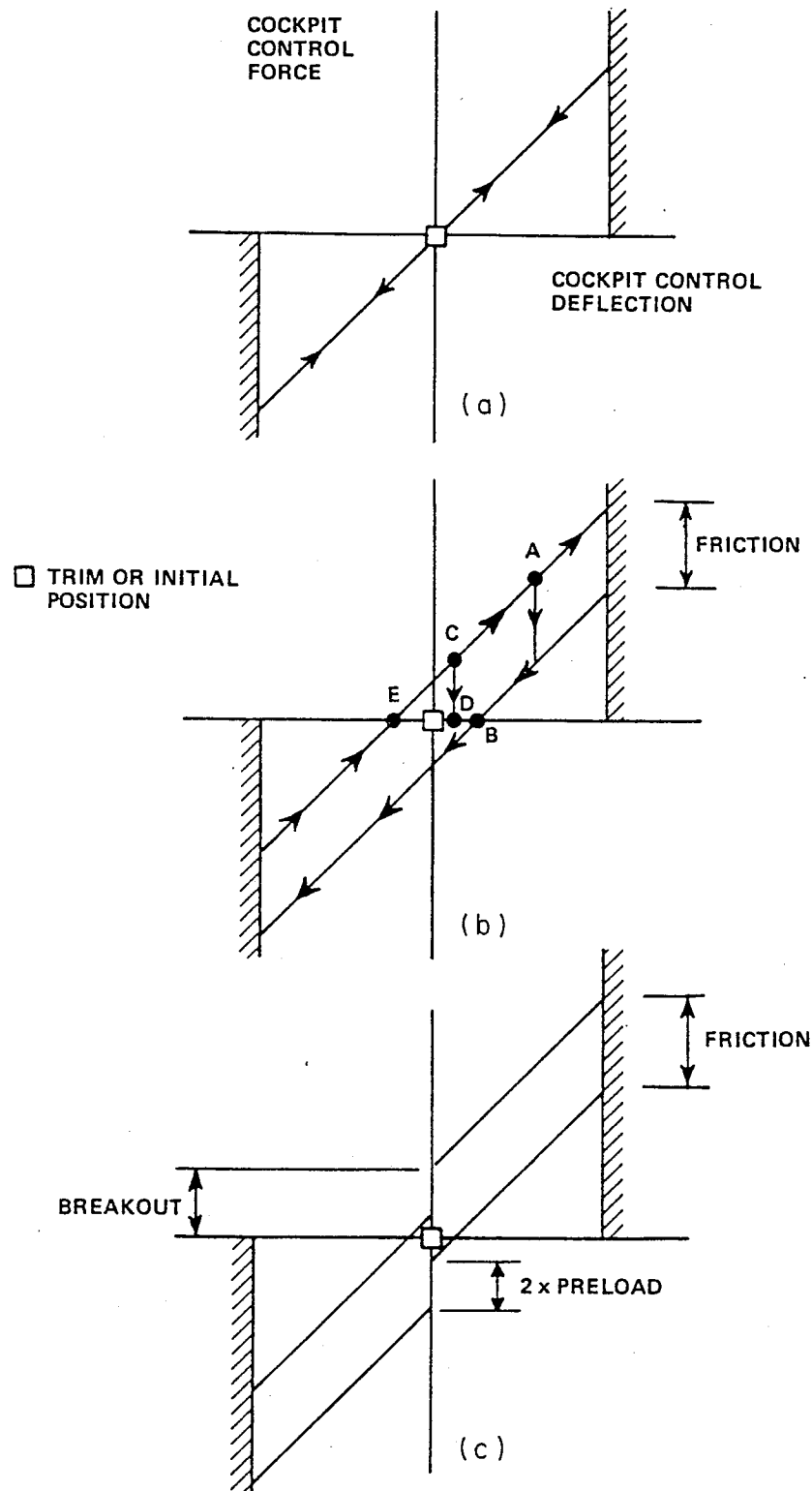


Figure 1(3.6.1.1). Examples of Cockpit Control Force-Deflection Characteristics (From Reference 30)

TABLE 1(3,6,1.1). LONGITUDINAL AND LATERAL CYCLIC BREAKOUT  
CHARACTERISTICS FOR SEVERAL HELICOPTERS

HELICOPTER (REF.)	LONGITUDINAL CYCLIC			LATERAL CYCLIC		
	BREAKOUT FORCE (lb)	FRICTION FORCE (lb)	COMMENTS	BREAKOUT FORCE (lb)	FRICTION FORCE (lb)	COMMENTS
MIL-H-8501A Requirements	0.5 min 1.5 max	N/A	Paragraph 3.2.7	0.5 min 1.5 max	N/A	Paragraph 3.3.13
MIL-P-83300 Level 1: Hover	0.5 min 1.5 max	N/A	Paragraph 3.5.1.1	0.5 min 1.5 max	N/A	Paragraph 3.5.1.1
Fwd Flight	0.5 min 3.0 max	N/A		0.5 min 2.0 max	N/A	
Bell 209 Huey- cobra (67)	1.4 fwd 2.0 aft	0	No shortcomings cited.	1.0 left 1.5 right	1.5 left 0-1 right	"Unique friction band...was manifested in flight by a tendency to roll gradually to the right."
AH-1G Hueycobra (68)	4.5	5-0	High breakout made diving flight at high speeds difficult (HQR-5)	3.5	6.0	High breakout made precise control difficult during hovering flight.
OH-6A (74)	2.6	>14 fwd 3.0 aft	Inadequate self-centering: "Required maximum tolerable pilot effort when establishing and main- taining trim airspeeds...(HQRS-7)."	0	0-0.5 left 0-1.8 right	Lack of breakout requires "considerable pilot effort during turning maneuvers and in turbulent conditions (HQRS-5)...."
AH-56A Cheyenne (70)	1.5 fwd 1.0 aft	2	No shortcomings cited.	1.0	0.9	No shortcomings cited.
S-67 Blackhawk (71)	2.25	4.7	High friction "was objectionable.... Considerable pilot compensation was required during precision hovering tasks (HQRS-5)...[and] to make precise airspeed changes in forward flight (HQRS-5)."	-	-	
Model 309 (72) Kingcobra	0.9 fwd 1.4 aft	1.5	Satisfactory.	1.4	1.5-2	Satisfactory.
OH-58A (73)	0.3 fwd 1.7 aft	2	Objectionable breakout plus friction precluded precise control centering at trim. Inadequate airspeed control during instrument approach (HQRS-6).	1.0 left 0.7 right	2.0 left 1.5 right	Breakout plus friction "masked the spring force gradient around trim....Extensive pilot compensation to maintain a desired roll attitude (HQRS-6)."
YAH-1R Cobra (75)	1.5 fwd 1.0 aft	3.1	Not objectionable in flight.	2.0 left 1.0 right	3.5 left 2.6 right	Not objectionable in flight.
YAH-1S Cobra (76)	3.0	4.0	Satisfactory for attack helicopter mission.	3.5 left 2.0 right	4.0 left 3.5 right	Satisfactory for attack helicopter mission.
YAH-64 (78)	2.4 fwd 2.0 aft	4.5 fwd 5.4 aft	Excessive breakout (plus friction) forces and weak control centering are shortcomings.	1.6 left 1.5 right	1.5	See comments for longitudinal cyclic.
YAH-64 (77)	1.0 fwd 1.6 aft	1.5	Not objectionable.	1.4 left 0.9 right	2.3 left 2.0 right	Not objectionable.
YAH-64 (79)	1.5 fwd 2.5 aft	1.4	"Longitudinal breakout (plus friction)...contributed to increased pilot workload when attempting to maintain a precise hover (HQRS-4)."	1.5 left 1.4 right	2.5	Satisfactory.
CH-47C Chinook (69)	0.6 fwd 1.3 aft	0.7	Satisfactory.	0.7 left 0.6 right	0.5 left 0.4 right	Satisfactory.

TABLE 2(3.6.1.1). PEDAL AND COLLECTIVE BREAKOUT CHARACTERISTICS  
FOR SEVERAL HELICOPTERS

HELICOPTER (REF.)	PEDALS			COLLECTIVE		
	BREAKOUT FORCE (lb)	FRICTION FORCE (lb)	COMMENTS	BREAKOUT FORCE (lb)	FRICTION FORCE (lb)	COMMENTS
MIL-H-8501A Requirements	3.0 min 7.0 max	N/A	Paragraph 3.3.13	1.0 min 3.0 max	N/A	Paragraph 3.4.2
MIL-F-83300 (Hover and Fwd Flight)	2.0 min 7.0 max	N/A	Paragraph 3.5.1.1	1.0 min 3.0 max	N/A	Paragraph 3.5.1.1
AH-1G Hueycobra (68)	5.0	17 left 13 right	Satisfactory.			
AH-56A Cheyenne (70)	8.0 left 7.0 right	0 left 5.5 right	"Large directional breakout forces were...apparent to the pilot and not in harmony with the lateral and longi- tudinal control forces....con- tributed to the considerable pilot effort required to maintain coordinated flight (HQRS-5).			
S-67 Blackhawk (71)				4.2	8-23	Satisfactory.
Model 309 Kingcobra (72)				9.5	14-16	"High breakout force was objectionable, in that excessive pilot effort was required to make small collective control adjustments."
YAH-1R Cobra (75)	1.0 left 3.0 right	5.0	Not objectionable in flight.			
YAH-1S Cobra (76)	3.0 left 4.0 right	5-25	Satisfactory for the attack helicopter mission.	4 up 9 down	8-18	Satisfactory for the attack helicopter mission.
YAH-64 (78)	4.5 left 6.0 right	6 left 6.5 right	No shortcomings cited.			
YAH-64 (77)	7 left 13.5 right	7.6 left 6.4 right	Not objectionable.			
YAH-64 (79)	5.0	7	Satisfactory.			
CH-47C Chinook (69)	14.0 left 14.5 right	0	Satisfactory.	2.0 up 1.5 down	0	Satisfactory.

a. Statement of Requirement

3.6.1.2 Force Gradients. The pitch, roll, and yaw control force gradients shall be within the range specified in Table 3(3.6) throughout the range of control deflections. In addition, the force produced by a 2.54 cm (one inch) travel from trim by the gradient chosen shall not be less than the breakout force. For the remaining control travel, the local gradients shall not change by more than 50 percent in 2.54 cm (one inch) of travel. The thrust magnitude control should preferably have zero force gradient unless an autothrottle function is active.

TABLE 3(3.6). ALLOWABLE CONTROL FORCE GRADIENTS,  
NEWTONS/CENTIMETER (POUNDS/INCH)

CONTROL	LEVEL 1		LEVEL 2	
	min	max	min	max
Pitch	0.9 (0.5)	5.3 (3.0)	0.9 (0.5)	5.3 (5.0)
Roll	0.9 (0.5)	4.4 (2.5)	0.9 (0.5)	8.8 (5.0)
Yaw	8.8 (5.0)	17.5 (10.0)	8.8 (5.0)	35.0 (20.0)

b. Rationale for Requirement

The control force gradients specified in Table 3(3.6) are taken from MIL-F-83300. As shown in the Supporting Data and Related Previous Requirements discussions, these gradients are somewhat larger than the corresponding MIL-H-8501A values, but encompass the force/deflection characteristics found on a large number of helicopters. Compliance with the requirements of this Paragraph, in combination with the breakout limits of Paragraph 3.6.1.1, should assure good control characteristics.

The requirement that the force for the first inch of travel from trim be greater than the breakout force will provide the pilot with a clear indication of breakout, since, ideally, a force input below breakout will result in no control movement. An additional statement guards against excessive nonlinearities in the gradient, although there are no quantitative data to support the requirement of less than a 50 percent change in one inch of travel. Indeed, there are several gradients shown in Supporting Data that change dramatically with no indications of piloting problems. In most of these cases, however, the change occurs only for large control inputs.

d. Guidance for Application

None.

e. Related Previous Requirements

In MIL-H-8501A, breakout forces are specified in Paragraphs 3.2.7 (longitudinal cyclic), 3.3.13 (lateral cyclic and directional), and 3.4.2 (collective), as given in Table 3. These values cover all helicopter types and all speed ranges, independent of mission. The MIL-F-83300 requirements (Paragraph 3.6.1.1) recognize that higher breakout forces are acceptable as speed increases, and these values were chosen for Tables 1(3.6) and 2(3.6).

TABLE 3(3.6.1.1). BREAKOUT FORCE LIMITS FROM MIL-H-8501A

CONTROL	LIMIT CONTROL FORCES FRICTION FORCES	
	MINIMUM	MAXIMUM
Longitudinal cyclic .....	0.5	1.5
Lateral cyclic .....	0.5	1.5
Collective .....	1.0*	3.0
Directional .....	3.0*	7.0

\*May be measured with adjustable friction set

### c. Supporting Data

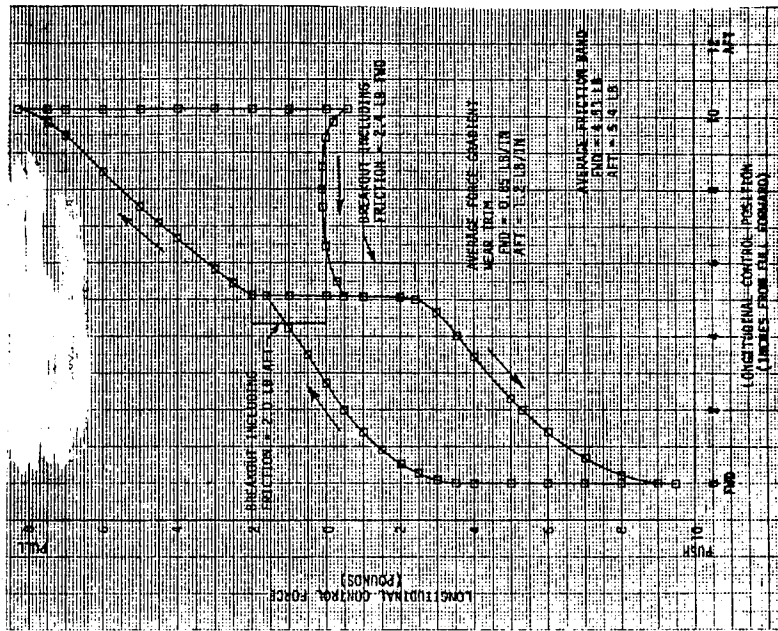
It is difficult to isolate the effects of control force gradient on pilot opinion sufficiently to support the Table 3(3.6) limits. This is because of the major influence of other factors, including breakout, friction, and freeplay. Acceptance of a particular force gradient is a direct function of the level of breakout and friction forces used: the higher either or both of these forces, the higher the desirable gradient.

A review of Army flight test reports was conducted, and the force/deflection characteristics noted. Except for very extreme cases, adverse pilot comments in these reports dealt with the combined effects of breakout, friction, and gradient. As an example of two unsatisfactory force/deflection curves, consider Figure 1. These curves were taken from tests conducted on the ground, but were subjectively considered to be representative of the flight characteristics. Figure 1a shows the curve for the S-67 Blackhawk helicopter (Reference 71). The longitudinal control exhibits high breakout and friction, resulting in a lack of control centering with essentially zero force/deflection gradients. From Reference 71, "Considerable pilot compensation was required during precision hover tasks (HQRS 5)... [and] to make precise airspeed changes in forward flight (HQRS 5)." (HQRS, Handling Qualities Rating Scale, refers to the Cooper-Harper pilot rating scale.) For the YAH-64, Figure 1b, deficiencies were noted in Reference 78 to be "excessive breakout (plus friction) forces, weak control force gradients, and weak control centering of the longitudinal and lateral cyclic control." Breakout and friction are covered in Paragraph 3.6.1.1; the control force gradients fall within the Table 3(3.6) limits -- but as will be seen shortly, the lateral gradient was considerably less than the longitudinal, and may have caused both gradients to seem weak. This suggests that the longitudinal and lateral gradients should be of similar slope.

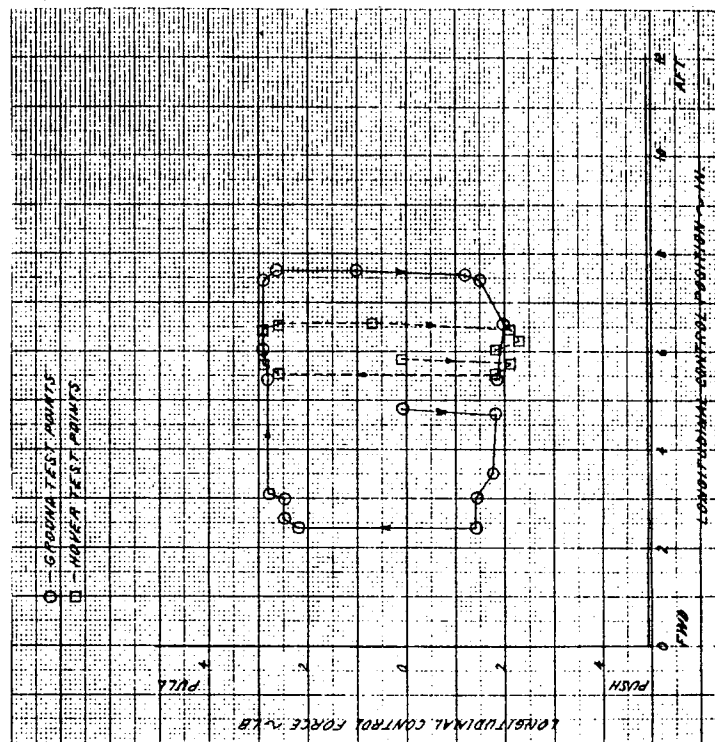
Figure 2 shows the force/deflection curves for a number of military helicopters. Where applicable, the MIL-H-8501A and MIL-F-83300 (Level 1) requirements are also shown (see Related Previous Requirements). For the longitudinal cyclic (Figure 2a), only the S-67 Black hawk (discussed above) falls significantly below either the MIL-H-8501A or MIL-F-83300 limit, while several of the curves exceed the upper limit of MIL-H-8501A, but not MIL-F-83300. The only specific comments pertaining to gradients were the two mentioned above -- the S-67 and the YAH-64. In the latter case, a later assessment (Reference 79) showed a lower gradient (Figure 2a), but it was considered satisfactory. Figure 2b, which shows lateral cyclic curves, suggests that the Reference 78 complaint about "weak" gradients may have resulted from the fact that the lateral gradient is much lower than the longitudinal, while for Reference 79 the two gradients are more in harmony. The lateral gradient for the YAH-64 is below either the MIL-H-8501A or MIL-F-83300 limit for both References 78 and 79, however.

Few of the directional force/deflection curves fall within the MIL-F-83300 Level 1 range (Figure 2c). The only adverse comments were for the AH-56A (Reference 70) where the gradient was considered "excessive." Since this gradient is near the Level 1 upper limit from MIL-F-83300, it



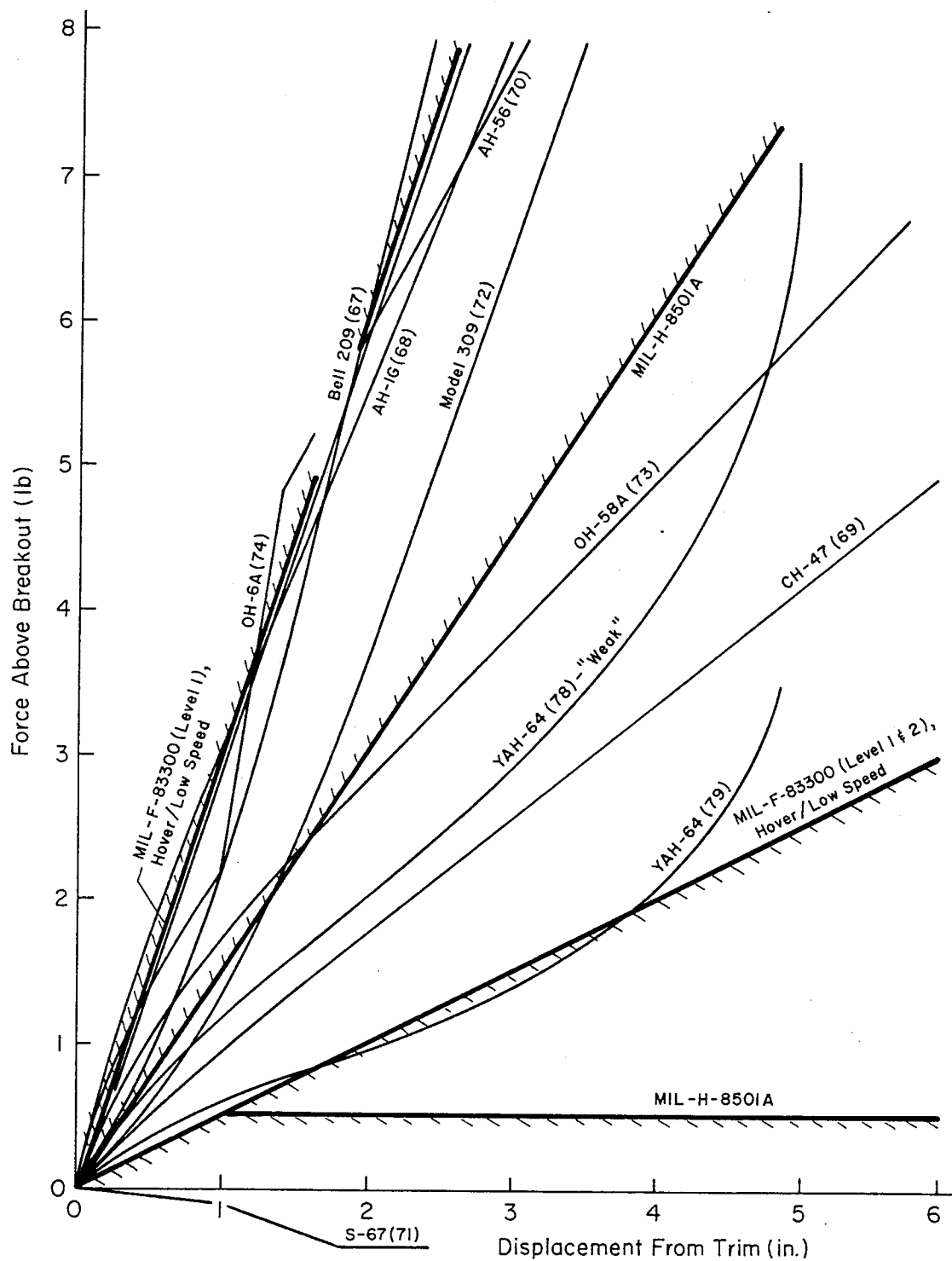


b) YAH-64 (Ref. 78) - Excessive Friction,  
Weak Control Force Gradients



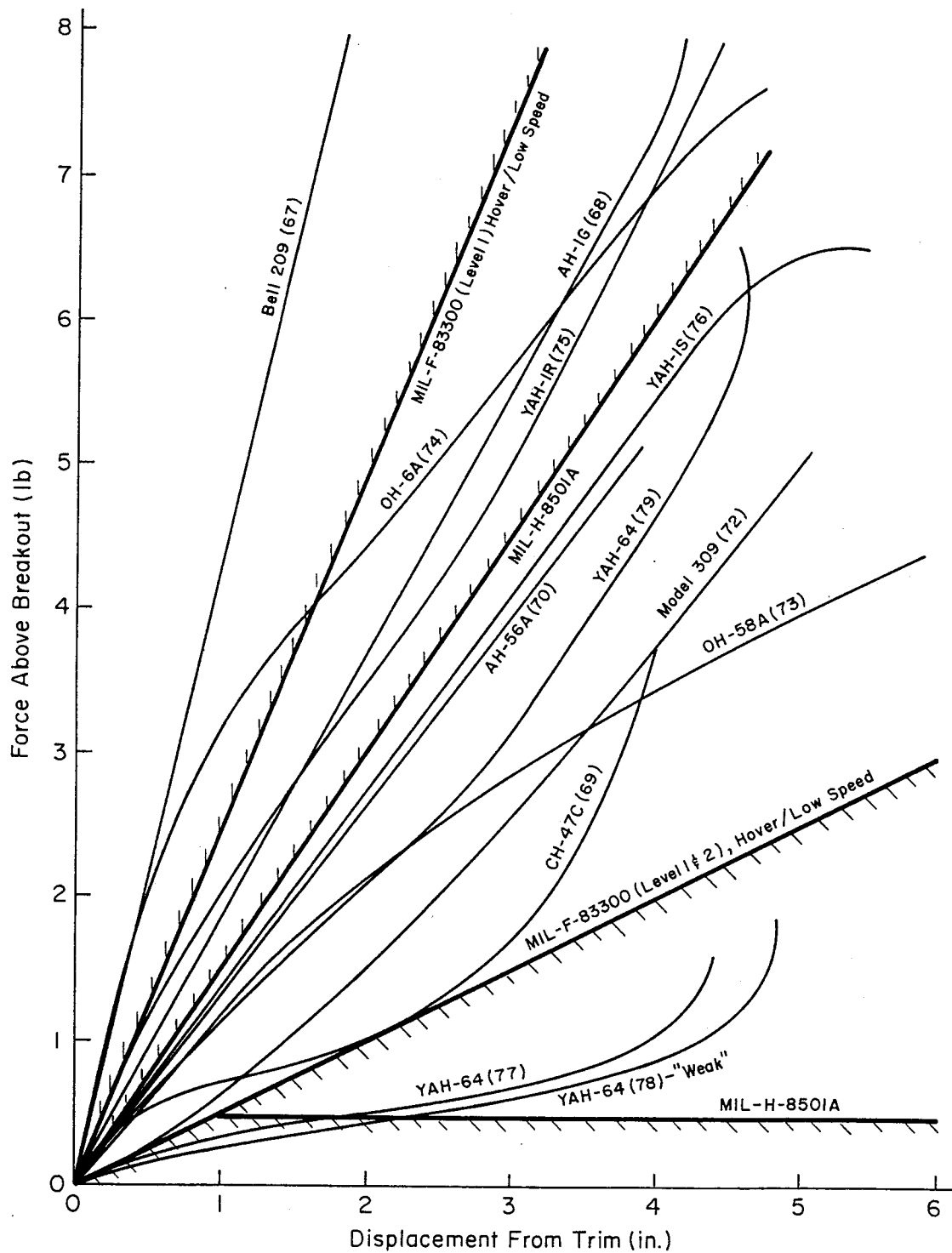
a) S-67 (Ref. 71) - Excessive Breakout,  
No Force Gradient

Figure 1(3.6.1.2). Examples of Unsatisfactory Longitudinal Cyclic Force/Deflection Characteristics



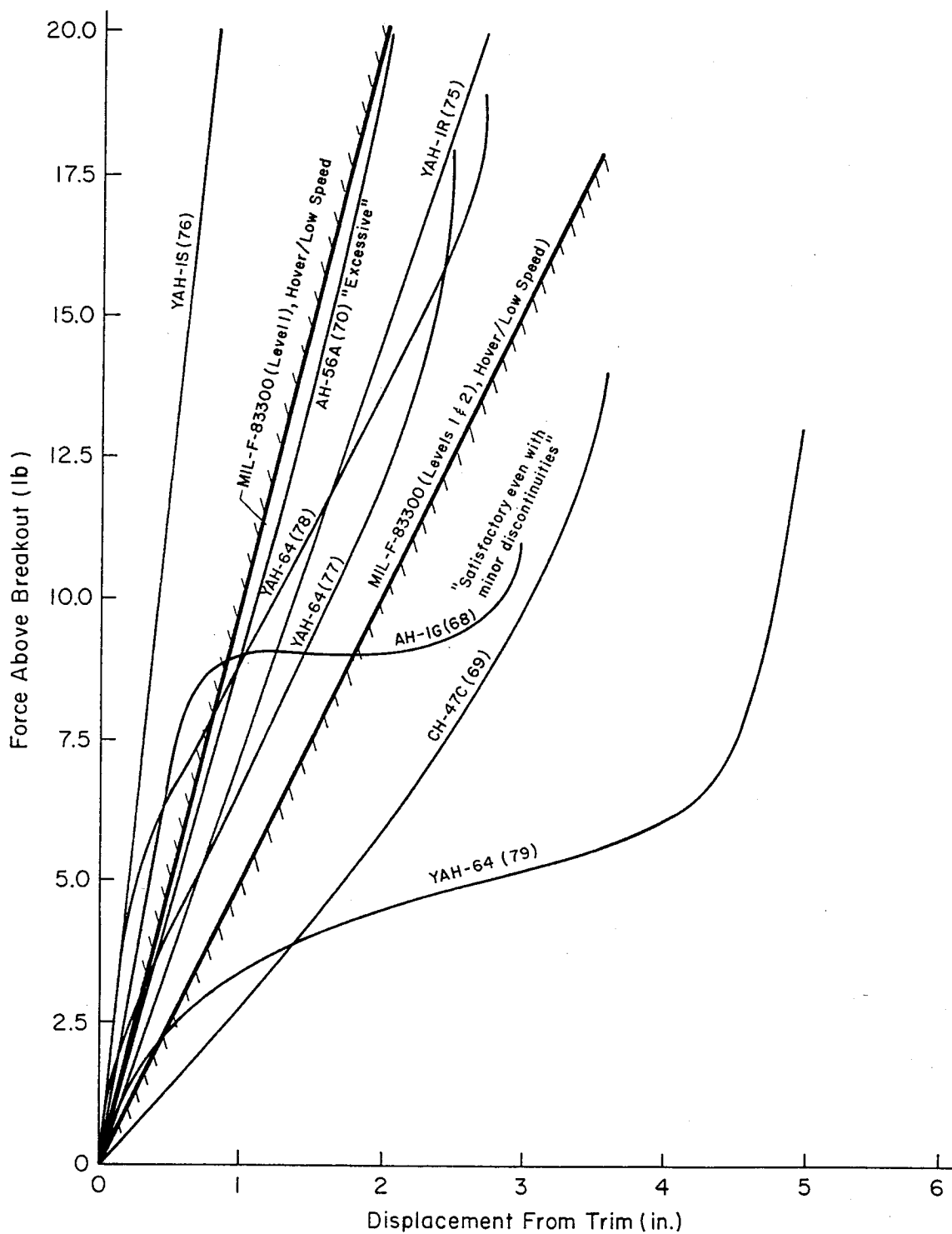
a) Longitudinal Cyclic

Figure 2(3.6.1.2). Control Force/Deflection Curves for Several Representative Helicopters. Numbers in Parentheses are Reference Numbers



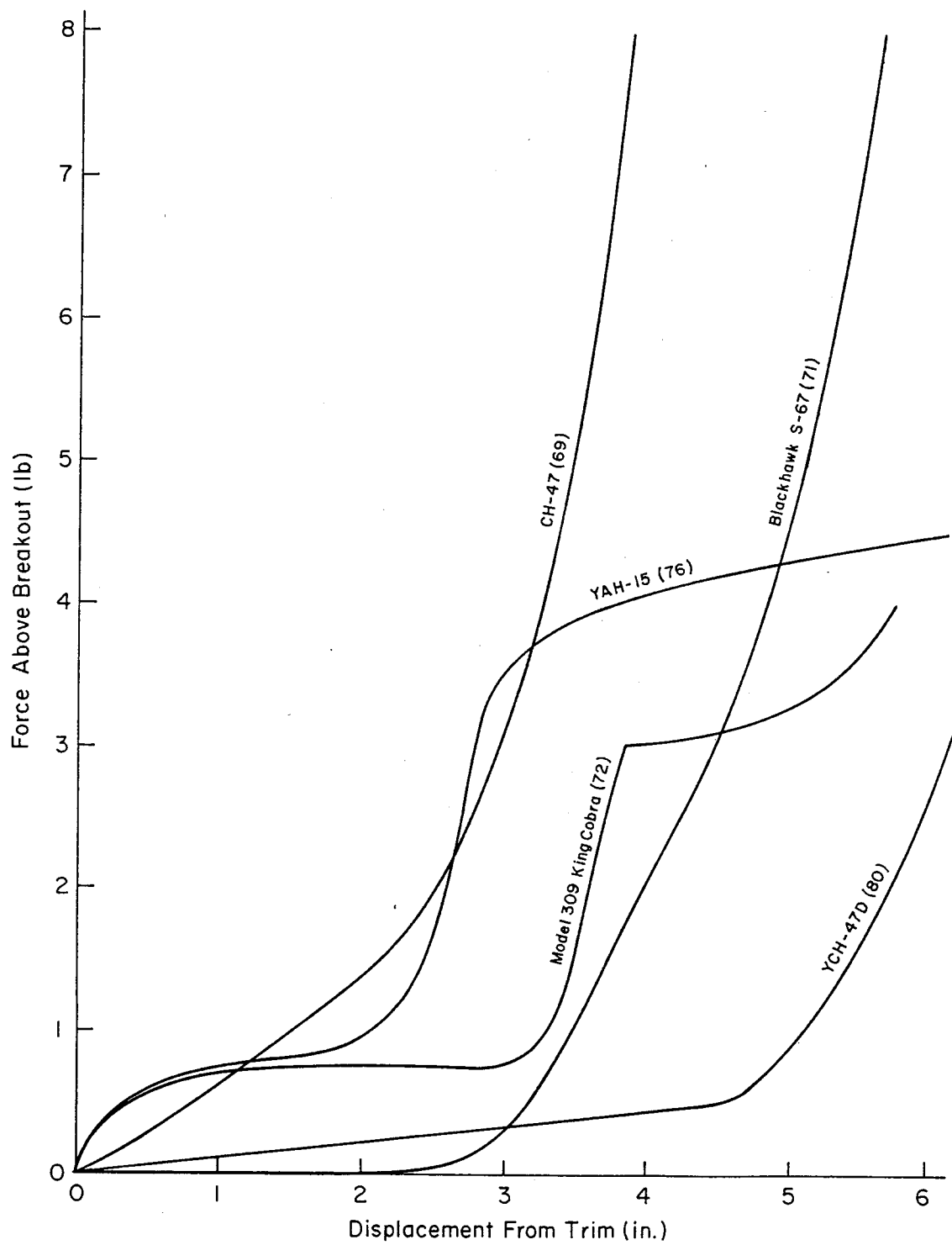
*b) Lateral Cyclic*

Figure 2(3.6.1.2) (Continued)



c) Directional Control

Figure 2(3.6.1.2). (Continued)



d) Collective Control

Figure 2(3.6.1.2). (Concluded)

is felt that this supports the upper value of 10 lb/in. for Level 1 in Table 3(3.6).

There is no limit on collective gradient, except that it should "preferably have zero force gradient," and the limited amount of data available (Figure 2d) does not provide enough information to set a limit. No adverse comments were found concerning collective force gradients.

d. Guidance for Application

None.

e. Related Previous Requirements

The helicopter specification, MIL-H-8501A (Reference 31), states quantitative requirements for longitudinal and lateral cyclic control force/deflection characteristics, with only a general statement for the directional controller, as follows:

- Longitudinal cyclic (MIL-H-8501A Paragraph 3.2.4):  
"The longitudinal force gradient for the first inch of travel from trim shall be no less than 0.5 pound per inch and no more than 2.0 pounds per inch. In addition, however, the force produced for a 1-inch travel from trim by the gradient chosen shall not be less than the breakout force (including friction) exhibited in flight. There shall be no undesirable discontinuities in the force gradient, and the slope of the curve of stick force vs. displacement shall be positive at all times with the slope for the first inch of travel from trim greater than or equal to the slope for the remaining stick travel."
- Lateral cyclic (3.3.11): Identical to longitudinal.
- Directional control (3.3.11): "The directional control shall have a limit force of 15 pounds at maximum deflection with a linear force gradient from trim position. There shall be no undesirable discontinuities in the... directional force gradients."

The requirements adopted for this paragraph -- based on MIL-F-83300 -- are more specific and, as shown in Supporting Data, better encompass the range of gradients that exist in Army helicopters.

a. Statement of Requirement

3.6.1.3 Limit Control Forces. Unless otherwise specified in particular requirements, the maximum control forces required, without retrimming, for any maneuver consistent with the anticipated Mission-Task-Elements (3.2.2), shall not exceed the values stated in Table 4(3.6).

TABLE 4(3.6). LIMIT COCKPIT CONTROL FORCE VALUES, NEWTONS (POUNDS)

COCKPIT CONTROL	HOVER AND LOW SPEED			FORWARD FLIGHT		
	LEVEL 1	LEVEL 2	LEVEL 3	LEVEL 1	LEVEL 2	LEVEL 3
Pitch (Centerstick)	67 (15.0)	69 (20.0)	178 (40.0)	133 (30.0)	156 (35.0)	178 (40.0)
Roll (Centerstick)	45 (10.0)	67 (15.0)	89 (20.0)	67 (15.0)	89 (20.0)	111 (25.0)
Yaw (Pedals)	133 (30.0)	178 (40.0)	356 (80.0)	334 (75.0)	445 (100.0)	556 (125.0)
Collective	45 (10.0)	45 (10.0)	45 (10.0)	45 (10.0)	45 (10.0)	45 (10.0)

b. Rationale for Requirement

The forces specified in this requirement are considered to be the maximum acceptable to assure satisfactory flying qualities and to be compatible with the other control characteristics requirements stated in Section 3.6 of the specification. Generally, limit forces will be set as a function of the combination of breakout (3.6.1.1), force/deflection gradient (3.6.1.2), and control deflection limit, set by cockpit and pilot considerations. They should also reflect comfortable maximum applicable forces that the pilot can sustain for a reasonable time.

c. Supporting Data

Limit control forces are basically set by the combination of breakout force (Paragraph 3.6.1.1) plus control force/deflection gradient (3.6.1.2) over the range of control deflections. Data available for supporting the limits of this requirement come from Army helicopter flight test reports (References 67-80). Limit forces for each controller were determined from curves of force vs. displacement; in most cases, forces were not symmetric (for example, the maximum force for full left cyclic deflection may be somewhat lower or higher than that for full right deflection).

Figure 1 summarizes the limit control forces for longitudinal and lateral cyclic (Figures 1a and 1b), pedals (1c), and collective lever (1d). They have been plotted against total control travel for reference. Also shown for comparison are the MIL-H-8501A and MIL-F-83300 (Level 1) limits. While most of the helicopters exceeded one or more of the force limits, there were no published complaints concerning the high levels. This is probably due in large part to the fact that the limit forces were rarely encountered in flight.

The data of Figure 1 suggest that the limit forces of both MIL-H-8501A and MIL-F-83300 could be increased somewhat. However, the requirements of Table 4(3.6) -- which, with only minor adjustments, are identical to the MIL-F-83300 requirements -- are felt to be a more reasonable protection against pilot fatigue.

Based on Figure 1a, the Level 1 pitch control force limit of MIL-F-83300 was increased from 10 lb to 15 lb; similarly, the data of Figure 1b support an increase from the MIL-F-83300 roll control force limit of 7 lb to 10 lb; and the collective lever forces of Figure 1d support an increase in the collective limit from the MIL-F-83300 value of 7 lb to 10 lb. The yaw requirement of MIL-F-83300, 30 lb for Level 1 at low speeds and 75 in forward flight, appears to be well-supported by Figure 1c.

#### d. Guidance for Application

This requirement, in combination with Paragraphs 3.6.1.1 and 3.6.1.2, defines the control force/deflection characteristics for the cockpit controllers. The user should bear in mind that the limit force requirements of Table 4(3.6) place limits on either the maximum control displacement, the maximum force gradient, or the shape of the force/deflection curve that can be used. For example, from Figure 1a we can see that a total longitudinal cyclic displacement of about 10 inches (5 inches forward and aft) is common; if this displacement were used in combination with the maximum breakout force (1.5 lb) and force gradient (3 lb/in.) allowed for Level 1, a limit force of 16.5 lb would result ( $3 \times 5 + 1.5 = 16.5$ ) -- 1.5 lb above the Level 1 limit of Table 4(3.6). Thus, one or more of the design variables must be reduced to comply with this paragraph.

#### e. Related Previous Requirements

Table II of MIL-H-8501A specifies the following force limits:

- longitudinal cyclic.....8.0 lb
- lateral cyclic.....7.0 lb
- collective.....7.0 lb
- directional.....15.0 lb



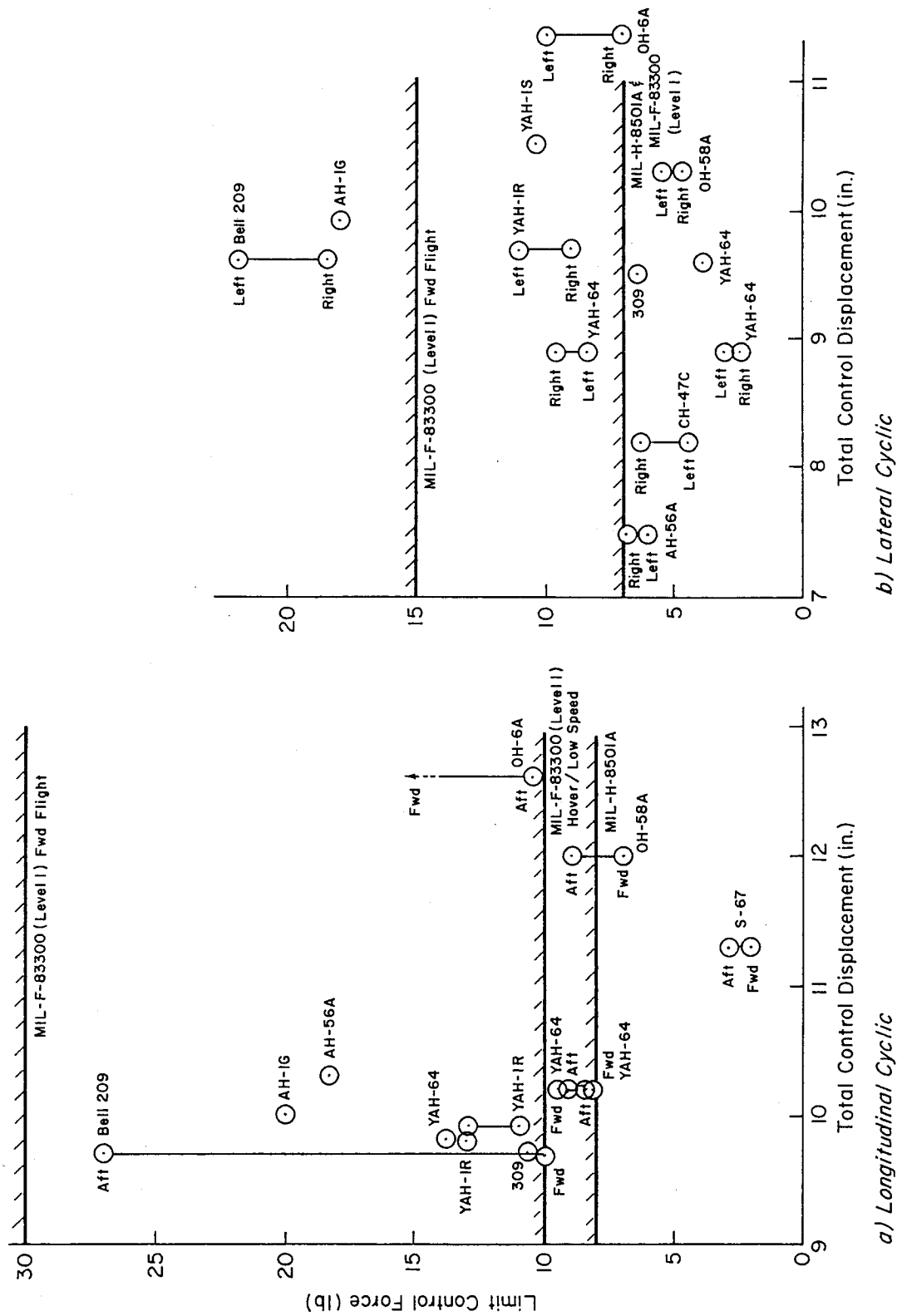


Figure 1(3.6.1.3). Control Force Limits for Representative Helicopters

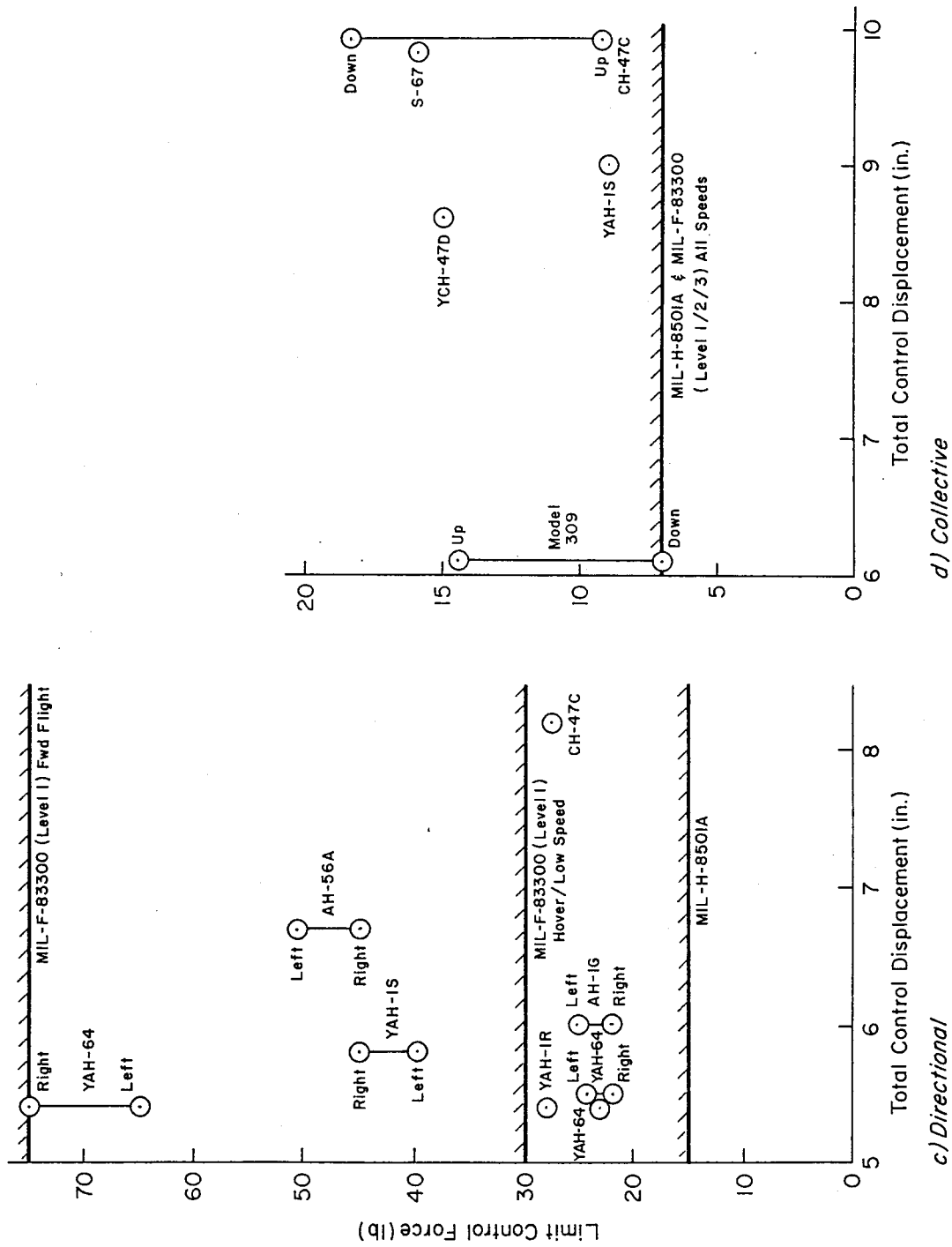


Figure 1(3.6.1.3). (Concluded)

These forces are to be applied over all speeds and all conditions. The Supporting Data discussed above suggest that all of these limits are lower than necessary; the MIL-F-83300 limits (3.6.3 of Reference 29) separate forces by speed range and by flying qualities Level, and hence are more realistic. The forces given in Table 4(3.6) are identical to the MIL-F-83300 limits with the exceptions stated in the Supporting Data discussion.

a. Statement of Requirement

3.6.2 Sidestick Controllers. This paragraph is reserved for future requirements.

b. Discussion

The requirements stated in 3.6.1.1 through 3.6.1.3 cover the mechanical characteristics of conventional pitch, roll, yaw, and vertical controllers. These requirements are relatively well-supported by available data, but there is little quantitative information for unconventional controllers (2-, 3-, or 4-axis sidesticks). This discussion reviews the results of recent experiments comparing sidestick controllers with conventional centerstick configurations. A more detailed discussion can be found in Reference 60.

The considerations involved in using sidestick controllers for two, three, or all four axes are primarily concerned with cockpit design, since reductions in weight and space required can be significant. From the standpoint of flying qualities, the considerations can be divided into two major areas: conventional vs. sidestick controllers for pitch and roll; and integrated yaw or heave vs. separate pedals or collective.

CONVENTIONAL VS. SIDESTICK CONTROLLERS

This is an area that is especially lacking in quantitative data. For a proper assessment of the effects of sidestick controllers, it is necessary to perform a test program in which the helicopter and the evaluation tasks are identical, with only the controller configurations changed. References 61, 62, and 157 report on tests of this type.

Reference 61 details the results of two piloted simulation experiments conducted at NASA Ames Research Center. While some reservations have been expressed about the use of simulation data for handling qualities studies, the interest here will be in relative evaluations -- i.e., we are concerned with the change in pilot ratings due to controller type, rather than the absolute pilot ratings.

The first simulation reported in Reference 61 was performed on the Flight Simulator for Advanced Aircraft (FSAA), with conventional pedals and collective. This simulation compared a conventional centerstick controller with a 2-axis displacement-type sidestick. The evaluation task consisted of a composite terrain flight mission. Pilot rating results from Reference 61 are shown in Figure 1. For the centerstick controller, pilot ratings improve by about one point as control system sophistication increases. Similar ratings were achieved for the first sidestick configuration (SSC 1), which had high breakout forces and high force gradients. For SSC 2, breakout forces were decreased and force gradients reduced, resulting in Level 1 average pilot ratings for all three SCAS types. This results in a noticeable improvement over the centerstick ratings, as Figure 1 shows.

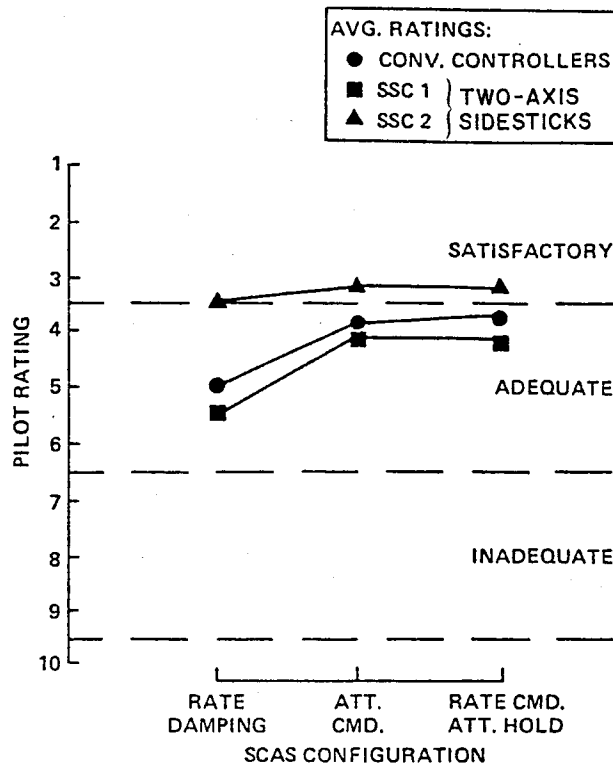


Figure 1(3.6.2). Experiment 1: Conventional vs. 2 Axis Sidestick for Composite Terrain Flight (HUD on) (From Reference 61).

For the second experiment of Reference 61, performed on the Vertical Motion Simulator (VMS), multi-axis sidestick controllers were evaluated: a 3-axis stick with twist for yaw (separate collective), and a 4-axis stick. Results of a portion of this experiment are shown in Figure 2. These data suggest that there is an interchange between control system sophistication and controller configuration, i.e., there is a large spread in pilot ratings for the rate damping system, while the ratings essentially coalesce for the Rate Command/Attitude Hold system. Reference 61 reports that pilot complaints with the sidestick and rate damping configurations dealt with "inability to perform coordinated turns..., low pitch-control sensitivity for maneuvering, collective-to-pitch coupling, and lack of vertical augmentation." In addition, "the 4-axis controller induces both cross-coupling between the longitudinal and vertical axes and collective control difficulties as a result of the vertical-acceleration command system implemented." This observation is consistent with the results of the Army's Tactical Aircraft Guidance System (TAGS) program (Reference 63), where a 4-axis controller "caused anthropometric command coupling and 4-axis maneuver control was too complex and confusing."

Based on the data of Figure 2, it is postulated in Reference 61 that multi-axis sidesticks require a more sophisticated pitch/roll Response-Type than conventional controllers or 2-axis sidesticks. It is important to note, however, that the RCAH system, which included input decoupling and vertical augmentation, should have lower control

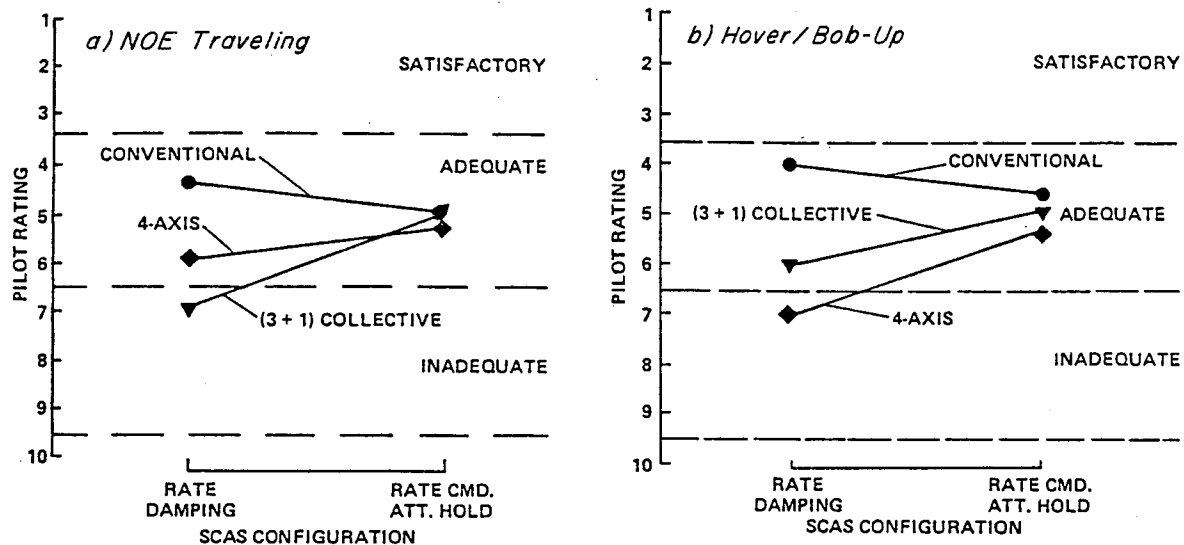
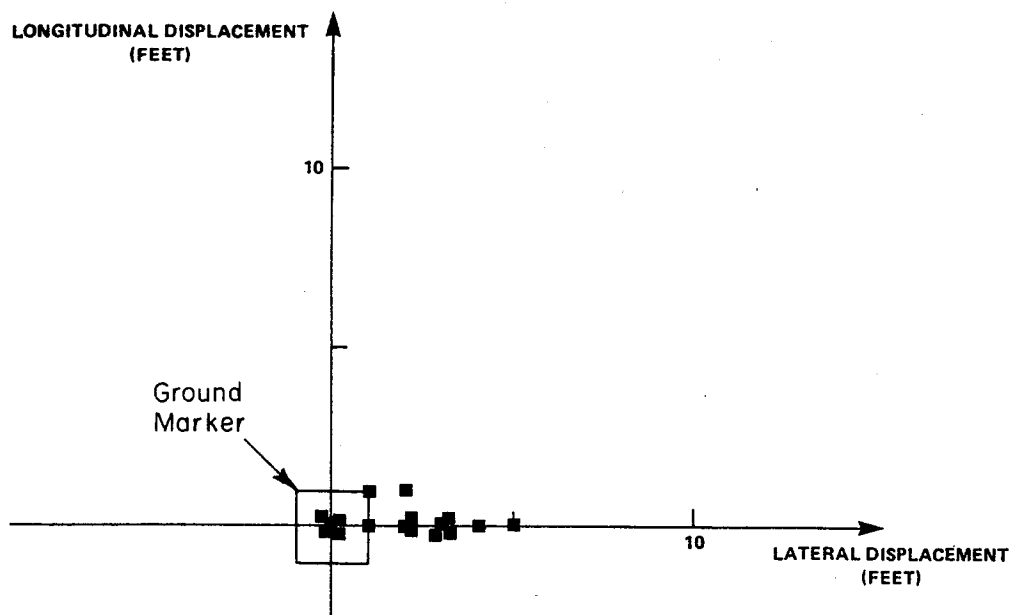


Figure 2(3.6.2). Conventional vs. Multi-Axis Sidestick Controllers (Reference 61).

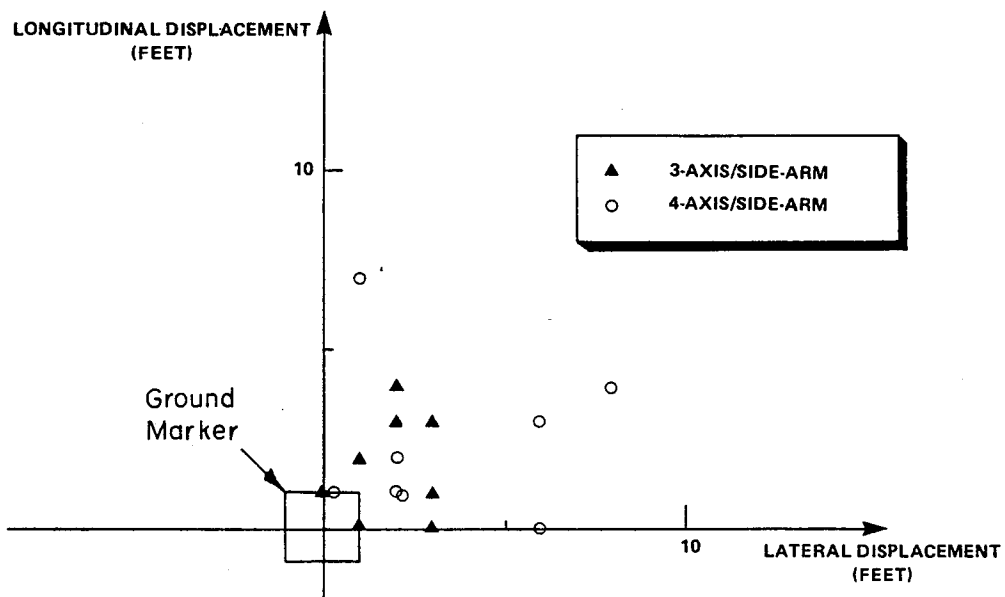
activity; but the simpler rate damping system was preferred for the centerstick controller. The multi-axis sidestick used was a rigid force-type stick, and the reduced control activity may account for the improved rating for the RCAH system. Numerous studies (e.g., References 14 and 64) show that displacement sidesticks are preferable to rigid sticks and this also affects the relative results shown in Figures 1 and 2.

Further information can be obtained from the flight tests reported in Reference 62. These tests, conducted by the NRC of Canada using the variable-stability NAE Bell 205A, compared a conventional control configuration with 3- and 4-axis isometric (rigid) sidestick controllers. Performance and subjective pilot opinion data were obtained (Cooper-Harper pilot ratings are not given in Reference 62). As an example of the former, Figure 3 shows touchdown precision for vertical landings for the conventional controls (Figure 3a) and sidestick configurations (Figure 3b). Dispersions are significantly higher for the sidestick controls. Figure 4 shows subjective comparisons between the sidestick and conventional controls of task difficulty and control precision. The shaded regions are for two pilots with previous experience with isometric sidestick controllers; these pilots found the controllers to be quite similar to or better than conventional controls. The evaluations from the novice pilots ranged from similar to worse. This reflects a natural learning process that is consistent with piloting experiences with the sidestick in the F-16.

The flight experiments of Reference 157 utilized both conventional controls and an integrated four-axis sidestick for evaluations of several Response-Types and tasks. The detailed results are documented

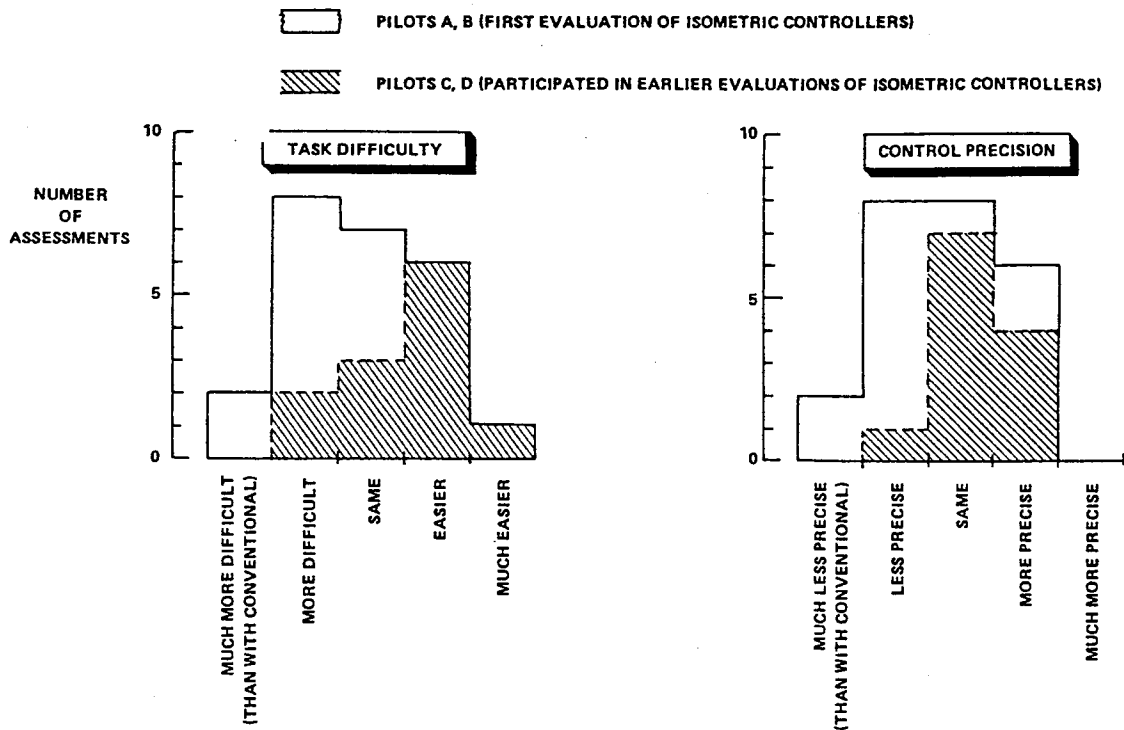


*a) Conventional Controls*

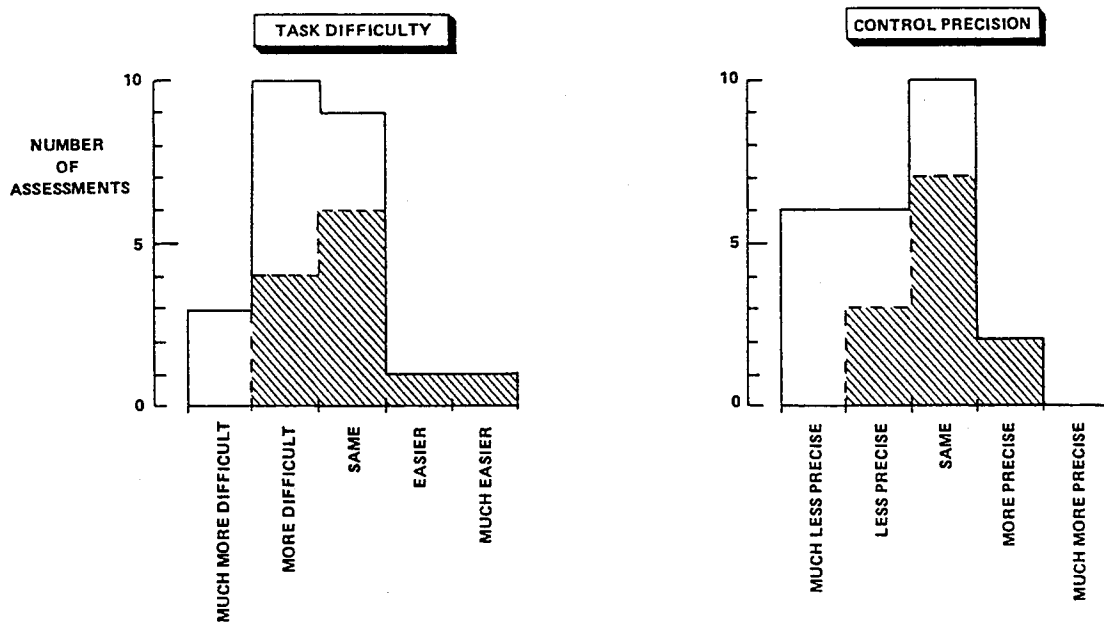


*b) Side-Arm Controllers*

Figure 3(3.6.2). Touchdown Precision for NAE 205A  
(From Reference 62)



a) 3-Axis Isometric Controller



b) 4-Axis Isometric Controller

Figure 4(3.6.2). Subjective Comparisons of Sidestick Controllers with Conventional Controls  
(From Reference 162)



in the Supporting Data for Paragraph 3.3.2.1; the following will compare the conventional vs. sidestick control evaluations. Four test pilots flew both the conventional and sidestick control arrangements.

Figure 5 shows a crossplot of Cooper-Harper pilot ratings from the conventional and sidestick evaluations. In a few instances, a repeat run was flown for one of the control configurations; these are included as separate data points in Figure 5. Pilot ratings that lie to the left of the diagonal line are those cases where the centerstick ratings were better than the sidestick ratings (i.e., centerstick was favored), and points to the right are cases where the sidestick was favored. Points along the diagonal line of agreement represent those cases where the ratings were equal for both controller types.

A linear regression fit (Reference 160) was performed for all of the data of Figure 5 to the standard form,

$$y = mx + b$$

where      $y$  = Cooper-Harper pilot rating for sidestick evaluation,  
           $x$  = Cooper-Harper pilot rating for conventional-control  
                    evaluation,  
           $m$  = slope, and  
           $b$  = y-axis intercept

The dashed lines in Figure 5 show the linear regression fits for each set of data. For almost every condition, the pilot ratings are correlated to greater than a 95% level of confidence (Reference 160). The only exceptions are the ratings for ACAH in the sidestep task (where scatter is large), and the ratings for the Rate Response-Types for the hover, landing, and quickstop tasks (where the linear regression fit is almost horizontal, and these ratings are uncorrelated at greater than a 95% level of confidence).

The linear regression lines provide considerable insight into the characteristics of the data plotted in Figure 5. For example, for the ACAH Response-Types, the lines are shifted to the left of the centerline for all tasks, indicating that the conventional controls were preferred for ACAH.

The slopes of the regression lines in Figure 5 indicate that the pilots were less sensitive to changes in dynamics with the sidestick than with conventional controls. This can be seen by observing that the slopes of the lines are all less than unity; a unity slope would reflect an equal incremental degradation in handling qualities for both controller arrangements, while a slope greater than unity would suggest that the pilots were more sensitive to changes in handling qualities with the sidestick than with the conventional controls.

The pilot rating comparisons in Figure 5 also suggest that the pilots preferred the conventional controls when handling qualities were good, but favored the sidestick as handling qualities degraded. If the conventional controls were uniformly better, most of the ratings would lie above the centerline, as they do for the ACAH Response-Types. For

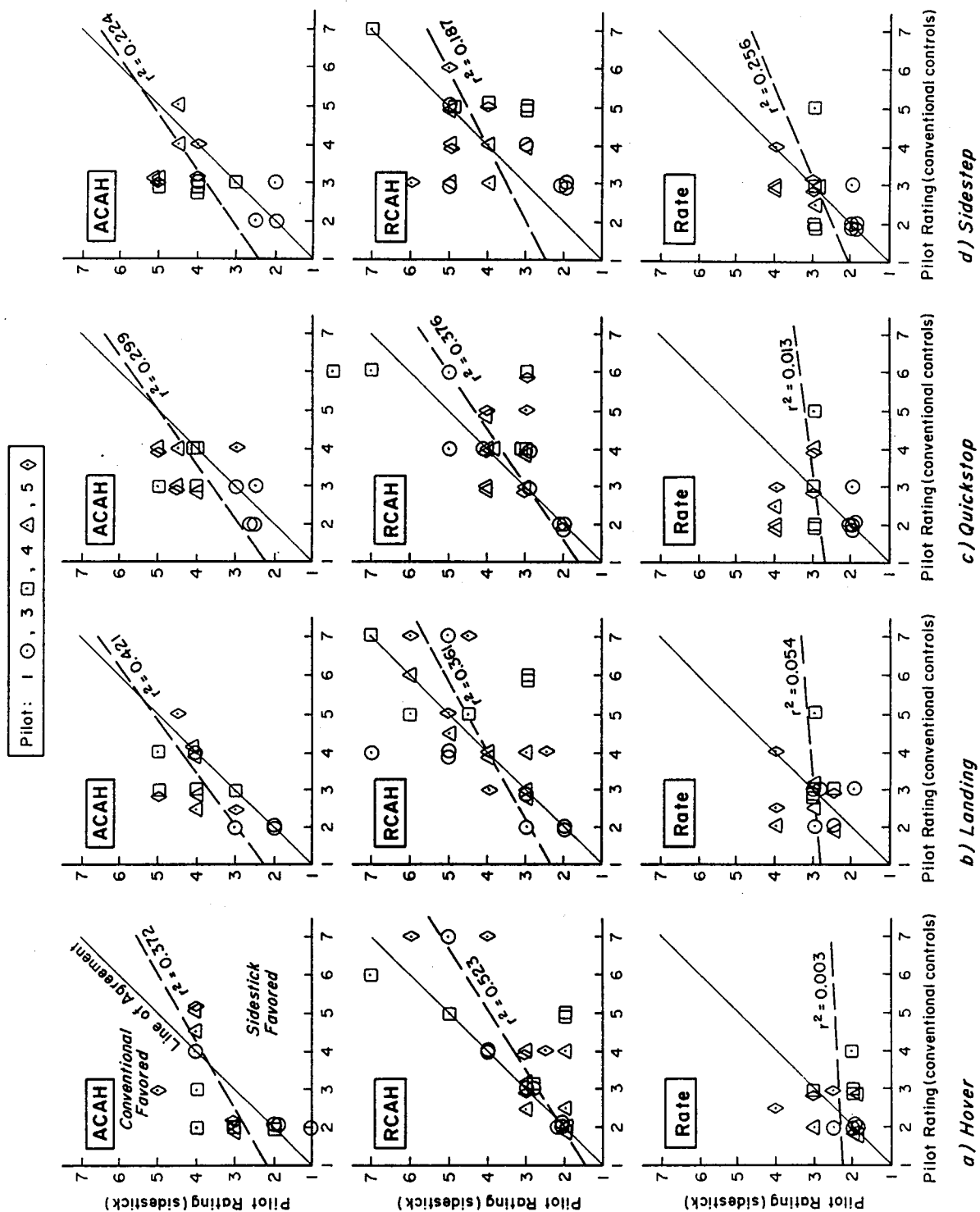


Figure 5(3.6.2). Crossplot of Cooper-Harper Pilot Ratings from Conventional-Control and 4-Axis Sidestick Evaluations. Dashed Lines are Linear Regression Fits to Data for NAE Flight Experiments (Reference 157)

the RCAH and Rate Response-Types, this is not true; for RCAH, especially, the majority of the ratings were better for the sidestick than for the conventional controls. This suggests that, as handling qualities degrade, a four-axis sidestick is preferred over conventional controls.

The foregoing examples reflect one very important aspect of sidestick manipulators -- the interdependence of flight task, command augmentation configuration, and manipulator detail design. For various mission task/control augmentation combinations it may be appropriate for the manipulator neutral position to command or represent trim surface position, zero attitude rate, level attitude, constant velocity, or constant position. Correspondingly, manipulator displacement (or applied force) may command vehicle angular acceleration, rate, or position, linear velocity, etc.

It must be kept in mind that the operator's wrist has quite limited motion and force capabilities when compared to combined arm/wrist action as employed with conventional centersticks. Thus it may be required to tailor sidestick force breakout, displacement/force gradient, neutral position orientation, deflection axis orientation, deflection maximums, and arm/wrist support to the mission task/effective vehicle dynamics or vice versa. Harmony of manipulator motion/force characteristics between various input axes can become particularly crucial for multiaxis manipulators, as can similar harmony between the forces required to operate various grip switches and buttons and those required to overcome flight command thresholds.

For specific background, flight experiences, and design guides in the above areas see References 14 and 134 through 139.

## TWO-AXIS SIDESTICK VERSUS INTEGRATED CONTROLS

The most significant gains in implementation of sidestick controls comes when either the vertical or yaw controls, or both, are integrated into the sidestick controllers. The effects of sidestick controller configuration were investigated extensively during the ADOCS simulations of Reference 14. Two simulation phases were conducted, the first on Boeing Vertol's Flight Simulation Facility, and the second on the NASA Ames Vertical Motion Simulator (VMS). Six pilots participated in the evaluations.

For the Phase 1 simulation, all of the integrated sidestick configurations were difficult to fly. It appears that this was due in part to the particular characteristics of the sticks used, not just because of the addition of control axes to the sticks. All of the sticks evaluated in Phase 1 were relatively stiff (i.e., little or no displacement, especially in the vertical and directional axes). The pilots had difficulty sorting out the responses obtained, and cross-axis inputs were common. Since this is a problem related to the design of the controller, and not to the level of integrated control, the Phase 1 results are not valid for evaluating the tradeoffs between control integration and flying qualities.

For the Phase 2 simulation at NASA Ames, two of the controllers evaluated had deflection in all axes, and these were preferred by the pilots. The following levels of control integration were evaluated:

- (2+1+1) -- Righthand sidestick cyclic, lefthand sidestick collective, and pedals;
- (3+1)C -- Yaw control with cyclic twist, separate collective;
- (4+0) -- Collective and yaw control on cyclic.

Six pitch and roll Response-Types were evaluated for these controller configurations:

- Angular Acceleration Command/Angular Rate Hold, ACCRH (AC/RA in Reference 14);
- Rate Command/Attitude Hold, RCAH (RA/AT);
- Attitude Command/Attitude Hold, ACAH (AT/AT);
- Attitude Command/Groundspeed Hold, ACVH (AT/LV);
- Translational Rate Command/Groundspeed Hold, TRCVH (LV/LV);
- Translational Rate Command/Position Hold, TRCPH (LV/PH).

For the forward-flight tasks, a "hybrid" system was also evaluated. This system consists of ACVH in pitch and roll at low speeds, with the roll Response-Type blending to RCAH above 45 kts.

In Reference 14, the results of the Phase 2 controller studies are analyzed by comparing average pilot ratings for four pilots as functions of Response-Type and task. Based on this analysis, the separated control configuration (2+1+1) was preferred overall. The three-axis sidestick with a separate lefthand collective, (3+1)C, was second in preference, while the four-axis stick, (4+0), resulted in a handling qualities degradation for almost every task and Response-Type. For this discussion, however, we are not as concerned with the effect of Response-Type, since that issue is covered in the discussion for Paragraph 3.2.2. Instead, the interest here is in a one-to-one comparison of separated versus integrated controls.

Figure 6 shows the pilot rating comparisons for the separated-controller, (2+1+1), and integrated four-axis controller, (4+0), evaluations. The ratings are plotted in a format similar to that used for Figure 5: different symbol shapes represent ratings for the four pilots, while the shading or flagging of the symbols represent the different Response-Types. The ratings shown are for both VMC (Figures 6a and 6b) and IMC (Figures 6c and 6d) operations, the latter using a helmet-mounted display system. In this type of plot, the specifics of

the Response-Type are not critical -- e.g., the Acceleration Command/Rate Hold (ACCRH) Response-Type is not an acceptable Response-Type in Table 1(3.2), and it is a clearly degraded system, but for Figure 6 the interest is in comparing the pilots' evaluations with each control arrangement, not in the details of the Response-Type.

The dashed lines on Figure 6 are linear regression fits to each set of data. The regression coefficient,  $r^2$ , is noted beside each regression line. As for Figure 5, pilot ratings above the diagonal line of agreement are those for which the pilot gave the separated-control configuration a better rating, and vice versa. Unlike Figure 5, each symbol on Figure 6 represents the average rating that the pilot gave each configuration. In most instances multiple evaluations are included.

The results of Figure 6 suggest that there is a clear preference for separated controls when handling qualities are good, and for the integrated controller when handling qualities degrade. (Note in Figure 6 that this is equivalent to saying that the pilots generally preferred the separated controls for the highest Response-Types, e.g., ACVH, but preferred the integrated control for the most degraded Response-Types, e.g., ACCRH.) This observation is very similar to that of Figure 5, where the comparison was between separated controls with conventional cyclic and collective, and a integrated four-axis sidestick.

Comparison of the separated-control and three-axis-plus-collective, (3+1)C, configurations shows similar trends (Figure 7).

The foregoing serves to illustrate that there is a definite preference for a separation of controls for Level 1 operation, at least for the Response-Types and tasks evaluated. It is important to consider that the tasks evaluated in the Reference 14 simulations, as well as the Reference 157 flight experiments, did not emphasize the areas of operation where integrated controls are expected to be most beneficial. For example, in a period of single-pilot, divided-attention operation, without benefit of extensive holding features such as position hold, it is almost certain that an integrated controller would be favored.

The rest of this discussion will focus on the details of design for sidestick controllers.

## BREAKOUT FORCES

There is evidence to indicate that the pitch and roll breakout forces for sidesticks should be somewhat less than the centerstick forces specified in Tables 1(3.6) and 2(3.6). For example, Reference 64 recommends a force between 1/2 and 1 lb for sidesticks in conventional airplanes. The ADOCS simulations of Reference 14 used even lower breakouts: 0.5 lb longitudinally and 0.25 lb laterally. There is no indication in Reference 14 that any of the pilots considered these forces to be too low. Without quantitative evidence, it would appear reasonable to design sidestick pitch and roll breakout forces to be near the lower limits of Tables 1(3.6) and 2(3.6).

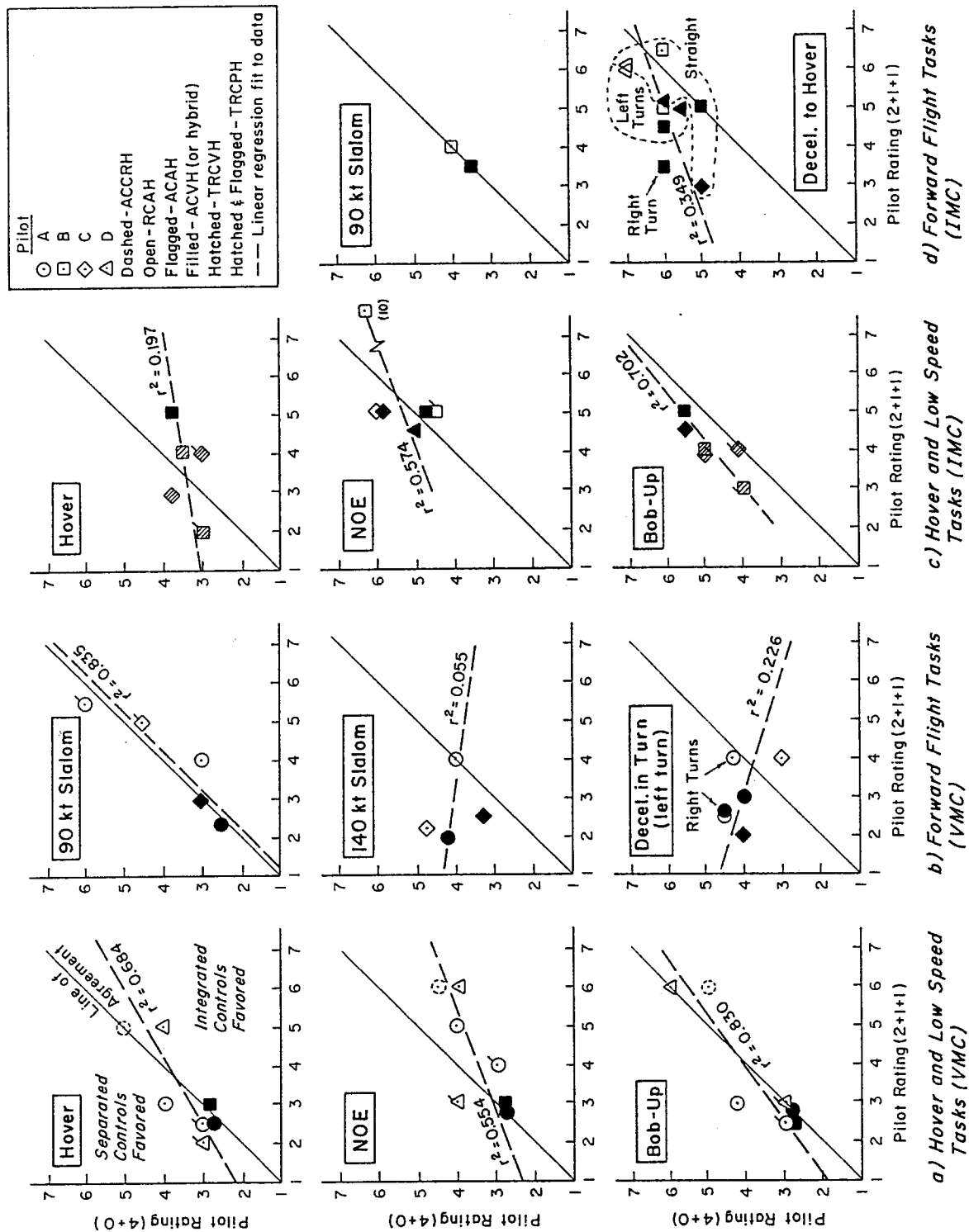


Figure 6(3.6.2). Crossplot of Cooper-Harper Pilot Ratings from Separated-Control (2+1+1) and Four-Axis Integrated-Control (4+0) Evaluations for ADOCS Simulation Phase 2 (Reference 14)

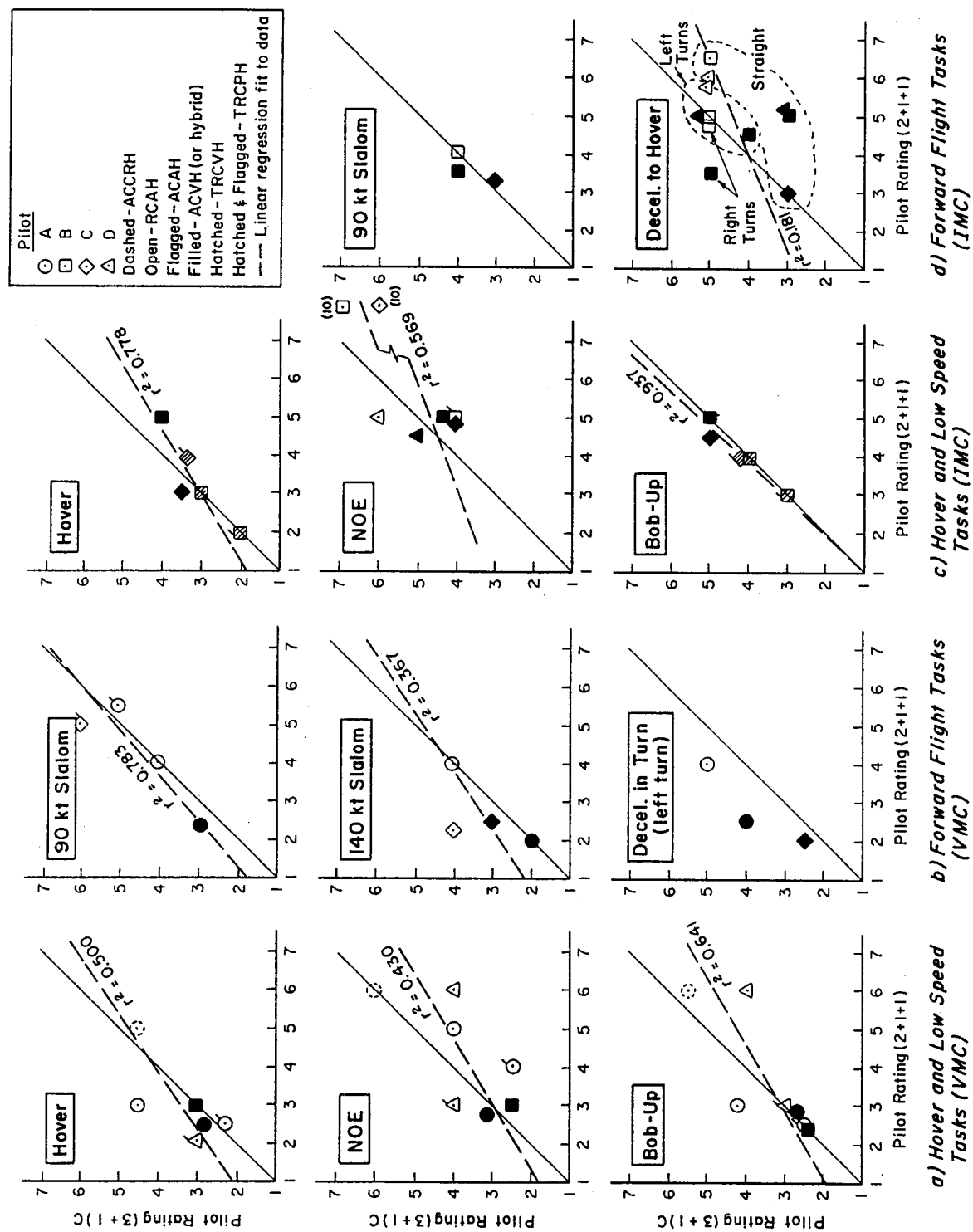


Figure 7(3.6.2). Crossplot of Cooper-Harper Pilot Ratings from Separated Control (2+1+1) and Three-Axis Integrated-Control with Separate Collective (3+1)C Evaluations for ADOCS Simulation Phase 2 (Reference 14)

Use of integrated sidestick controllers, where yaw is commanded via twist grip, or collective via vertical force, or both, opens an entirely new field requiring intensive study. At issue here will be not only the individual breakout forces to be used, but the interplay between axes to assure force harmony. Again, there is essentially no real data available in this area. In the Reference 14 ADOCS study, a yaw breakout moment of 0.4 in-lb was used; however, the experiment of Reference 61 had a relatively high breakout of 3 in-lb. In the vertical axis, a non-symmetric breakout was implemented on the ADOCS simulations with 1.25 lb required in the upward direction and 0.625 lb downward. Reference 61 used 1 lb in both directions. Again a good design guide for integrated collective control may be the lower breakout limit from Table 1(3.6).

#### FORCE GRADIENTS

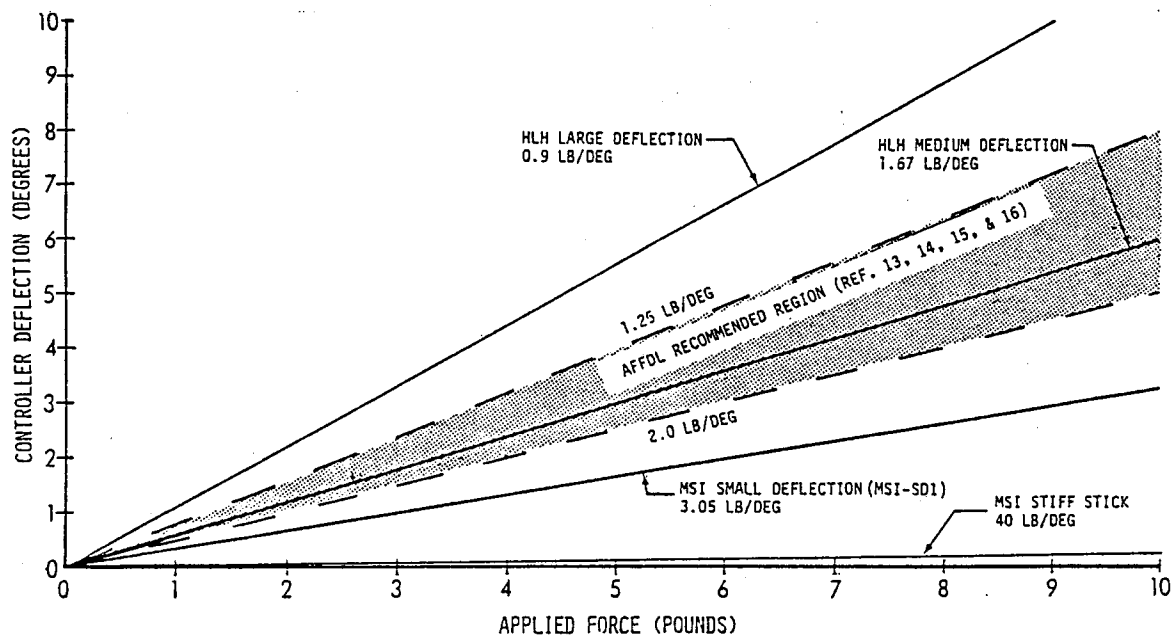
Control force/deflection characteristics for sidesticks are not specified in this paragraph due to a lack of supporting data. Some information can be obtained from the ADOCS simulations of Reference 14. Seven 4-axis sidestick configurations with varying force/deflection curves were evaluated during the two simulation phases, as listed in Table 1; the first four listed in the table were used to evaluate effects of control force/deflection characteristics. The longitudinal and lateral gradients (expressed in terms of lb/deg rather than lb/in.) for these four controllers are shown in Figure 8. Also shown in this figure is the range recommended by the Air Force (e.g., Reference 64) for conventional airplanes.

Figure 9 shows the results of the longitudinal/lateral gradient evaluations from the Phase 1 ADOCS simulation. Gradients shown represent the average of the longitudinal and lateral values from Table 1, while pilot ratings are the best ratings obtained for each configuration. Because of the shortcomings associated with using simulators for defining flying qualities limits, the results of Figure 9 should not be weighed too heavily. These data indicate that some small amount of stick motion is necessary, though it may be somewhat lower (in terms of deg/lb) than the region recommended by the Air Force (in Figure 8, the small-deflection stick gradient is outside the shaded region).



TABLE 1(3.6.2). 4-AXIS CONTROLLER CONFIGURATIONS EVALUATED IN  
ADOCS SIMULATIONS (REFERENCE 14)

4-AXIS CONTROLLER CONFIGURATIONS	SIMULATION PHASES		OPERATING FORCE LINEAR RANGE (±)						MAXIMUM DEFLECTION (±)						FORCE/DEFLECTION							
	PHASE 1	PHASE 2	X		Y		Z		ψ		X		Y		Z		ψ		X LONG DEG	Y LAT DEG	Z VERT IN	ψ YAW IN-DEG
			LONG LBS	LBS	LAT LBS	LBS	VERT LBS	IN- LBS	LONG DEG	LAT DEG	VERT IN	ψ DEG	LONG DEG	LAT DEG	VERT IN	ψ DEG						
(1) LARGE DEFLECTION - HLH PROTOTYPE	X		—	—	—	—	—	—	12.0	12.0	0.5	15.0		0.9	0.6	15.0	0.7					
(2) MEDIUM DEFLECTION - HLH PROTOTYPE	X		—	—	—	—	—	—	12.0	12.0	0.5	15.0		1.67	1.05	35.0	2.7					
(3) SMALL DEFLECTION - (PITCH AND ROLL) MSI-SD1	X		20	20	20	40	60		5.3	5.3	0.1	4.0		3.05	2.25	400	15.0					
(4) STIFF-STICK MSI-SS	X	X	20	20	20	40	60		0.5	0.5	—	—		40	40	—	—					
(5) SMALL DEFLECTION - (PITCH AND ROLL) MSI-SD2	X	X	20	20	20	40	60		8.3	8.3	0.15	6.0		1.82	1.45	267	10.0					
(6) SMALL DEFLECTION - (ALL AXES) MSI-SD3		X	12	12	12	+24 -21	36		6.6	6.6	.25	10.0		1.82	1.82	+95 -85	3.6					
(7) SMALL DEFLECTION - (ALL AXES) LSI - BRASSBOARD		X	15.9	12.8	15.8	35			7.6	7.6	.156	7		2.09	1.67	+95 -85	5.0					



### LATERAL AXIS

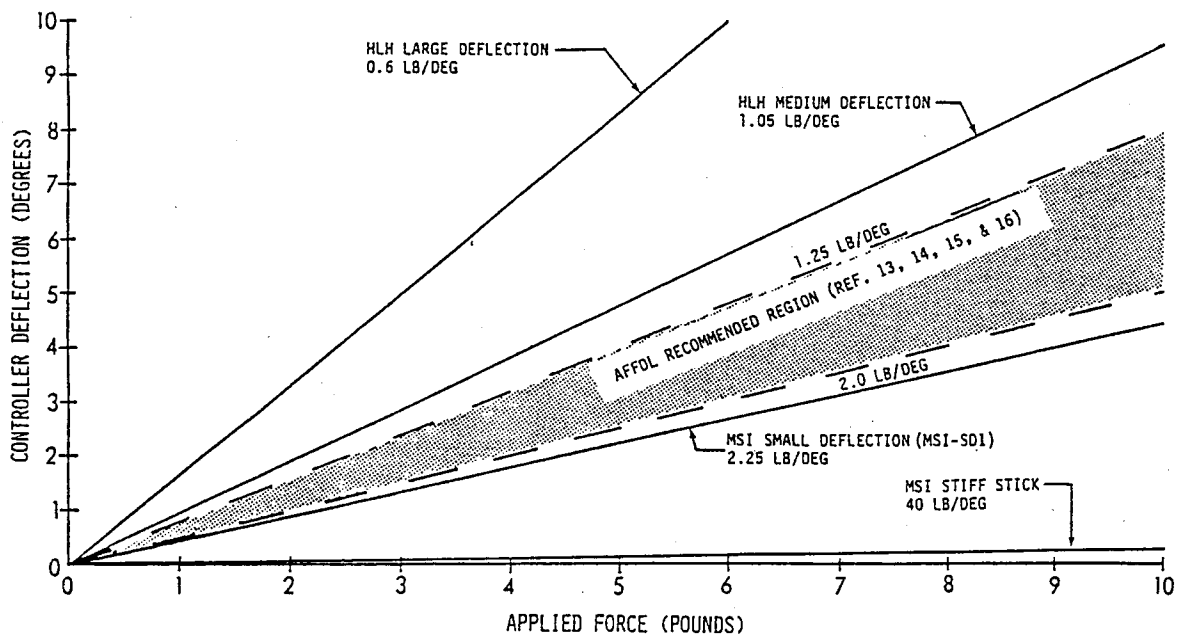


Figure 8(3.6.2). Simulation Force/Deflection Characteristics from ADOCS (Reference 14; Reference Numbers are for Reference 14)

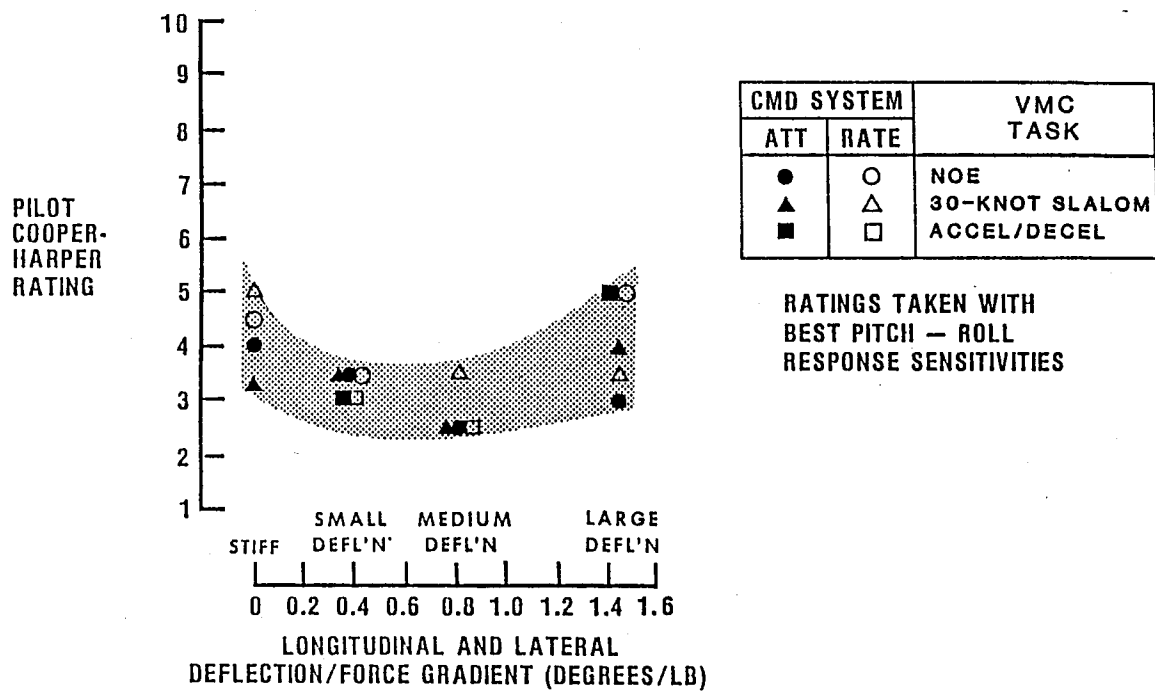


Figure 9(3.6.2). Effect of Sidestick Controller Deflection/Force Gradient on Pilot Ratings (From Reference 14; Best Rating Shown; Gradient is Average of Longitudinal and Lateral Gradient)

a. Statement of Requirement

3.6.3 Sensitivity and Gradients. The pitch, roll, yaw, and collective controller sensitivities and gradients shall be consistent with the aircraft dynamic response characteristics in each axis at all flight conditions. In no case shall the controller sensitivity or gradient produce responses which are objectionably abrupt or sluggish. Compliance with this requirement must be in actual flight, and must involve at least the maneuvers in Sections 4.1 and 4.2, and transitions between cruise and hover.

b. Rationale for Requirement

The systematic variation of pitch, roll, and yaw dynamic response characteristics conducted in the Reference 4 simulation and the Reference 157 NRC flight tests has shown that controller sensitivity was an extremely important factor in the pilots' evaluations. It was found that the sensitivities had to be tuned to the response, which, in general, meant that rapid responses required relatively low sensitivities and sluggish responses required comparatively high sensitivities. As noted in Reference 12, the appropriate value is most likely associated with the magnitude of the response in the region of piloted cross-over (between 0.5 and 4 rad/sec). Since controller sensitivity does not drive the design, and can be varied with relative ease, there is little motivation to make determinations of the complex relationship with response bandwidth, Response-Type, controller type (center vs. side-stick) and task. Therefore, only a qualitative requirement for controller sensitivity and gradient is included here.

c. Supporting Data

None.

d. Guidance for Application

Experience has shown that ground-based simulation is a poor place to set controller sensitivities and gradients, due to limitations in the motion and visual systems. Therefore, compliance is required in a flight environment. The Mission-Task-Elements specified for flight test compliance were selected to span the range of pilot activity from precision to large amplitude maneuvering.

Finally, research is required to investigate the effect of sensitivities on augmentation system failures. Of particular interest would be the determination of controller sensitivities that are optimized for the primary SCAS, but are unacceptable for the backup SCAS. Such cases should be investigated in a flight test environment for final specification compliance.

e. Related Previous Requirements

None.

a. Statement of Requirement

3.6.4 Cockpit Control Free Play. The free play in each control, that is, any motion of the cockpit control which does not move the appropriate moment- or force-producing device in flight, shall be compatible with the required Level of flying qualities.

b. Rationale for Requirement

Some limitation on the allowable control free play is essential. There are no data available to place a specific number on the limit, and hence this qualitative requirement from MIL-F-83300 is given. The amount of free play the pilot is willing to accept is obviously a function of both Mission-Task-Element and flying qualities Level.

c. Supporting Data

Cockpit control free play is not commonly cited in Army flight test reports (e.g., References 67-80) as a shortcoming. No supporting data is available for this paragraph.

d. Guidance for Application

None.

e. Related Previous Requirements

The MIL-H-8501A requirement (3.5.10) reads as follows: "For all operating conditions, there shall be no dead spots in any of the control systems which permit more than  $\pm 0.2$ -inch motion of the cockpit control without corresponding motion of the rotor blades, control surfaces, etc." The only difference between this requirement and the MIL-F-83300 requirement (3.5.1.3) adopted here, other than details of the wording, is the specific statement of 0.2 inches as a maximum value of free play. There is no compelling evidence that this is a good limit; in Reference 81, flight tests of the Bell Jet Ranger support a much lower limit, as the following statements indicate: "poor force harmony coupled with a longitudinal free play (boost-on) of approximately 1/8 inch contributed to generally poor flying qualities," Cooper rating of A-4 ("Acceptable, but with unpleasant characteristics; see Table 3(3.2.2)) in forward flight; and for directional controls, "free play was 1/8 inch boost on and was excessive in combination with other items contributing to spastic control action in hover," Cooper rating of A-5 ("Unacceptable for normal operation"). This is not strong evidence, but does suggest that there may be occasions when 0.2 inch of free play may be deemed excessive. It is felt that the general statement from MIL-F-83300 is more appropriate.

a. Statement of Requirement

3.6.5 Control Harmony. The control forces, displacements, and sensitivities of the pitch, roll, yaw, and collective controls shall be compatible, and their responses shall be harmonious.

b. Rationale for Requirement

Ideally, this requirement would state specific control force and displacement ratios for each of the cockpit controllers to assure control harmony. Lack of harmony is sometimes cited as a shortcoming for operational helicopters (References 67-81), but there is insufficient data to determine numerical ratios. Such ratios are to some extent a function of the other control characteristics (including breakout and limit force and deflection) and are thus even more difficult to quantify.

c. Supporting Data

Lack of control force harmony for the AH-56A (Reference 70) resulted from excessive directional control breakout and force gradients: "Large directional breakout and friction forces were... apparent to the pilot and not in harmony with the lateral and longitudinal control forces." For the Bell Jet Ranger (Reference 81) lack of longitudinal force feel required an increase in the force gradient in the feel system, resulting in poor longitudinal/lateral control force harmony.

d. Guidance for Application

None.

e. Related Previous Requirements

This requirement is taken directly from MIL-F-83300 (3.5.1.6).

a. Statement of Requirement

3.6.6 Dynamic Coupling. There shall be no tendency for dynamic coupling between the rotorcraft and the controller with or without the pilot in the loop. In particular, lightly damped, high frequency oscillations which cease when the pilot releases the controller are not permitted.

b. Rationale for Requirement

The motivation for this requirement stems from unpublished results obtained on the XV-15 tilt rotor aircraft, and unlike other requirements in this section, relates from experience with a sidestick controller. Specifically, the accelerations at the cockpit resulted in uncommanded controller inputs due to mass imbalance effects (hands on and hands off). This resulted in serious aircraft oscillations immediately after liftoff. The immediate fix has been to increase the damping of the controller to very high values, which in turn results in a significant penalty in the achievable bandwidth of the attitude response. Work is currently in progress to develop and test a filtering scheme which effectively results in an inverse model of the local cockpit accelerations -- such a solution may be thought of as an electrically mechanized double bobweight.

The arm-bobweight effect has been observed on numerous occasions in flight when very high bandwidth systems are programmed on inflight simulators (i.e., NAE Bell 205). The motion systems of current simulators do not have sufficient fidelity to reproduce this effect. It has been observed on the VMS, however, due to a resonant peak in the motion system at approximately 4 Hz.

c. Supporting Data

None.

d. Guidance for Application

None.

e. Related Previous Requirements

None.



a. Statement of Requirement

3.7 SPECIFIC FAILURES

The requirements in this paragraph relate to specific system failures that are to be treated without regard to their probability of occurrence (Paragraph 3.1.7.2).

3.7.1 Failures of the Flight Control System. The following events shall not cause dangerous or intolerable flying qualities:

- Complete or partial loss of any function of the flight control system following any single failure.
- Failure-induced transient motions and trim changes either immediately after failure or upon subsequent transfer to alternate control modes.
- Configuration changes required or recommended following failure.

b. Discussion

Paragraph 3.7.1 is taken, with slight modification, from MIL-F-8785C (3.5.5). The intent is to assure that a single flight-control-system failure, as well as any subsequent automatic or pilot-initiated correction for the failure, will not result in a severe degradation in flying qualities. This paragraph specifies what is allowable, in terms of handling qualities, for the failure analysis of Paragraph 3.1.7.4 when the components under consideration comprise the flight control system.

a. Statement of Requirements

3.7.2 Engine Failures. From any condition within the Service Flight Envelope the rotorcraft shall be capable of:

- Safely sustaining a single-engine failure.
- Safely entering into power-OFF autorotation.
- Safely landing from all points outside the height-velocity curve.

3.7.2.1 Altitude Loss. For multi-engined rotorcraft, in level flight, at the minimum speed where altitude can be maintained with one engine inoperative, the allowable altitude loss following a single-engine failure shall be no more than 15 m (50 ft).

b. Discussion

The rotorcraft engine failure requirements are provided to insure that an engine failure by itself (a loss of power) does not result in loss of the crew and the rotorcraft.

The need for these requirements and the form presented is supported by numerous paragraphs in MIL-H-8501A (Reference 31), the AAH PIDS (Reference 130), the UTTAS PIDS (Reference 131), as well as other flying qualities specifications.

Ultimately, these requirements shall be demonstrated in flight test for the critical power settings, loadings, and airspeeds. As an aid in reducing the number of flight test data points, the critical conditions should be investigated prior to flight test through some form of simulation. Based upon these results, a more efficient test plan for proof of compliance can then be negotiated with the procuring activity.

a. Statement of Requirement

3.7.3 Loss of Engine and/or Electrical Power. Complete or partial loss of engine power and/or loss of an electrical subsystem shall not result in handling qualities worse than Level 2. Engine start during single engine or autorotational flight shall not cause the flight control system(s) to fail or become inoperative.

b. Discussion

The requirement for not permitting a degradation to Level 3 handling qualities following loss of engine or electrical power is for safety reasons. Following an engine failure, especially in a single-engine rotorcraft, there can be a significant rotorcraft angular response before the pilot can enter stabilized autorotation. The handling qualities must remain sufficiently good for the pilot to maintain control in this situation all the way through to a safe autorotational landing. Likewise, the loss of one of the several electrical subsystems should not result in an uncontrollable handover or component shutdown because the pilot's attention will need to be focused on the electrical failure and not on rotorcraft stability at the same time.

This specification is supported by similar requirements in the AAH PIDS, (Reference 130). Background for existence of this paragraph can be supported by pilot comments such as those of Reference 132. In this flight test evaluation, the simulation of single generator failures on either engine of the AH-64 resulted in electrical transients or spikes which often disabled the automatic flight control system (AFCS). The mode of operation, upon sensing of an electrical failure of the AFCS of the AH-64, is to center the AFCS actuators and lock them. These "centering hardovers" were judged in this case to be too severe for the pilot to be capable of safely recovering from in night NOE flight, since the available visual cues might not be very useful in the short amount of time available for recovery.

a. Statement of Requirements

3.8 TRANSFER BETWEEN RESPONSE-TYPES

The transients and trim changes caused by the intentional transfer between Response-Types shall not be objectionable.

3.8.1 Annunciation of Response-Type to the Pilot. If more than one Response-Type can be selected in a given axis, there shall be a clear and easily interpretable annunciation to the pilot indicating which of the Response-Types are currently engaged or armed. For near-earth operations, it shall not be necessary for the pilot to significantly shift his eye point of regard from the forward near-field or to look around, or refocus any vision aid to view the required annunciation.

3.8.2 Control Forces During Transfer. With the rotorcraft initially trimmed at a fixed operating point, the peak pitch, roll, yaw, and collective control forces required to suppress the transient aircraft motions resulting from the transfer and maintain the desired heading, attitude, altitude, rate of climb or descent, or speed without use of the trimmer control, shall not exceed one-third of the appropriate limit control force in Table 4(3.6). For blending, this applies over the time interval specified in 3.8.3 following completion of the pilot action initiating the blend. There shall be no objectionable buffeting or oscillations of the control device during the blend.

3.8.3 Control System Blending. Blending between Response-Types shall be essentially linear with time and shall occur within the time limitations specified below.

Blending During Deceleration       $2 \text{ sec} \leq t_{\text{blend}} \leq 10 \text{ sec}$

Blending During Acceleration       $2 \text{ sec} \leq t_{\text{blend}} \leq 5 \text{ sec}$

When blending from a series to a parallel trim system, a longitudinal or directional trim follow-up may be used as long as the cockpit controller does not move more than 20 percent of its travel in either direction during the blend.

b. Discussion

If Mission-Task-Elements are specified that require different Response-Types in the same axis, it is necessary to blend between the Response-Types.

The specified limits on blending time are based on pilot comments from the Reference 3 simulation of transitions from forward flight to hover and are not supported by Cooper-Harper handling qualities ratings. Blending times over 10 sec were found to be excessive and resulted in blends which continued to occur even after the transition to hover was completed. This tended to degrade hover performance during conditions of low visibility or high winds. Blends under 2 sec tended to be abrupt under some conditions. The most critical of these conditions is blending from an Attitude Response-Type in hover to a Rate Response-Type during a maximum effort acceleration from the hover flight condition. Such an acceleration involves full cockpit control deflections to get the pitch attitude required to accelerate. A blend to the Rate Response-Type results in a momentary command of full nose-down pitch rate which can cause attitude upsets if the pilot is not extremely alert.

Control system blending characteristics depend on the trim system used in the Attitude Response-Type (Rate Response-Types are inherently series trimmed in that neutral stick always corresponds to zero pitch rate). The attitude command system can be either parallel or series trim. Parallel trim is generally preferred by pilots because it is more natural, i.e., forces can be trimmed off without repositioning the stick to its zero position. In addition, it is possible to determine the remaining control power by the position of the stick. For example, when hovering in a strong tailwind a large aft stick position tells the pilot that he is near the limit. The drawback to using parallel trim is that blending between series and parallel involves a shift in the stick position during the blend. Most simulations where control system blending has been employed have resorted to using series trim in the attitude command mode specifically to avoid this issue. In these simulations it was usually not necessary to do a great deal of retrimming in the hover mode. In the real-life situation, however, retrimming is quite common during nap-of-the-earth maneuvering or simply regulating against different steady wind conditions which may occur due to shadowing behind trees, ridges, or parts of a ship.

Three types of blends were investigated during the piloted moving-base simulation to study transition reported in Reference 3. These were:

- 1) series rate command to series attitude command
- 2) series rate command to parallel attitude command with a spring-loaded thumb switch on top of the stick for manual retrimming
- 3) series rate command to parallel attitude command with an automatic trim follow-up which drove the stick to the trim position during the blend

Comparison of the Cooper-Harper handling qualities ratings between these different type blends showed no particular preference. However, manual retrimming after the blend was noted by the pilots to be an objectionable feature in that it resulted in some pitch bobbling. The use of a trim follow-up which automatically repositioned the stick to hold the attitude existing at the initiation of blend was found to be desirable. The only problem with a trim follow-up is in cases where off-nominal pitch attitudes exist at blend initiation. Such situations would require manual retrimming to establish a stable hover with zero stick force. Without an automatic trim follow-up, manual retrimming is required in every case. Therefore, it is felt that the best compromise is to allow either manual trim or an automatic trim follow-up in the specification, with a limit on the off-nominal stick forces that can result from control system blending, and which must be trimmed out by the pilot during the initial stages of hover.

Control system blending may also be accomplished at higher speeds, such as during the transition to STOL for a tilt-rotor or tilt-wing rotorcraft. All of the arguments discussed above still apply except that off-nominal stick forces following a blend could result in large normal accelerations if not corrected. Therefore, the specification is written to require that a specified level of stick force be required to maintain trim. The allowable level of stick force that would have to be trimmed off following a blend is set at 1/3 of the limit cockpit control forces in Table 4(3.6).

Finally, a limit on the amount of cockpit control travel that can occur automatically during a blend is limited to 20 percent of full travel to avoid large transients that could occur due to a failure in the blending system or blending during some off-nominal condition.

a. Statement of Requirements

3.9 GROUND HANDLING AND DITCHING CHARACTERISTICS

3.9.1 Rotor Start/Stop. It shall be possible, while on the ground, to start and stop the rotor blades in mean winds up to at least 45 knots (23 m/s) from the most critical direction.

3.9.1.1 Shipboard Operation. It shall be possible to bring the engines to idle power without engaging the rotor(s), and to stop the blades within 20 seconds after engine shutdown.

b. Discussion

The use of a helicopter in all weather conditions will require that missions be started and ended in windy and turbulent weather. Therefore, it must be a requirement that the rotor blades be capable of being started and stopped safely in the mission weather conditions.

The same requirement was utilized in the specification of the UTTAS and AAH as based on the mission requirements defined by the U.S. Army. MIL-H-8501A (3.5.1) specifies, in addition, that for ship-based helicopters the wind limit be increased to at least 60 kts. The requirements in Paragraph 3.9.1.1 are based on Navy operational experience.

a. Statement of Requirements

3.9.2 Parked Position Requirement. It shall be possible, without the use of wheel chocks or skid restraints, to maintain a fixed position on a level paved surface at the normal takeoff rotor speed as power is increased prior to lift-off.

3.9.3 Wheeled Rotorcraft Ground Requirements. The following ground handling conditions shall be met for all operational weather conditions.

a. It shall be possible, without the use of brakes, to maintain a straight path while taxiing or performing rolling takeoffs or landings in a wind of up to 45 knots (23 m/s) from any direction.

b. It shall be possible to make complete 360-degree turns in either direction by pivoting on either main landing gear in a wind of up to 45 knots (23 m/s) from any direction. These turns shall be made within a radius equaling the major dimension of the rotorcraft.

c. It shall be possible to perform all required maneuvers, including taxiing, rolling takeoffs and landings, and pivoting, without damage to rotor stops and without contact between the main rotor or tail rotor blades and any part of the rotorcraft structure.

b. Discussion

The specified requirements for ground handling qualities are provided to ensure both crisp and safe control of the rotorcraft by the pilot while on the ground.

These requirements are supported by similar paragraphs in MIL-H-8501A, MIL-F-83300, the UTTAS PIDS (Reference 131), and the AAH PIDS (Reference 130). Paragraphs 3.9.1.3a and 3.9.1.3b are included from MIL-F-83300 in order to provide detailed design guidance for taxiing. In several of the paragraphs in previous specifications, the wind requirement was 35 knots instead of the 45 knots specified here. The 45-knot requirement was chosen in order to ensure that the ground handling requirements of the rotorcraft be compatible with the rotor start/stop requirements and the ground handling/mission profiles of the AAH and UTTAS, as well as the limits of MIL-H-8501A (3.5.4.1).



a. Statement of Requirements

3.9.4 Ditching Characteristics. If required by the procuring activity, the following characteristics shall be provided either as part of the rotorcraft design or in supplementary kit form.

3.9.4.1 Water Landing Requirement. In both power-ON and power-OFF autorotative conditions, it shall be possible to make a safe landing on smooth water up to at least 20 knots (10.5 m/s) surface speed with an 8 ft/sec (2.4 m/s) rate of descent and at least 30 knots (16 m/s) with a 5 ft/sec (1.5 m/s) rate of descent at angles of yaw up to 15 degrees.

3.9.4.2 Ditching Requirement. Techniques and procedures (i.e., recommended pitch attitude and airspeed conditions) shall be established for ditching the rotorcraft on water in the event of:

- a. The loss of all engine power.
- b. The failure of one engine in a multi-engine rotorcraft.

The state of the rotorcraft during ditching up to Sea State 6 shall not be such as to cause immediate injury to the occupants or to make it impossible for the occupants to exit the rotorcraft from the emergency exits provided (i.e., due to an adverse static water level on the cabin emergency exits or the blocking of emergency exits by all or part of the flotation system).

3.9.4.3 Flotation and Trim Requirements. The flotation time and trim attitude characteristics of the rotorcraft shall be such that the crew and passengers are provided with a sufficient length of time to exit the rotorcraft safely and to enter life rafts without application of a rotor brake. A sea anchor, or similar device, shall not be used in demonstrating compliance; however, it may be used to assist in deployment of life rafts. The flotation time and trim attitude characteristics shall be investigated throughout the range of Sea States from 0 to 6 and shall be satisfactory in waves having height/length ratios typical of these Sea States.

3.9.4.4 Single Failures of the Flotation Equipment. The flotation time and trim attitude requirements of Paragraph 3.9.4.3 shall also be met with the most critical compartment of the flotation system inoperative (i.e., as caused by a leak deflating the compartment or a failure in actuation of the inflation mechanism) up to Sea State 3, and if possible up to Sea State 6.

b. Discussion

Ditching may be defined as an emergency landing on water, deliberately executed with the intent of abandoning the rotorcraft as soon as

practical. The rotorcraft is assumed to be intact prior to water entry with all controls and essential systems, except engines, functioning properly. The main objective of these paragraphs on water handling characteristics is to provide handling qualities design guidance to:

- a. Preserve the lives of, and minimize injury to, the occupants of the rotorcraft during water impact.
- b. Enhance the opportunities for occupant survivability after impact by providing time for exit from the rotorcraft into life rafts for subsequent rescue.

Very little guidance is provided in MIL-H-8501A for water handling qualities, and that which is provided does not address the environmental conditions in which the rotorcraft is expected to operate. The UTTAS and AAH were not intended for extensive over-water operation; therefore, the respective PIDS never specified any requirements for ditching or water handling characteristics.

Serious specification development on this topic has almost exclusively been conducted by Great Britain in the development of British Civil Aviation Regulations (BCARs). Much of this work was begun in the 1970s in response to a need for regulations on rotorcraft operating between Scotland and the North Sea oil fields. This work resulted in the introduction of Section G4-10 into the BCARs in 1975 (Reference-128). This work was further expanded and revised in 1983 to its present form following a review of the North Sea accident data base and the operational profiles being used by the North Sea operators. Reasons for the changes from the original regulations are contained in Reference 129. Recent work being conducted by the FAA in an effort to develop ditching requirements also relies heavily on the British work. These updated requirements help to form the basis for the specifications developed in Paragraphs 3.9.4.1 through 3.9.4.4 of this specification.

Demonstration of specification compliance for ditching and water handling characteristics is clearly not easily nor cheaply demonstrated by flight test. Therefore, for Paragraphs 3.9.4.2 through 3.9.4.4, it is required as a minimum that model testing be used to insure that ditching as well as flotation and trim requirements are clearly acceptable. This type of testing has been utilized effectively by the civilian sector of the rotorcraft industry for more than a decade. It is also recommended strongly that the procuring activity require the manufacturer to demonstrate the use of flotation equipment and crew egress capabilities by immersion of a full-scale fuselage in calm water (i.e., in a water tank before systems are installed, ballasting may therefore be required). This type of test is crucial in demonstrating that crew emergency exits/doors are not blocked by the inflated flotation equipment or by an adverse water level on the fuselage. This type of test has also been utilized quite extensively in the civilian rotorcraft sector and has been quite successful in uncovering numerous equipment and design deficiencies (i.e., accidental puncture of flotation bags during inflation) which were corrected prior to rotorcraft certification.

Table 1 presents the currently accepted definition of Sea States.

TABLE 1(3.9.4). SEA STATE CODE  
(WORLD METEOROLOGICAL ORGANIZATION)

SEA STATE	SIGNIFICANT WAVE HEIGHT	DESCRIPTION OF SEA	SEA STATE	SIGNIFICANT WAVE HEIGHT	DESCRIPTION OF SEA
0	0 ft (0 m)	Glassy (Calm)	5	8 to 13 ft (2.5 to 4 m)	Rough
1	0.0 to 0.33 ft (0.0 to 0.1 m)	Glassy (Rippled)	6	13 to 20 ft (4 to 6 m)	Very Rough
2	0.33 to 1.67 ft (0.1 to 0.5 m)	Smooth (Wavelets)	7	20 to 30 ft (6 to 9 m)	High
3	1.67 to 4 ft (0.5 to 1.25 m)	Slight	8	30 to 45 ft (9 to 14 m)	Very High
4	4 to 8 ft (1.25 to 2.5 m)	Moderate	9	Over 45 ft (over 14 m)	Phenomenal (Confused Sea)

Notes:

1. The significant wave height is defined as the average value of the height (vertical distance between trough and crest) of the largest one-third of the waves present.
2. Maximum wave height is usually taken to be 1.6 times the significant wave height, i.e., a significant wave height of 6 m gives a maximum wave height of 9.6 m.

a. Statement of Requirement

4. FLIGHT TEST MANEUVERS

A selection of flight test maneuvers are specified to provide an overall assessment of the rotorcraft's ability to perform selected critical tasks. Many, but not all, of the maneuvers correspond to Mission-Task-Elements in Tables 1(3.2) and 2(3.2). In Sections 4.3 and 4.5, only those maneuvers corresponding to required Mission-Task-Elements must be demonstrated, whereas all maneuvers in Sections 4.1, 4.2, and 4.4 must be accomplished. The maneuvers have been tailored to emphasize ease of testing; they are not comprehensive, and demonstration of desired performance does not imply compliance with Section 3.

The demonstration maneuvers must be accomplished by at least three pilots. The arithmetic average of the Cooper-Harper pilot ratings must be in the Level 1 range when operating in the Operational Flight Envelopes. All individual ratings and associated commentary must be documented and supplied to the procuring activity. It is desired that the maneuvers be performed at the Normal States within the Operational Flight Envelope that are most critical from the standpoint of flying qualities. It is emphasized that the rotorcraft performance is not an issue in these tests, and that the flight conditions should be selected accordingly. It is not necessary to test in a high density altitude environment unless the rotorcraft does not meet the boundaries in Section 3 for such a condition, and satisfactory results in this section are being used to support a deviation. Altitude requirements refer to a selected reference point on the rotorcraft. The physical location of the selected reference point is not critical from the handling qualities point-of-view, since it is the change in altitude that defines the maneuver performance.

b. Discussion

The qualitative flight test evaluations in this section are included as an integral part of the specification, to provide an independent assessment of the rotorcraft handling qualities. It is recognized that the design criteria in Section 3 are based on present knowledge, and in some cases a limited data base, and therefore, meeting these criteria may not guarantee desirable handling qualities. Conversely, failing one or more of the requirements is not necessarily a guarantee of less than desirable handling qualities (although it is highly probable).

It is important to recognize that the maneuvers in this section represent a compromise between a formal definition of some of the Mission-Task-Elements specified in Tables 1(3.2) and 2(3.2), and tasks which were found to be easily flight testable. Therefore, they are not a substitute for meeting the quantitative, and much more comprehensive, criteria in Section 3.

The specific maneuver definitions were initially formulated based on previous flight tests and Letter-of-Agreement (LOA) requirements for the proposed LHX helicopter (Reference 133). These "strawman" maneuvers were then tested at the Army Aviation Engineering Flight Activity (AEFA) at Edwards Air Force Base using a UH-60 Blackhawk and two Army test pilots (Reference 195). Based on these tests, the strawman maneuvers were refined or completely modified to obtain a compromise between being easily flight testable, and still a valid test of the rotorcraft handling qualities, as required to perform the Mission-Task-Elements. Finally, the maneuvers were reviewed by industry and government technical personnel and pilots, and refined accordingly.

To the extent possible, the demonstration maneuvers are to be conducted in the style of a carefully developed handling qualities experiment. In that context, at least three evaluation pilots are required (and five would be preferable), Cooper-Harper ratings are required for each maneuver, desired performance is specified for each maneuver, and pilot commentary must be documented and supplied to the procuring activity. It is important to understand that the "desired performance" specified for each maneuver is provided for direct use by the pilot in using the Cooper-Harper scale [see Figure 1(2.8)]. Specifications of adequate performance are not supplied since it was felt that verifying the boundary between Level 2 and Level 3 is beyond the scope of these tests (recall that "adequate performance" forms a part of the Cooper-Harper definitions for ratings of 5, 6, and 7).

The intent of these maneuvers is to check the handling qualities at the most critical flight conditions and loadings, from the standpoint of controllability, not performance.

a. Statement of Requirement

4.1 PRECISION TASKS

4.1.1 Hover. Maintain a precision hover for at least 30 sec in winds of at least 20 knots from the most critical direction. If a critical direction has not been defined, the hover shall be accomplished with the wind blowing directly from the rear of the rotorcraft. The hover altitude shall be equal to or less than 6.1 m (20 ft).

- Desired Performance

- Maintain horizontal position of the pilot's station within 0.91 m (3 ft) of a reference point on the ground.
- Maintain altitude within  $\pm 0.61$  m (2 ft).
- Maintain heading within  $\pm 5$  degrees.
- There shall be no objectionable oscillations in any axis. In particular, oscillations which interfere with precision control, or with operation of controls or switches, would be deemed objectionable.

b. Discussion

This maneuver was included as an overall check on the handling qualities in a steady hover. Helicopter handling qualities in hover are typically degraded in winds over 15 knots, hence it is required that the maneuver be demonstrated in winds of at least 20 knots. From a handling qualities standpoint, a tailwind usually represents the worst case due to the destabilizing effect of the tail rotor. However, if some other wind azimuth is known to be more critical, or equally critical, it should be tested. It is recognized that the wind magnitude and direction will vary somewhat during the testing. This is not deemed to be critical as long as the average wind-speed is at least 20 knots, and the variability is not excessive.

A requirement for Level 1 flying qualities while hovering in a 20 knot tailwind is quite stringent, but is consistent with the specification philosophy (i.e., Level 1 handling qualities inside the Operational Flight Envelope).

The values for desired performance are based on experience in testing helicopters with Level 1 and Level 2 handling qualities. In most cases, the performance limits were derived from variable stability testing at the NRC, and the Reference 195 AEFA tests.

a. Statement of Requirement

4.1.2 Hovering Turn. From a steady hover at an altitude of not greater than 6.1 m (20 ft), complete a 180 deg turn as rapidly as possible, in both directions, with a wind of at least 20 knots from the most critical direction. If a critical direction has not been defined, the turn shall be completed with the wind blowing directly from the rear of the rotorcraft.

- Desired Performance
  - Maintain the horizontal position of the pilot's station within 1.8 m (6 ft) of a reference point on the ground.
  - Maintain altitude within  $\pm 0.91$  m (3 ft).
  - The final rotorcraft heading shall be stabilized at 180 degrees from the initial heading within  $\pm 3$  degrees.
  - The turn shall be completed (within the  $\pm 3$  degree window) in a time which is equal to or less than 5 sec from initiation of the maneuver.

b. Discussion

This maneuver is intended to simulate a rapid turn to engage a threat from the rear, and is one of the LOA requirements proposed for the LHX, i.e., the 5 second time requirement was taken from the Reference 133 LOA. The  $\pm 3$  degree heading tolerance is based on a need to aim and fire a weapon at the completion of the turn. This was relaxed from the LOA requirement of  $\pm 2$  degrees, as being excessively stringent, based on the AEFA tests. Similarly, the allowable position excursion of 6 feet represents a slight relaxation from the LOA requirement of 5 feet.

Time histories of a similar, but less aggressive, maneuver as performed at AEFA while tracking a pace car at 35 knots with the initial "wind" from 180 degrees are shown in Figure 1. Based on numerous review comments, the concept of using a pace car for these maneuvers has been deleted. Most adverse commentary centered about the lack of adequate visual cueing while tracking a pace car, a factor which is not significant for performance testing, but is of primary importance in handling qualities evaluations.

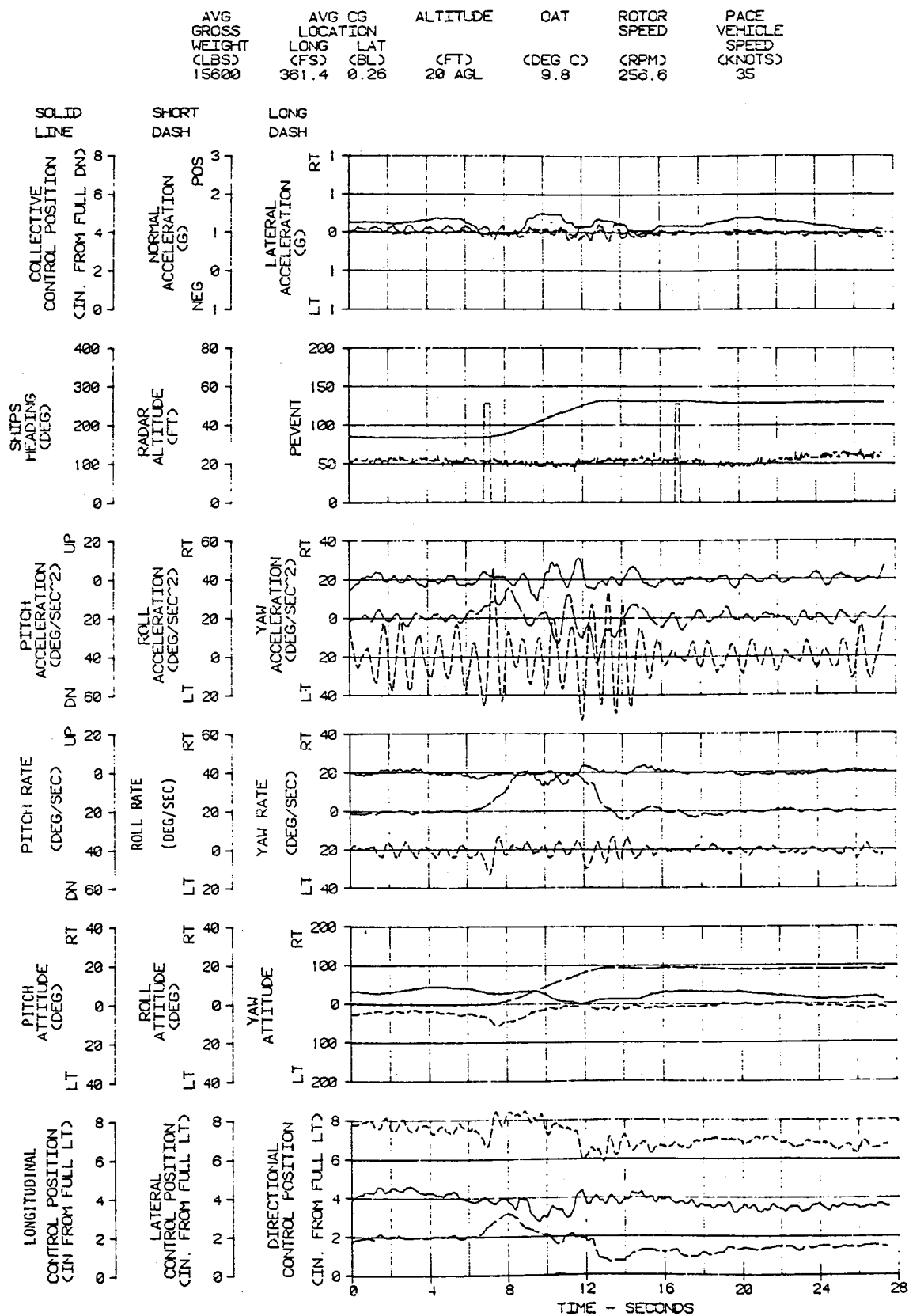


Figure 1(4.1.2). Maximum Hovering Turn: 90 Deg Right Turn



a. Statement of Requirement

4.1.3 Vertical Landing. Perform a rapid vertical landing from a stabilized hover at an altitude of 7.9 m (26 ft), or the maximum safe altitude in case of an engine failure, in winds of at least 20 knots.

- Desired Performance
  - Maintain horizontal position within 0.91 m (3 ft) of a defined reference point on the ground for the entire descent and touchdown.
  - Maintain heading within  $\pm 5$  degrees.
  - The vertical descent must be continuous, without the need to arrest the sink-rate to eliminate horizontal drift. There shall be no perceptible horizontal drift at touchdown.
  - The time from maneuver initiation to touchdown shall be equal to or less than 6 sec.

b. Discussion

This maneuver was developed by the AEFA during the test program, and is intended to be representative of tactical landings where time is more important than finesse. The time limit of 6 sec is based on the maneuver conducted at AEFA. The time histories of the maneuver are shown in Figure 1. It was felt that the UH-60A results are valid as it has Level 1 heave damping based on this specification. The touchdown performance limit on horizontal position of 3 feet was felt to be reasonable by the AEFA flight test personnel. However, it was pointed out that the ability to make such precision landings required a field-of-view such that the pilot can see the landing gear.

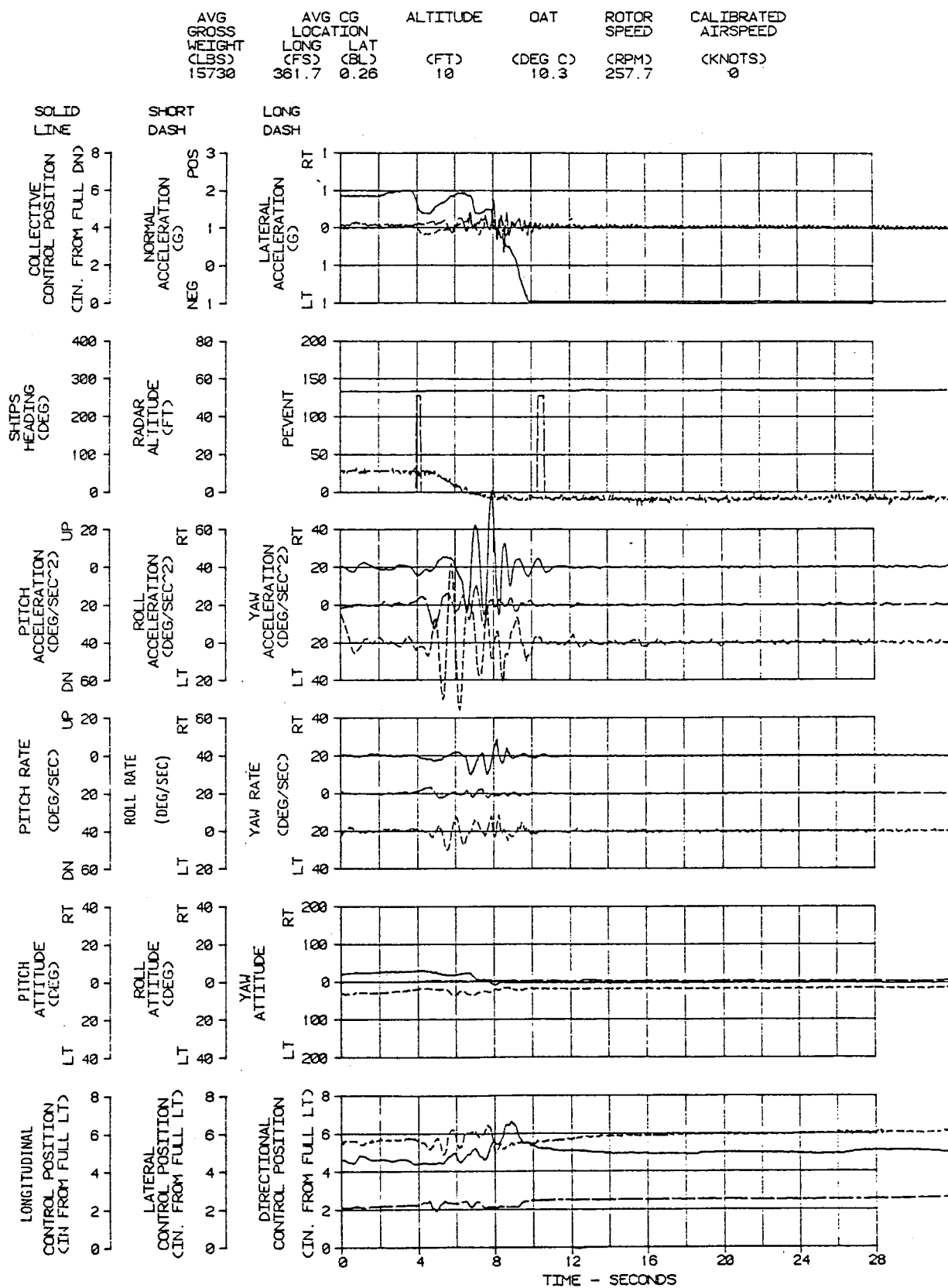


Figure 1(4.1.3). Vertical Landing From In-Ground-Effect Hover

a. Statement of Requirement

4.1.4 Pirouette. Initiate the maneuver from a stabilized hover over a point on the circumference of a 30.5 m (100 ft) radius circle, marked on the ground, with the nose of the rotorcraft pointed at a reference point at the center of the circle, and at a hover altitude of approximately 3 m (10 ft). Accomplish a lateral translation around the circle, keeping the nose of the rotorcraft pointed at the center of the circle, and the circumference of the circle under the pilot station. Perform the maneuver in both directions.

- Desired Performance
  - Maintain the circumference of the circle within  $\pm 3$  m (10 ft) of the pilot station.
  - Maintain altitude within  $\pm 0.91$  m (3 ft).
  - Maintain heading so that the nose of the rotorcraft points at the center reference point within  $\pm 10$  degrees.
  - The maneuver shall be completed in less than 45 sec, starting from the initial control input, and terminating in a stabilized hover.
  - The lateral velocity should be smooth and controlled throughout the maneuver. Note: the nominal lateral velocity will be approximately 8 knots.

b. Discussion

This maneuver was developed and refined during several flight tests using the NRC variable stability Bell 205A. It involves considerable coordination between all of the helicopter controls, especially in a wind of 10 knots or greater. For example, experience with testing various sidestick controllers at the NRC has shown that the pirouette tends to expose deficiencies in multiple axis manipulators that do not show up in other tasks, such as hover and vertical landings. A specific wind magnitude has not been specified for the maneuver since experience has shown that evaluation pilots have been able to distinguish between good and bad configurations in no-wind testing. However, when performed in winds of over 10 knots, the pirouette tends to magnify handling deficiencies, especially if the trim attitude is a strong function of airspeed. Even though the pirouette is not a Mission-Task-Element, it has been included as a proven maneuver to expose handling qualities deficiencies, especially those associated with manipulators and feel systems. The pirouette has also proven to be an excellent maneuver for evaluating handling qualities in degraded visual environments (hence its inclusion in Section 4.4).

a. Statement of Requirement

4.1.5 Slope Landing. Perform vertical landings to a sloped surface pointing upslope, downslope, and in both cross-slope directions using the approximate maximum slope achievable, as defined by the aircraft performance limits. The test may be performed in winds less than 5 knots.

- Desired Performance
  - Maintain horizontal position within 0.91 m (3 ft) of a defined reference point on the ground for the entire descent and touchdown.
  - Maintain heading within  $\pm 5$  degrees.
  - The vertical descent must be continuous, without the need to arrest the sink-rate to eliminate horizontal drift. There shall be no perceptible horizontal drift at touchdown.
  - Maintain precise control of downslope skid or wheel height.

b. Discussion

The requirement for precision slope landings is self-evident. It is important to note that the flight control system must not have any active integrators during this maneuver, a requirement established in Paragraph 3.2.9. This maneuver was recommended by AEFA but was not demonstrated during the test program.

a. Statement of Requirement

4.2 AGGRESSIVE TASKS

4.2.1 Acceleration and Deceleration. Starting from a stabilized hover, initiate a rapid and aggressive longitudinal acceleration up to an airspeed of at least 60 knots, and immediately decelerate to hover over a defined reference point. Maintain a constant altitude at or below 12.1 m (40 ft).

- Desired Performance

- Complete maneuver over the reference point at the end of the course. Tolerance is plus zero and minus 3 m (10 ft) (positive forward).
- Maintain altitude within  $\pm 4.6$  m (15 ft).
- Maintain heading within  $\pm 10$  degrees.
- Achieve maximum longitudinal acceleration in 1.5 sec or less.
- Achieve maximum longitudinal deceleration within 3.0 sec of initiating the deceleration phase.

b. Discussion

This maneuver is a stringent test of pitch attitude controllability and control power, engine/rotor RPM matching, and field-of-view. The ability to pitch to a large attitude in 1.5 sec is taken from a strawman specification developed by Sikorsky test pilots and flight test engineers. As shown in Figure 1, the AEFA pilots were able to achieve a pitch attitude change of approximately 30 deg in approximately 2 sec.

A particular pitch attitude or longitudinal acceleration is not specified because of concern that such a requirement would constitute a performance limit rather than a handling qualities limit. The intent of the paragraph is to insure desirable handling qualities throughout the region of achievable performance in the Operational Flight Envelope.

The maximum speed attained in the maneuver is not critical, and it was found that specifying a particular speed resulted in excessive concern with avoiding a speed overshoot. Therefore a minimum of 60 knots was specified to set the level of aggressiveness of the task. Likewise, altitude control is not intended to be a critical factor as long as excessive ballooning is not allowed during the initial deceleration, and the rotorcraft does not settle during the final deceleration. Considerable effort was expended trying to determine at what point on the aircraft the altitude sensor should be located. While no specific location is specified, there was general agreement that the appropriate altitude is that of the pilot's station.

a. Statement of Requirement

4.2.2 Rapid Sidestep. Starting from a stabilized hover, with the rotorcraft oriented 90 degrees to a reference line marked on the ground (or series of objects such as traffic cones, etc.), initiate a rapid and aggressive lateral translation at approximately constant heading up to a speed of between 30 and 45 knots. Maintain 30 to 45 knots for approximately 5 sec, followed by an aggressive lateral deceleration to hover. The maneuver is to be conducted at a constant altitude at or below 9.1 m (30 ft). Maintain the cockpit station over the reference line. The maneuver shall be performed in both directions.

- Desired Performance

- Maintain the cockpit station within  $\pm 3.3$  m (10 ft) of the ground reference line.
- Maintain altitude within  $\pm 3$  m (10 ft).
- Maintain heading within  $\pm 10$  degrees.
- Attain maximum achievable lateral acceleration within 1.5 sec of initiating the maneuver.
- Attain maximum achievable deceleration within 3.0 sec of initiating the deceleration phase.
- Stabilize within 1.5 sec of achieving hover. Heading tolerance is  $\pm 2$  degrees.

b. Discussion

This maneuver is patterned after the rapid acceleration and deceleration maneuver of Paragraph 4.2.1 except that it applies to the lateral axis. While heading control is not intended to be critical, a tolerance of  $\pm 10$  degrees is included to keep the nose pointed approximately perpendicular to the flight path. Time histories of the maneuver as performed on the UH-60 are given in Figure 1.

A pace car was used in the AEFA tests to assist in judging the lateral velocity, and the pilots noted some problems in matching the aircraft velocity with that of the pace car. Such velocity matching is not part of the task and should not be included in the task, or at least should be ignored when assigning a Cooper-Harper rating. It was noted that the final portion of the task was more demanding, and also more representative of a tactical maneuver. Unmasking requires less precision, as it is not done to a specified point.

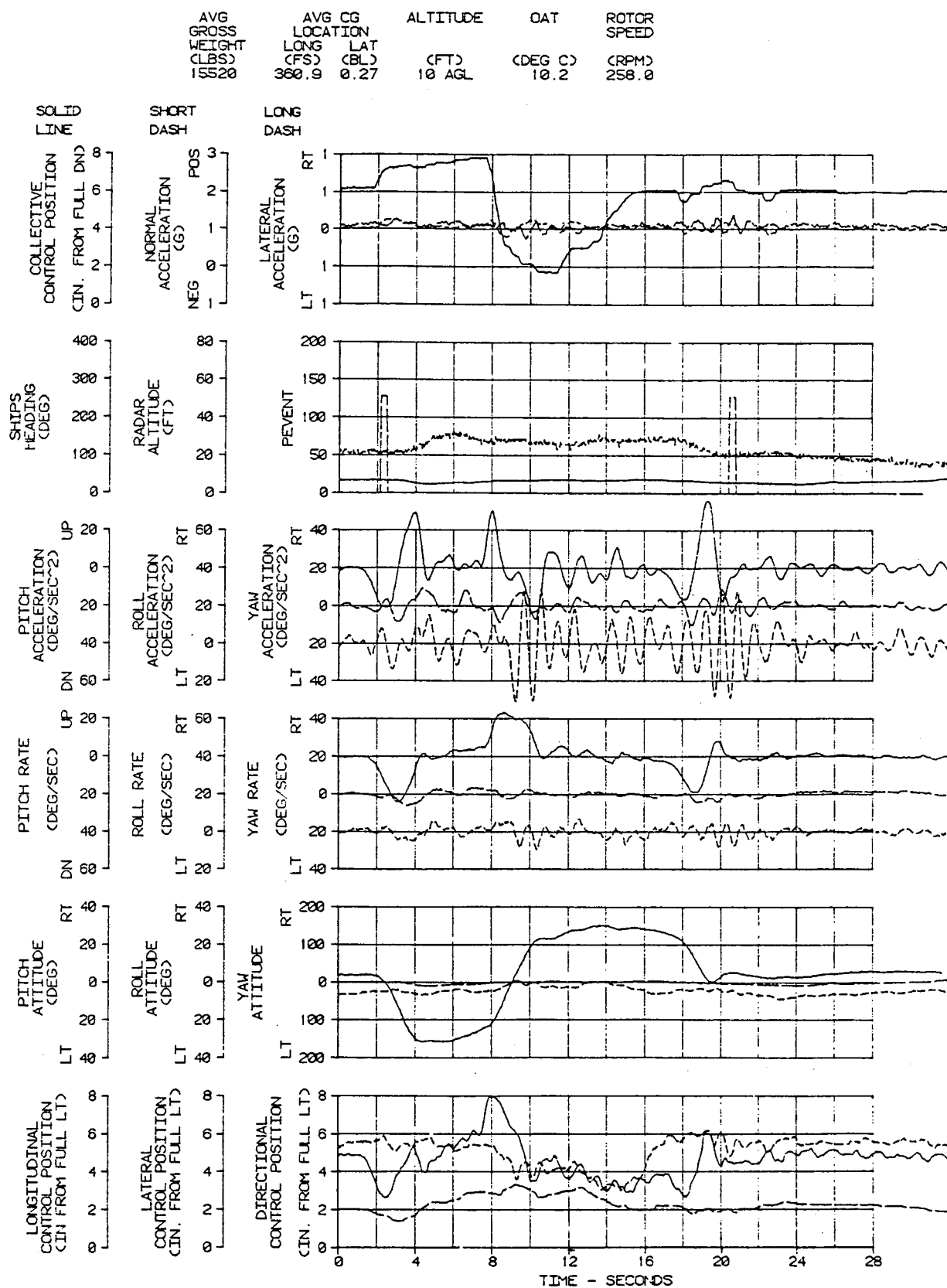


Figure 1(4.2.1). Accel/Decel

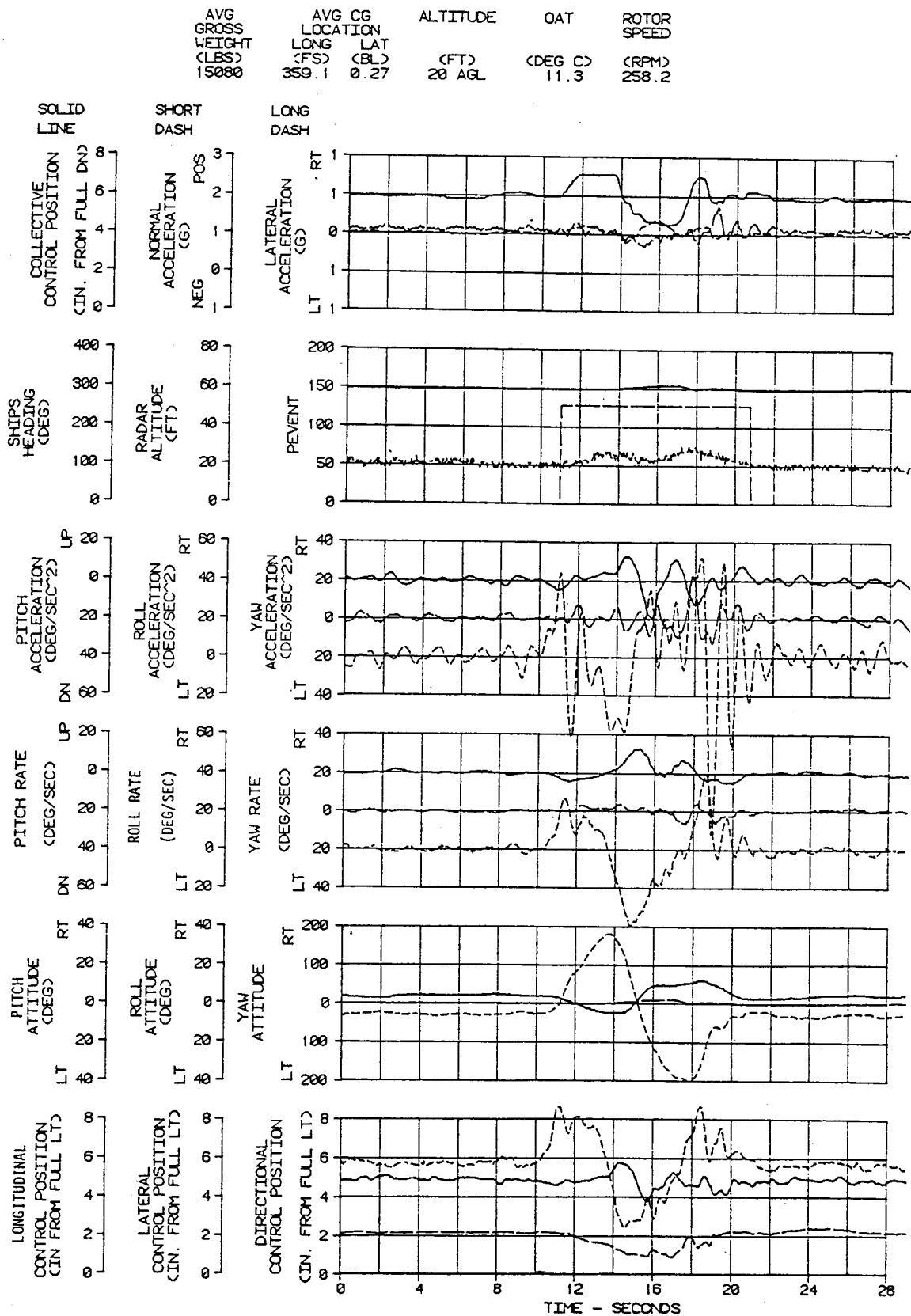


Figure 1(4.2.2). Right Lateral Sidestep



a. Statement of Requirement

4.2.3 Rapid Bob-up and Bob-down. From a stabilized hover at an altitude of 3 m (10 ft), bob-up to clear an obstacle approximately 7.6 m (25 ft) high to achieve a line-of-sight with a simulated threat. Simulate the attack using a fixed gun-sight. As soon as the target is stabilized in the sight, perform a descent to the initial hover position.

- Desired Performance
  - Maintain horizontal position so that the pilot station remains within 2.4 m (8 ft) of a reference point on the ground.
  - The overshoot at the top of the bob-up shall not exceed 1.5 m (5 ft) from the unmask altitude (defined by the line-of-sight).
  - The heading shall be within  $\pm 2$  degrees of the simulated target for at least 2 sec at the top of the bob-up.
  - Complete the maneuver in 8 sec or less.
  - Time at or above 7.6 m (25 ft) shall be no greater than 4 sec.

b. Discussion

This task has been included as a check on the vertical axis damping and control power. The maneuver was conducted behind a hangar during the AEFA tests to require a 15 ft vertical climb from a 10 ft hover for the pilot to achieve line-of-sight with a simulated threat (Figure 1). The pilot simulated the attack using a gun-sight drawn on the windshield. As soon as the pilot stabilized the target in the sight, a descent to the initial hover position was performed. As shown in Figure 1 the maneuver was performed in 8 - 9 sec. Since the heave axis handling qualities of the UH-60 meet the Level 1 requirements of the specification, 8 seconds was adopted as the standard for desired performance. While no Cooper-Harper ratings were given, the pilots indicated that the bob-up/bob-down was easy to accomplish.

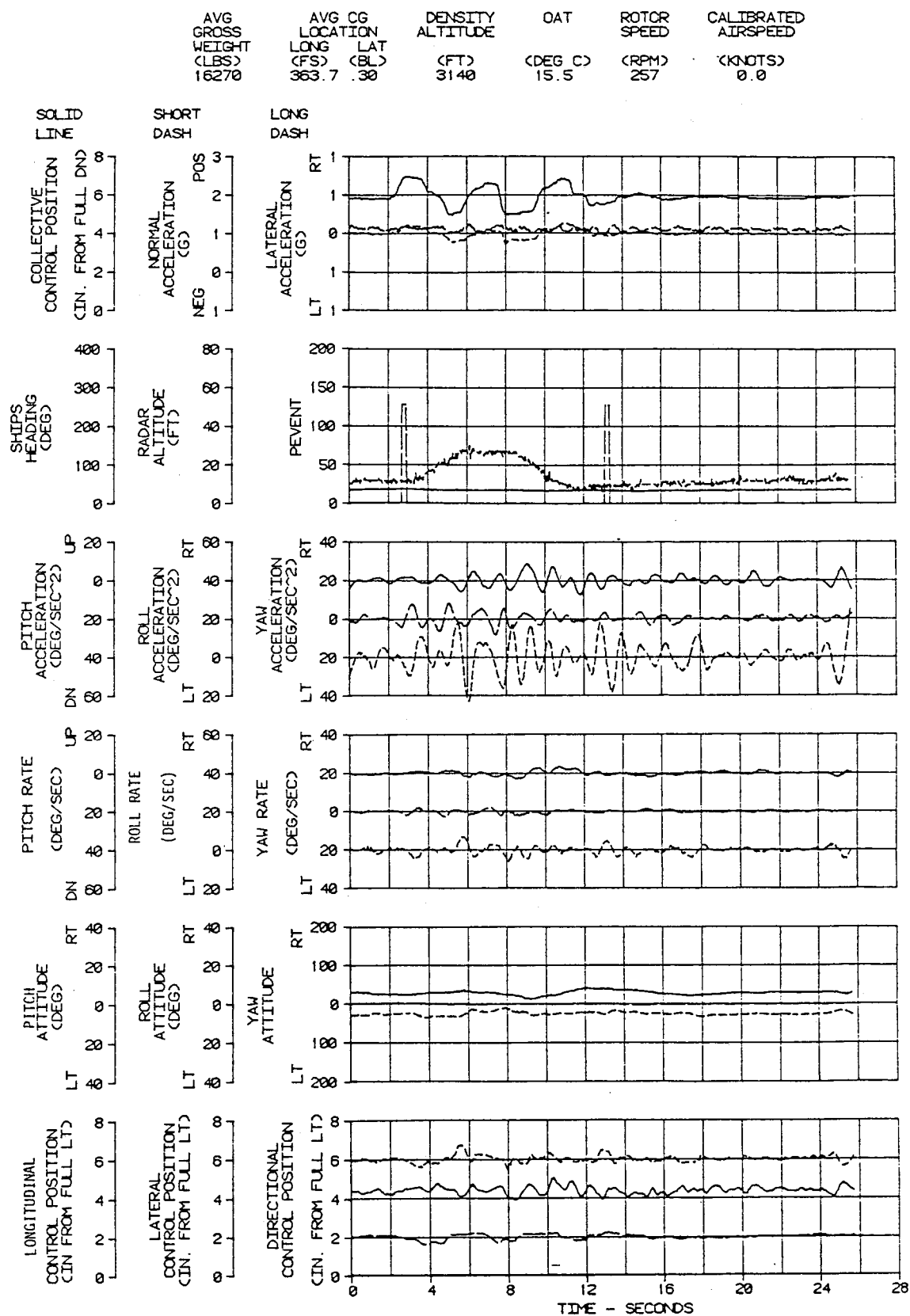


Figure 1(4.2.3). Bob-Up/Bob-Down From In-Ground-Effect Hover

a. Statement of Requirement

4.2.4 Pull-up/Push-over. From a level unaccelerated flight condition with maximum continuous power, attain a sustained positive load factor in a symmetrical pullup. Following this load factor buildup, maintain that load factor for at least 1 sec. Immediately transition via a symmetrical pushover to a sustained negative load factor, and sustain that load factor as required to achieve the airspeed at maneuver entry.

- Desired Performance

- Attain at least 2.0g within 1 sec from the initial control input.
- Transition from the positive 2g pullup to pushover of not greater than 0.0g shall be accomplished in less than 2 sec.
- Angular deviations in roll and yaw shall not exceed  $\pm 10$  degrees from the initial unaccelerated level flight condition to completion of the maneuver.

b. Discussion

This maneuver is the UTTAS maneuver (see References 94 and 140) and is essentially unchanged, except that a 2g (as opposed to 1.75g) load factor is required to be held for 1 sec (as opposed to 3 sec). The maneuver was performed with constant collective, which tends to reduce the variability in the way the maneuver is performed, an often-heard criticism in the past. The 2g load factor is held for only 1 sec to avoid excessive pitch attitudes during the maneuver.

When initiated at maximum continuous power, the maneuver resulted in a transient over-torque, and hence, the AEFA pilots recommended that it be conducted at a lower airspeed. However, it was decided that it is necessary to be able to conduct this maneuver from level flight at maximum continuous power to accomplish the mission objectives for an aggressive attack helicopter.

a. Statement of Requirement

4.2.5 Rapid Slalom. The maneuver is initiated in level unaccelerated flight, and in the direction of a line or series of objects on the ground. Maneuver rapidly to displace the aircraft 15.2 m (50 ft) laterally from the centerline and immediately reverse direction to displace the aircraft 15.2 m (50 ft) on the opposite side of the centerline. Return to the centerline as quickly as possible. Maintain a reference altitude below 15.2 m (50 ft). Accomplish the maneuver so that the initial turn is both to the right and to the left.

- Desired Performance
  - Maintain altitude within  $\pm 3$  m (10 ft).
  - Maintain airspeed at or above 60 knots.
  - Maximum bank angles should be at least 50 degrees.

b. Discussion

This maneuver was originally configured as a ground reference task which required setting up a series of pylons, spaced to require aggressive lateral maneuvering. However, the AEFA pilots indicated that simply maneuvering from one side of a 100 ft wide runway to the other in an aggressive fashion (bank angles of at least 50 degrees) represented a valid handling qualities test. As can be seen from the time histories in Figure 1, a bank angle of 65 deg was achieved with the UH-60A during the AEFA testing. The aggressiveness of the maneuver is established in the requirement by setting a bank angle limit of at least 50 deg for desirable performance. A transient over-torque of 113% (limit = 110%) occurred with the UH-60A during this maneuver. Such over-torques would be unacceptable for compliance with this specification.

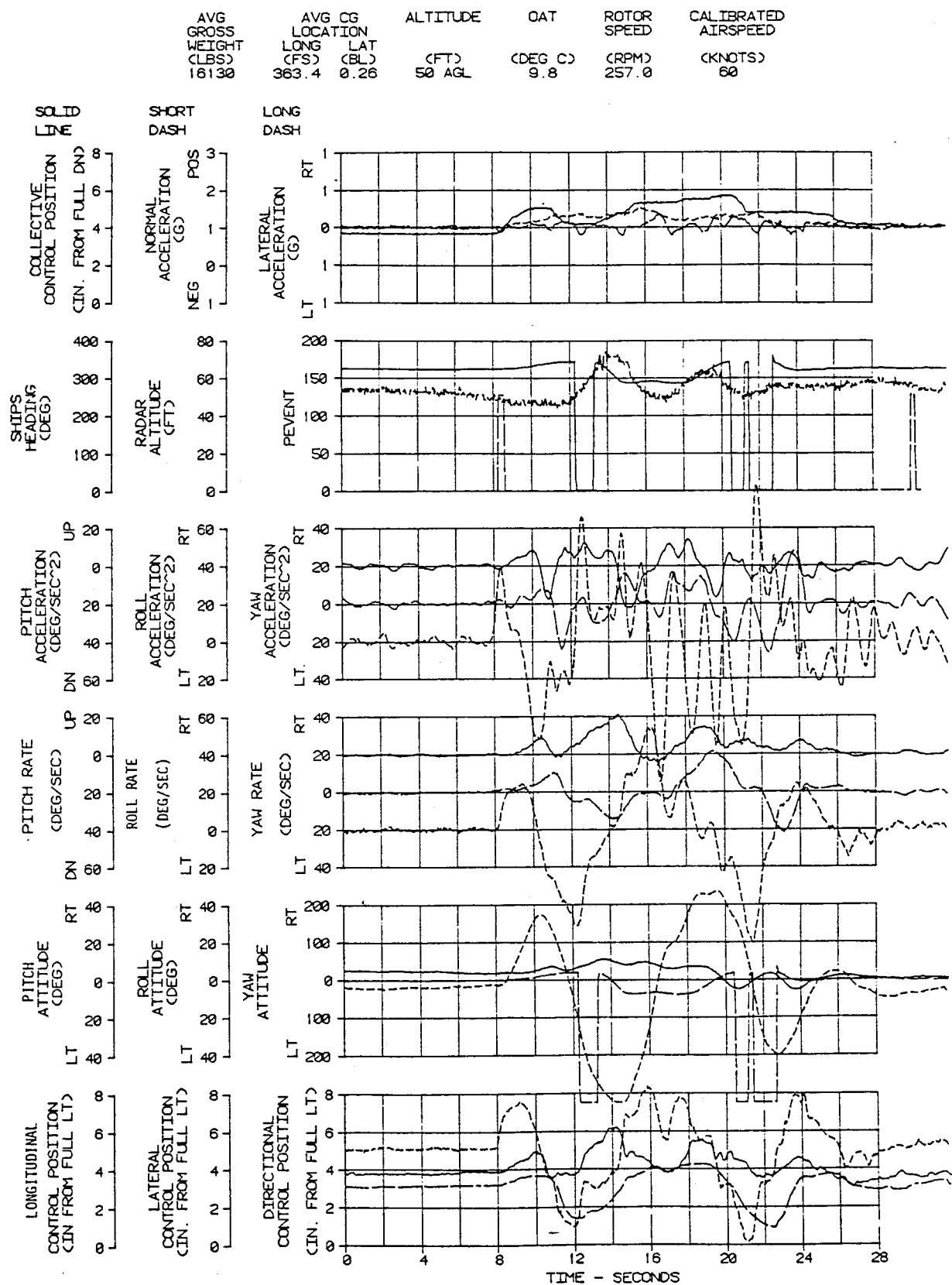


Figure 1(4.2.5). Rapid Slalom Over Runway

a. Statement of Requirement

4.2.6 Transient Turn. Starting at 120 knots and an altitude at or above 30.5 m (100 ft), accomplish a 180 degree heading change in as little time as possible. Use of pedals to induce a lateral acceleration in the direction of the turn is acceptable. Perform the maneuver both to the right and to the left.

- Desired Performance
  - Peak normal load factor to the limit ( $\pm 5$  percent) of the Operational Flight Envelope.
  - Complete 180 degree heading change in less than 10 sec.
  - Maintain altitude within 15.2 m (50 ft) of altitude at maneuver initiation.

b. Discussion

The transient turn maneuver was taken from the results of the U.S. Army Helicopter Air-to-Air-Combat Test Program (see Reference 94). As performed in that program, the decelerating turn maneuver involved peak transient load factors of 2.5 to 3.5g and deceleration rates of 15 to 20 knots per second. One transient turn with yaw, initiated from 120 knots, was clocked at less than five seconds for 180 deg of turn. Based on recommendations from the AEFA flight test personnel, a somewhat less aggressive maneuver is required, i.e., complete the 180 deg turn in less than 10 sec; 17 sec were required in the UH-60A tests, see Figure 1. The maneuver, as performed at AEFA, was coordinated although the use of sideslip to increase the turn rate is specifically allowed. It helps to put these tests into perspective if one notes that the UH-60 tail rotor had to be replaced at the completion of the air combat testing. The rotorcraft required by this specification will be more maneuverable than current models.

It was suggested by the AEFA pilots that the maneuver is more meaningful when conducted at an altitude of 100 ft, as opposed to up-and-away.

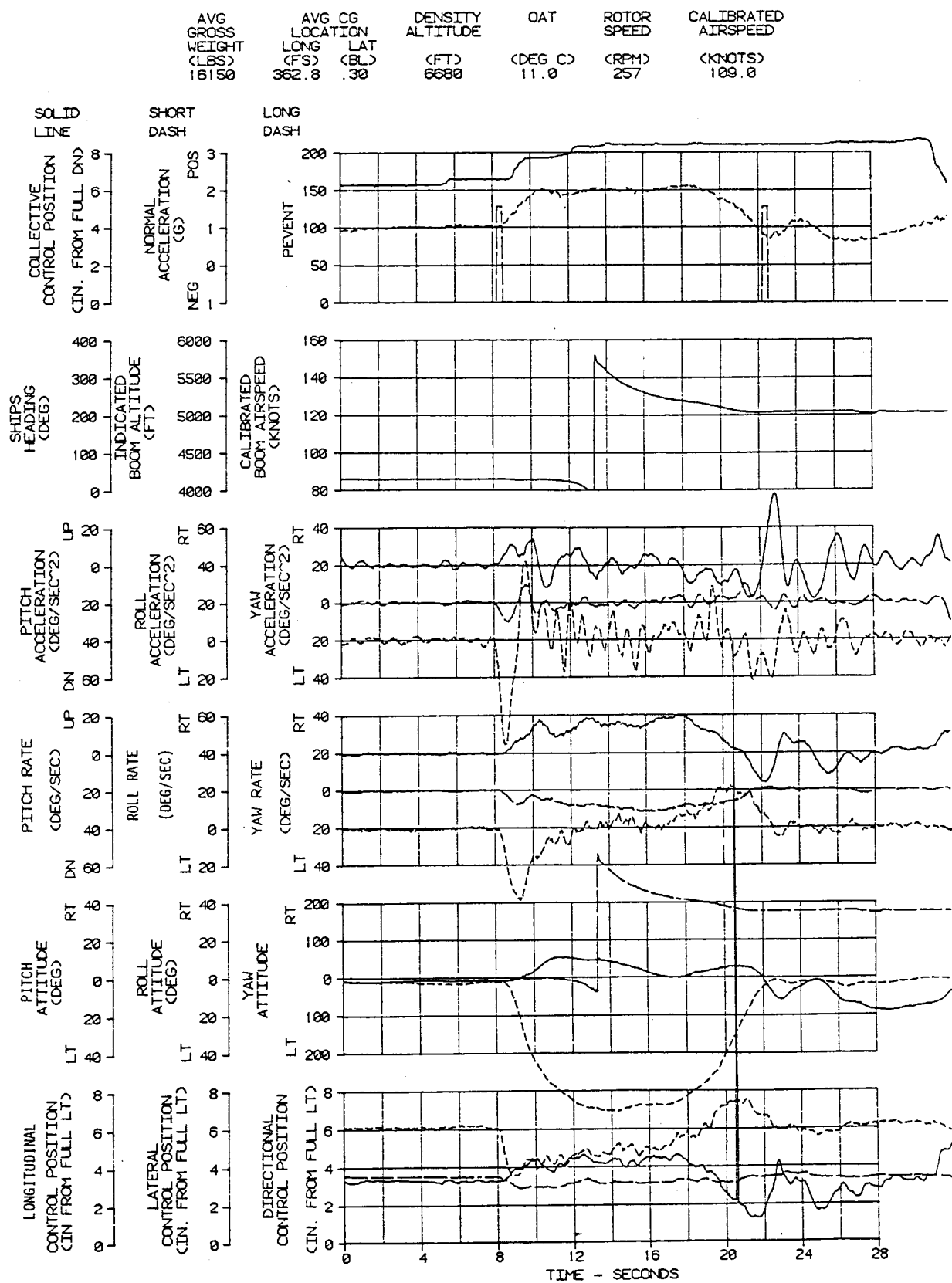


Figure 1(4.2.6). Transient Left 180 Deg Level Turn

a. Statement of Requirement

4.2.7 Roll Reversal at Reduced and Elevated Load Factors. From level flight at maximum continuous power, enter a pull-up (push-over) to achieve normal accelerations representative of the positive and negative boundaries of the Operational Flight Envelope. Immediately execute a rapid roll to approximately 45 degree bank and back to level, while maintaining the load factor essentially constant. Rolls are to be performed to the left and to the right.

- Desired Performance
  - Peak roll rate of at least 50 percent of the maximum achievable steady state roll rate.
  - No undesirable oscillation in any axis.
  - No sudden change in roll control power during maneuver.

b. Discussion

This maneuver was derived from the U.S. Army Air-to-Air Combat Tests, and was modified based on testing at AEFA. The maneuver as reported in Reference 94 involved a rapid roll reversal. This was modified to a rapid roll from level to about 45 deg and back to level at varying levels of normal acceleration. This is a particularly desirable maneuver as it tests the handling qualities at the limits of the Operational Flight Envelopes defined by load factor. The test requires some practice to master as the rapid roll must be superimposed on a pull-up or push-over. Both AEFA pilots found that they could satisfactorily perform the maneuver after a few practice runs.

AEFA test pilots suggested that the handling qualities were adequately tested at roll rates substantially less than the maximum achievable. On this basis, only 50% of the maximum achievable roll rate is required in the maneuver. As a matter of interest, the UH-60A exhibited undesirable yaw oscillations ("fishtailing") at 0g and significantly reduced lateral cyclic sensitivity at 2g. The requirements for "no undesirable oscillations" and "no sudden change in roll control power" derive from that experience.



a. Statement of Requirement

4.3 DECELERATING APPROACH TO HOVER

Starting on a 4 degree glide slope at 100 knots, perform a manual deceleration to an airspeed of 25 knots at an altitude of 15.2 m (50 ft). Guidance commands may be generated using onboard sensors, or from ground-based transmitters.

- Desired Performance
  - Glide slope errors not to exceed  $\pm 3.8$  m (12.5 ft).
  - Localizer errors not to exceed  $\pm 15.2$  m (50 ft).
  - Airspeed error within  $\pm 5$  knots of commanded value.

b. Discussion

The decelerating approach is an extremely demanding instrument task, especially during the final segment of the approach; see the discussion for Paragraph 3.2.2. This task was included under the assumption that if it can be accomplished with Level 1 pilot ratings, all other instrument tasks will also be Level 1. The parameters defining desirable performance were taken from flight tests conducted at the Canadian National Research Council (NRC) with the variable stability Bell 205A (see References 38 and 194).

Deceleration on the glideslope in IMC is a required Mission-Task-Element anytime the mission description involves forward area operations (landing for resupply or transition to NOE) in very low visibility and ceilings. It is assumed that approach guidance will consist of portable microwave landing transmitters (already in existence), autonomous navigation systems (INS, GPS, Loran C, Omega, etc.), in combination with night vision goggles and/or FLIR.

The decelerating IMC approach to hover was not accomplished during the AEFA tests due to unavailability of the necessary avionics on the test aircraft.

a. Statement of Requirements

4.4 PRECISION TASKS IN DEGRADED VISUAL ENVIRONMENT

The following precision maneuvers shall be flown in the Degraded Visual Environments (DVE) specified in Paragraph 3.1.1, and using the displays and vision aids that will be available to the pilot.

The maneuvers shall be accomplished in sequence on a defined test course. Each maneuver shall be separately rated after at least three trial runs through the entire course. The wind conditions may be calm, but it would be desirable to demonstrate the maneuvers in stronger winds.

4.4.1 Hover. Maintain a steady hover at an altitude of not more than 6.1 m (20 ft) above the ground.

- Desired Performance
  - Maintain horizontal position of the pilot station within 0.9 m (3 ft) of a reference point on the ground.
  - Maintain altitude within  $\pm 0.91$  m (3 ft).
  - Maintain heading within  $\pm 5$  degrees.
  - There shall be no objectionable oscillations in attitude or position.

4.4.2 Hovering Turn. From a steady hover at an altitude of not greater than 6.1 m (20 ft), complete a 180 degree turn as rapidly as possible, in both directions.

- Desired Performance
  - Maintain horizontal position of the pilot's station within 1.8 m (6 ft) of a reference point on the ground.
  - Maintain altitude within  $\pm 0.91$  m (3 ft).
  - Maintain heading within  $\pm 5$  degrees.
  - Complete hover turn in less than 10 sec.

4.4.3 Vertical landing. Perform a vertical landing from a stabilized hover.

- Desired Performance
  - Maintain horizontal position with respect to the pilot's station within 0.9 m (3 ft) of a

defined reference point on the ground.

- Maintain heading within  $\pm 5$  degrees.
- The vertical descent should be reasonably continuous with minimal need to arrest the sink-rate to eliminate horizontal drifting. There shall be no perceptible drift at touchdown.
- The time from maneuver initiation to touchdown and stopped shall be less than 10 sec.

4.4.4 Pirouette. Initiate the maneuver from a stabilized hover over a point on the circumference of a 30.5 m (100 ft) radius circle, marked on the ground, with the nose of the rotorcraft pointed at a reference point at the center of the circle, and at a hover altitude of approximately 3 m (10 ft). Accomplish a lateral translation around the circle, keeping the nose of the rotorcraft pointed at the center of the circle, and the circumference of the circle under the pilot station. Perform the maneuver in both directions.

- Desired Performance
  - Maintain the circumference of the circle within  $\pm 3.0$  m (10 ft) of the pilot's station.
  - Maintain altitude within  $\pm 1.2$  m (4 ft).
  - Maintain heading so that the nose of the rotorcraft points at the center reference point within  $\pm 10$  degrees.
  - The maneuver shall be completed in less than 60 sec, starting from the initial control input, and terminating in a stabilized hover.
  - The lateral velocity should be smooth and controlled throughout the maneuver. Note: the nominal lateral velocity will be approximately 6 knots.

b. Discussion

The intent of this section is to verify that the combination of Response-Type and vision aids/displays produces Level 1 flying qualities in representative worst-case conditions that are expected to occur in the operational environment. Specifically, the tests are to be performed in the Degraded Visual Environments (DVEs) specified in Paragraph 3.1.1. These DVEs will generally be tailored to investigate the worst-case conditions expected in operational use. For example, the following DVEs have been recommended for the LHX.

1. Moonless night, overcast with a defined ceiling.
2. Night in rain and/or fog, following a day with not more than two hours of sunshine.

Night is defined as two hours after sunset to two hours before sunrise, and the maneuvers are to be flown in a flat open area such as a grass field, or an airport, and in an area which is clear of artificial lighting. Note that the first of the above DVEs will tend to degrade the performance of light intensification devices, such as Night Vision Goggles (NVGs), and the second will have a deleterious effect on Forward Looking Infrared (FLIR) devices. Both vision devices will typically suffer from a lack of fine-grained texture when operating over a flat open area such as a grass field.

The philosophy of these tests is to operate in the real environment, as opposed to simulation devices, such as daylight training filters on NVGs. It is felt that such devices may not accurately reproduce the available cueing, which could cause misleading results.

The maneuvers have been adapted directly from Section 4.1, with changes in the desired performance tolerances to allow for the inherent reduction in aggressiveness that accompanies operation in conditions of degraded visual cueing. A comparison of the desired performance for the maneuvers required in good visual cueing (GVE), and in the degraded visual environment (DVE), is given in Table 1 for precision tasks. The maneuvers are not discussed individually, as this is done in Section 4.1.

TABLE 1(4.4). COMPARISON OF DESIRED PERFORMANCE IN GOOD VISUAL CUEING  
VS. DEGRADED VISUAL CUEING FOR PRECISION MANEUVERS

MANEUVER	Position Tolerance (ft)		Altitude Tolerance (ft)		Heading Tolerance (deg)		Time to Complete (sec)		Required in Winds	
	GVE	DVE	GVE	DVE	GVE	DVE	GVE	DVE	GVE	DVE
Hover	3	3	±2	±3	±5	±5	NA	NA	Yes	No
Hover Turn	6	6	±3	±3	±3	±5	5	10	Yes	No
Vertical Landing	3	3	NA	NA	±5	±5	6	10	No	No
Pirouette	10	10	±3	±4	±10	±10	45	60	No	No
Slope Landing	3	NA	NA	NA	±5	NA	—	—	No	—

GVE = Good Visual Environment

DVE = Degraded Visual Environment

NA = Not Applicable

a. Statement of Requirements

4.5 MODERATELY AGGRESSIVE TASKS IN DEGRADED VISUAL ENVIRONMENT

The following aggressive maneuvers shall be flown in the Degraded Visual Environment (DVE) specified in Paragraph 3.1.1, and using the displays and vision aids which will be available to the pilot.

The maneuvers shall be accomplished in sequence on a defined test course. Each maneuver shall be separately rated after at least three trial runs through the entire course. The wind conditions may be calm, but it would be desirable to demonstrate the maneuvers in stronger winds.

4.5.1 Acceleration and Deceleration. Starting from a hover over a defined point, accelerate to a ground speed of approximately 50 knots, and immediately decelerate to hover over a defined point. Maintain a constant altitude at or below 12.1 m (40 ft).

- Desired Performance
  - Complete the maneuver over a defined reference point. Tolerance is plus zero and minus 3 m (10 ft) (positive forward).
  - The pitch attitude used for acceleration shall be at least 12 degrees nose-down, and for deceleration at least 15 degrees nose-up.
  - Maintain altitude within  $\pm 4.6$  m (15 ft).
  - Maintain heading within  $\pm 10$  degrees.
  - The transition between deceleration and hover shall occur smoothly, and without uncommanded transients in position or attitude.

4.5.2 Sidestep. Starting from a stabilized hover, with the rotorcraft oriented 90 degrees to a reference line marked on the ground (or series of objects such as traffic cones, etc.), initial a lateral translation at approximately constant heading up to a speed of at least 17 knots. Maintain constant speed for approximately 5 sec, followed by a lateral deceleration to hover. The maneuver is to be conducted at a constant altitude at or below 9.1 m (30 ft). Maintain the cockpit station over the reference line. The maneuver shall be performed in both directions.

- Desired Performance
  - Maintain the cockpit station within  $\pm 3$  m (10 ft) of the ground reference line.
  - Maintain altitude within  $\pm 3$  m (10 ft).

- Maintain the reference heading within  $\pm 10$  degrees.
- The peak bank angle during acceleration and deceleration shall be at least 20 degrees.
- The transitions between sideward flight and hover shall occur without noticeable oscillations in rotorcraft attitude and position.

4.5.3 Bob-up and Bob-down. From a stabilized hover at an altitude of 3 m (10 ft), bob-up to clear an obstacle approximately 7.6 m (25 ft) high to achieve a line-of-sight with a simulated threat. Simulate the attack using a fixed gun-sight. As soon as the target is stabilized in the sight, perform a descent to the initial hover position.

- Desired Performance

- Maintain horizontal position so that the pilot station remains within 3.7 m (12 ft) of a reference point on the ground.
- The overshoot at the top of the bob-up shall not exceed 1.5 m (5 ft) from the unmask altitude (defined by the line-of-sight).
- The heading shall be within  $\pm 2$  degrees of the simulated target for at least 2 sec at the top of the bob-up.
- Complete the maneuver in 15 sec or less.
- Time at or above 7.6 m (25 ft) shall be no greater than 6 sec.

4.5.4 Slalom. The maneuver is initiated in level unaccelerated flight, and in the direction of a line or series of objects on the ground. Maneuver rapidly to displace the aircraft 15.2 m (50 ft) laterally from the centerline, and immediately reverse direction to displace the aircraft 15.2 m (50 ft) on the opposite side of the centerline. Return to the centerline as quickly as possible. Maintain a reference altitude below 15.2 m (50 ft). Accomplish the maneuver so that the initial turn is both to the right and to the left.

- Desired Performance

- Maintain altitude within  $\pm 4.6$  m (15 ft).
- Maintain airspeed at or above 30 knots.
- Maximum bank angles should be at least 25 degrees.

b. Discussion

The discussion for Section 4.4 applies to Section 4.5. A comparison of the required performance in good and degraded visual cueing for aggressive maneuvers is given in Table 1.

TABLE 1(4.5). COMPARISON OF DESIRED PERFORMANCE IN GOOD VISUAL CUEING VS. DEGRADED VISUAL CUEING FOR AGGRESSIVE MANEUVERS

MANEUVER	Position Tolerance (ft)		Altitude Tolerance (ft)		Heading Tolerance (deg)		Time to Complete (sec)		Maximum Pitch/Bank Angle (deg)	
	GVE	DVE	GVE	DVE	GVE	DVE	GVE	DVE	GVE	DVE
Accel/Decel	+0	+0	±15	±15	±10	±15	1.5	NA	—	+12
	-10	-10					3.0			-15
Sidestep	±10	±10	±10	±10	±10	±10	1.5	NA	—	20
							3.0			
Bob-Up/Down	8	12	5	5	± 2	± 2	8	15	—	—
							4	6		
Slalom	NA	NA	10	15	NA	NA	NA	NA	50	25

GVE = Good Visual Environment

DVE = Degraded Visual Environment

NA = Not Applicable



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## APPENDIX A

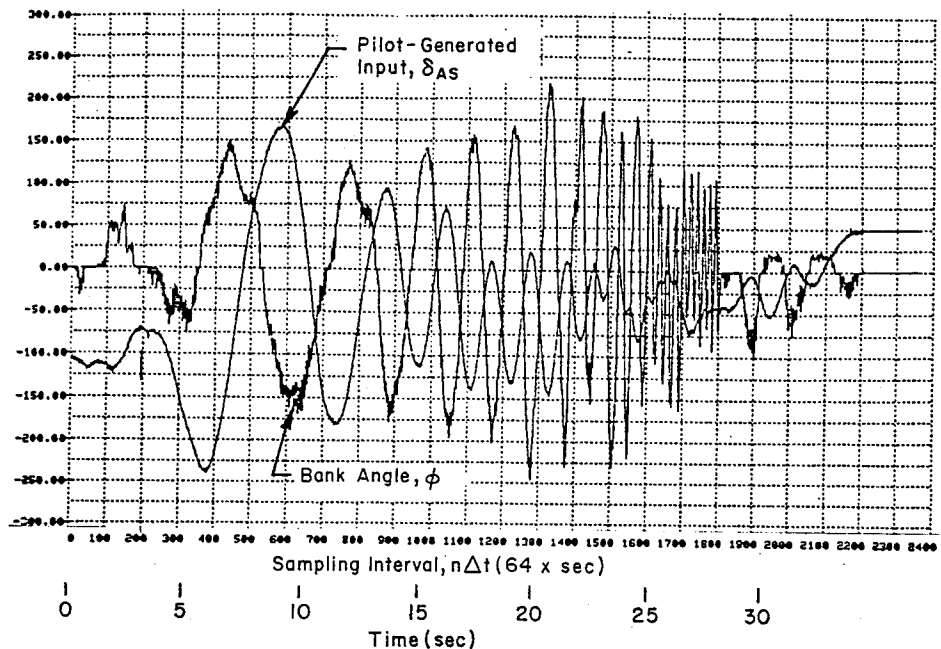
### DETERMINATION OF BANDWIDTH FROM FLIGHT DATA

#### 1. Obtaining the Frequency Response (Bode Plot)

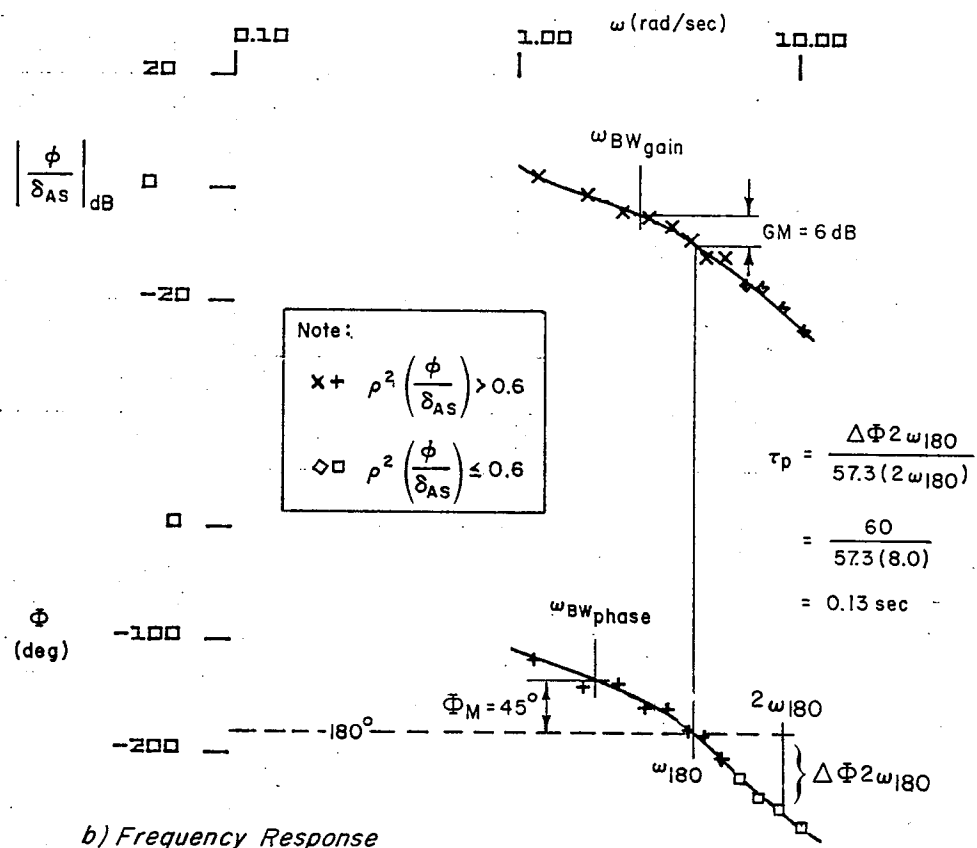
Since compliance demonstration for many of the requirements in the specification requires determination of bandwidth and phase slope from flight data, some method must be used to obtain frequency responses such as that sketched in Figure 2(3.3). The most common method involves the use of control-input "frequency sweeps," in which sinusoidal control inputs are made with each cycle at a slightly higher frequency than the previous cycle -- thus a "sweep" of frequencies is performed. Recorded input and response signals (e.g., pitch attitude for longitudinal cyclic, bank angle for lateral cyclic, etc.) are then reduced via fast-Fourier transform (FFT) algorithms to frequency response data. The forcing command can be provided by the pilot with only a minimum of practice. This technique has been used successfully to identify the dynamics of the XV-15 (Reference 100), as well as conventional airplanes (Reference 101), single-rotor helicopters (Reference 117), and unconventional aircraft (Reference 118).

Figure A-1 shows an example of the time and frequency-response data obtained from a "frequency sweep" for a single-rotor helicopter in hover. In Figure A-1a the time histories of the input command and bank angle response are plotted as functions of sampling interval,  $n\Delta t$ , where  $n = 64$  samples per second. This data was FFT'd to produce the Bode plot of Figure A-1b. The X and + symbols represent data at discrete frequencies with a relatively high level of input/output correlation. The . and . data represent somewhat poorer correlation. Values of  $\omega_{BW}$  and  $\tau_p$  can be read directly from the plot. The frequency response shows a consistent trend from about 1.2 to 13 rad/sec; this corresponds roughly to the frequencies of the command at the beginning and end of the sweep, Figure A-1a, and is typically a sufficiently wide frequency range for determining bandwidths. The quality and frequency range of the Figure A-1b data could be improved if necessary, by including more low-frequency inputs and by averaging the data from this example with data from at least one more sweep.

The procedures involved in obtaining pilot-generated frequency-sweep data, and in reducing this data to frequency-response form, have been refined over the years, see for example References 100, 117, 118, 119, and 171. Table A-1 summarizes the key points to be considered in these procedures.



a) Time History of Command and Response



b) Frequency Response

Figure A-1. Example of Determination of  $\omega_{BW}$  and  $\tau_p$  from Frequency Sweep Data (NRC of Canada Variable Stability Helicopter; Hover; Rate Response-Type)

TABLE A-1. GENERAL RULES FOR OBTAINING FREQUENCY RESPONSES  
FROM FLIGHT DATA\*

PILOT-GENERATED SWEEPS

- Initial and final conditions must be in steady trim, and several seconds of this trim must be included in recorded data. The frequency sweep is considered to be a "transient" in the FFT analysis.
- Input quasi-sinusoidal waves starting at a period at least as long as the lowest frequency desired for the frequency response (typically on the order of 20-30 seconds), and terminating at a period for the highest frequency (typically about 1/2 second per cycle).
- Each cycle of the sweep should be at about half the period of the previous cycle.
- Magnitude of the input can be adjusted to keep attitudes and translations small and close to trim values.
- Input form (shape) should be adjusted during the run to avoid large asymmetrical response; however, large shifts in either the reference control position or the reference aircraft state degrade the quality of the data. The largest attitude excursions occur during the low-frequency inputs at the beginning of the runs; these excursions generally do not exceed  $\pm 10$  degrees (pitch or roll). The maximum angular rates generally do not exceed  $\pm 20$  deg/sec, and the maximum stick deflections generally do not exceed  $\pm 20$  percent.
- Off-axis regulation:
  - Hover. Lateral stick inputs may cause significant coupling to the directional axis. Heading excursions should be reduced (to roughly  $\pm 20$  deg) with directional control inputs. This should be considered a low-frequency and low-priority piloting function. Similarly, power inputs should be used to avoid large height excursions during longitudinal sweeps.
  - Forward flight. Lateral stick inputs will cause sideslip excursions. Pedals should not be used during lateral sweeps unless sideslip operational limitations are encountered. It is preferable to reduce the magnitude of the lateral inputs if sideslip excursions are too great, rather than to use large pedal inputs.

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\*Compiled from References 117, 118, and 119.

TABLE A-1. (Continued)

- Cockpit control position indicators are very useful for aiding the pilot in maintaining symmetrical wave forms.
- "Coaching" by a flight test engineer looking at the on-line data may help the pilot to perform the proper sweeps at proper input periods.
- Previous experience suggests that the following order of inputs is useful for pilot training, and reflects an increasing degree of difficulty:
  - 1) Yaw
  - 2) Heave
  - 3) Pitch
  - 4) Roll
- The ground reference makes hover tests useful as a first flight condition to aid the pilot in learning the test technique.
- Each sweep should be performed at least twice to improve the quality of the reduced data.

#### PILOTING CONSIDERATIONS

- Recommend that the pilot be "coached" during the input. It is very easy to remain at one frequency too long; having the engineer tell the pilot to dwell on a specific frequency longer or increase frequency during a data run aids data acquisition. This assumes the engineer has real time data.
- Recommend that the copilot or flight test engineer coach the pilot for the low frequency responses by counting seconds for timing the quarter periods. This should only be done for the lowest frequencies. It was found that if the copilot tried counting at the higher frequency it only mixed up the pilot and resulted in the pilot following the copilot's counting rather than increasing the frequency as required by the test.
- Recommend that pilots practice the inputs utilizing external power supply (hydraulic and electrical power) on the ground prior to testing. This should significantly reduce test flight time.
- Recommend a "scope" with "freeze" capability be installed in the test aircraft unless "real time" telemetry is available.

TABLE A-1. (Concluded)

- Recommend that the test team be briefed on all "aircraft" natural frequencies below 4.0 Hz and any resultant problems which may be encountered at these frequencies.
- During flights that require longitudinal inputs the instrumentated airspeed boom must be monitored closely for deflection beyond limits or removed.

#### DATA REDUCTION

- Following are only rules of thumb; the reader is referred to References 117, 118, and 119 for more details.
- It is important to include both pre- and post-sweep trim period in data analyzed.
- Input and output signals should be filtered through an anti-aliasing filter to remove digital artifacts.
- Besides the response transfer function (magnitude and phase), the power spectrum of the input ( $\Phi_{xx}$ ) and output/input coherency function [ $\rho^2(y/x)$ ] should be determined. The former gives an indication of the power supplied by the pilot, and the latter reflects the degree of correlation (from 0 to 1) between the response output and control input. High values of both are desired; typically,  $\rho^2(y/x)$  should be greater than about 0.6-0.8.

## 2. Measurement of Bandwidth

The following two examples, for Rate and Attitude Response-Types, show the methods involved in determining bandwidth frequency.

### a. Example for Rate Response-Types

Figure A-2 shows the frequency and time (step) responses for two Rate Response-Types. These two systems have identical forms, i.e.,

$$\frac{q}{q_c} = \frac{K_q (s + 0.75) e^{-\tau s}}{[s^2 + 2(0.35)(\omega_n) s + (\omega_n)^2]}$$

Two values of time delay,  $\tau = 0.1$  and  $0.3$  sec, were used, and the natural frequency,  $\omega_n$ , adjusted to keep the phase bandwidth constant at 2 rad/sec. Thus both systems have the same phase bandwidth, but, as Figure A-2a shows, differing gain bandwidths.

From Figure A-2a, for the low-time-delay case  $\omega_{BW\text{gain}} > \omega_{BW\text{phase}}$ , so that  $\omega_{BW} = \omega_{BW\text{phase}} = 2$  rad/sec. For the high-delay case, however,  $\omega_{BW} = \omega_{BW\text{gain}} = 0.4$  rad/sec. Thus, for Rate Response-Types,  $\omega_{BW\text{gain}}$  is defined as the lowest frequency at which 6 dB of gain margin exists. Typically, "gain-margin-limited" systems such as the example in Figure A-2 only occur for large values of delay.

### b. Example for Attitude Response-Types

Figure A-3 shows two example Attitude Response-Types with equal phase bandwidths at 2 rad/sec. These systems have the following form:

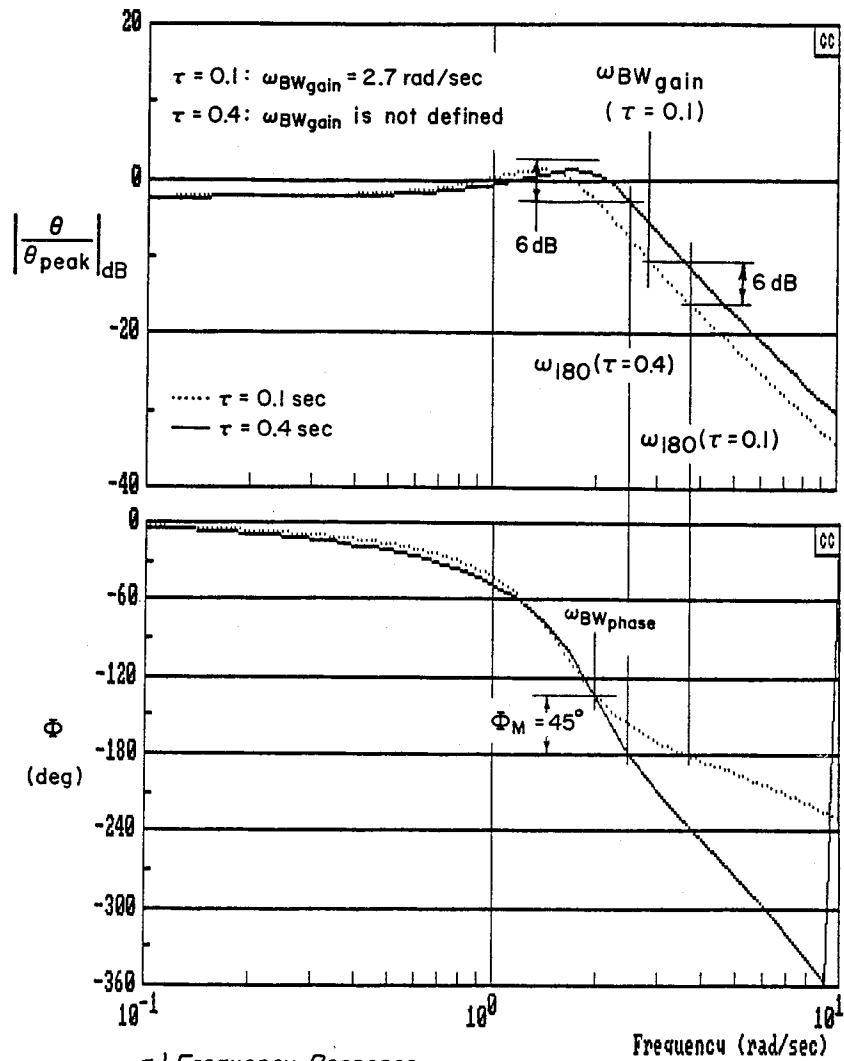
$$\frac{\theta}{\theta_c} = \frac{K_\theta e^{-\tau s}}{[s^2 + 2(0.35)(\omega_n) s + (\omega_n)^2]}$$

As with the Rate Response-Types, two values of time delay are used, and  $\omega_n$  varied to keep  $\omega_{BW\text{phase}} = 2$  rad/sec.

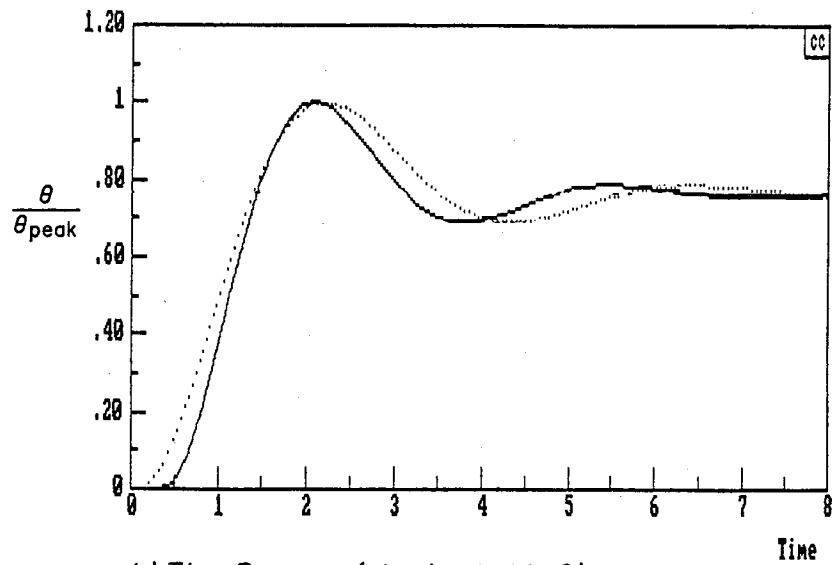
For the low-time-delay case, Figure A-3a shows  $\omega_{BW\text{gain}} > \omega_{BW\text{phase}}$ . With high time delay, however, the "gain margin bandwidth" corresponding to 6 dB of gain margin is indeterminate. In Figure 1(3.3), the bandwidth limits for Attitude Response-Types are based on  $\omega_{BW\text{phase}}$  only.

Attitude Response-Types with indeterminate gain margins such as that sketched in Figure A-3 are not uncommon for helicopters with large





a) Frequency Response



b) Time Response (step input at  $t = 0$ )

Figure A-3. Example of Two Attitude Response-Types with  $\zeta = 0.35$  and  $\omega_{\text{BWphase}} = 2.0$  rad/sec



rotor lags. For example, most of the Attitude Response-Type configurations flown in the Reference 157 flight research program had very low gain margins. The physical significance of a low or undefined gain margin for Attitude Response-Types has been the subject of discussion throughout the development of the specification. The fundamental philosophy behind the bandwidth criterion (e.g., References 1, 2, and 3) has assumed that low gain margin results in inadequate handling qualities for the closed-loop, pilot/vehicle system. This appears to be less severe for Attitude Response-Types (see discussion in Paragraph 3.3.2.1).

An analysis of the closed-loop pilot/vehicle system for Rate versus Attitude Response-Types suggests that the pilot's approach to these two systems is fundamentally different: for a Rate Response-Type, the pilot is continuously making inputs, keeping his gain very high all the time. For an Attitude Response-Type, the inputs tend to be less continuous, since the pilot can relax somewhat and allow the inherent stability of the aircraft assist him. In addition, if a pitch bobble or PIO begins to develop, the pilot can "back out" if the loop with an attitude system with only minor excursions in hover position. With a Rate system this is not possible and the pilot is forced to choose between a PIO and a larger excursion in position.

### 3. Measurement of Phase Delay

The specification defines phase delay,  $\tau_p$ , using a two-point approximation to the shape of the phase curve above the instability frequency; from Figure 2(3.3),

$$\tau_p = \frac{\Delta\phi_{2\omega_{180}}}{57.3 (2\omega_{180})}$$

This definition implicitly assumes that the phase curve is relatively uniform between the frequencies  $\omega_{180}$  and  $2\omega_{180}$ , so that  $\tau_p$  represents the slope of the phase curve in this frequency range (note that  $e^{-\tau s}$  is linear on a linear scale, i.e.,  $\phi = \tau\omega$ ). In a few cases it has been found that the frequency response data do not produce a linear phase curve at the higher frequencies, and this is especially true for single-rotor helicopters with vibrational modes around the frequencies of interest.

An example of such a frequency response phase curve is shown in Figure A-4 (from Reference 171). The curve is plotted on a linear frequency scale to highlight the nonlinearity that occurs at approximately 12 rad/sec. For this helicopter (the Bell 214-ST), a rotor/pylon mode at 11.9 rad/sec causes a dip in the phase response, and results in low-coherency frequency-response data above this point (i.e., the confidence in the data above 11.9 rad/sec is low). If this dip had occurred at precisely  $2\omega_{180}$ , the value of  $\tau_p$  would have been artificially large. On the other hand, in some cases such a "dip" may be indicative of actual handling qualities problems if it results in a significant phase drop over a significantly large frequency range.

Consideration was given to requiring that phase delay be measured from a linear fit to the phase data at frequencies between  $\omega_{180}$  and  $2\omega_{180}$ , thereby minimizing the effects of small irregularities in the phase response. Such a requirement would, however, involve specification of many parameters, including minimum number of data points required, correlation coefficient, etc. In addition, several methods are available in the data reduction process itself (such as "binning" frequency ranges, etc.) to reduce irregularities that are not due to aircraft physical characteristics, and some credit should be given for these analysis methods. The frequency response of Figure A-1 shows the phase curve for a helicopter where several smoothing techniques have been applied, resulting in a few discrete data points (symbols on Figure A-1b) that represent a smooth phase rolloff. For the example in Figure A-1 it is clearly sufficient to use the two-point definition of  $\tau_p$ .

The approach taken in the specification is to require that, if phase is nonlinear between  $\omega_{180}$  and  $2\omega_{180}$ ,  $\tau_p$  be determined from a linear least squares fit to the phase curve between  $\omega_{180}$  and  $2\omega_{180}$ , with the details of this fit left to the user. In applying this requirement, the first logical step is to plot the phase curve on a linear scale as in Figure A-4, emphasizing the frequency region between  $\omega_{180}$  and  $2\omega_{180}$ , and determine, by subjective judgement, if the curve is linear. If there is any doubt whatsoever, or if precision is desired, a least-squares curve fit should be performed and  $\tau_p$  calculated from the curve fit.

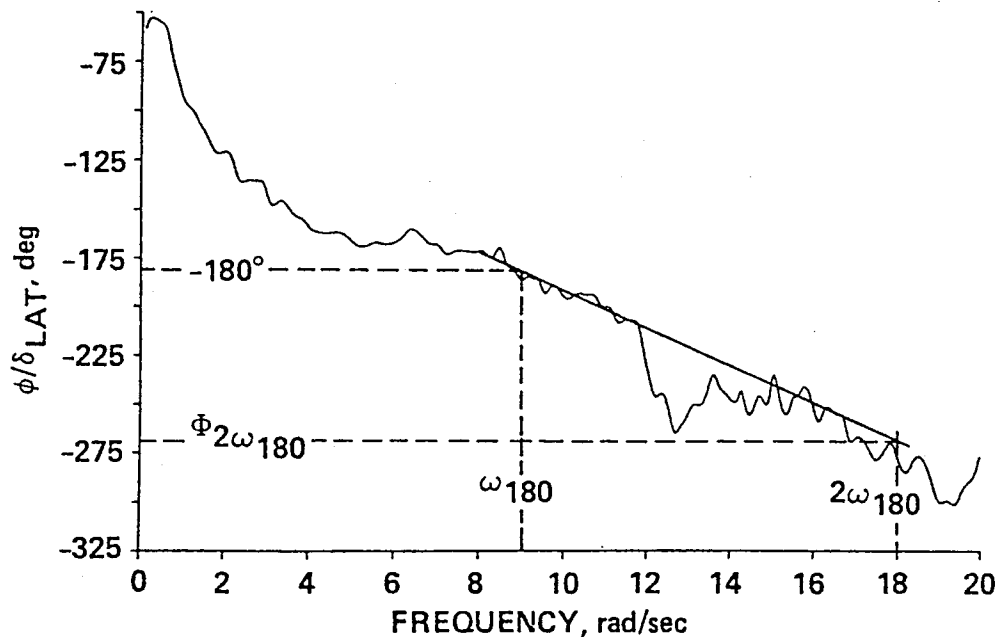


Figure A-4. Example of Nonlinearity in Phase Curve  
(from Reference 117)

## APPENDIX B

### CALCULATION OF DAMPING RATIO FROM TIME RESPONSES

Several specification requirements necessitate the determination of an effective second-order damping ratio. Graphical methods are available for estimating damping ratio from time responses.

For low values of damping ratio (typically less than about 0.5), the subsidence ratio  $x_2/x_1$  defined in Figure B-1 will be measurably large. For an ideal second-order system, subsidence ratio is directly related to damping ratio by:

$$\frac{x_2}{x_1} = e^{-\frac{\zeta \pi}{\sqrt{1-\zeta^2}}}$$

Or:

$$\zeta = \left[ \frac{\left( \ln \frac{x_2}{x_1} \right)^2}{\pi^2 + \left( \ln \frac{x_2}{x_1} \right)^2} \right]^{1/2}$$

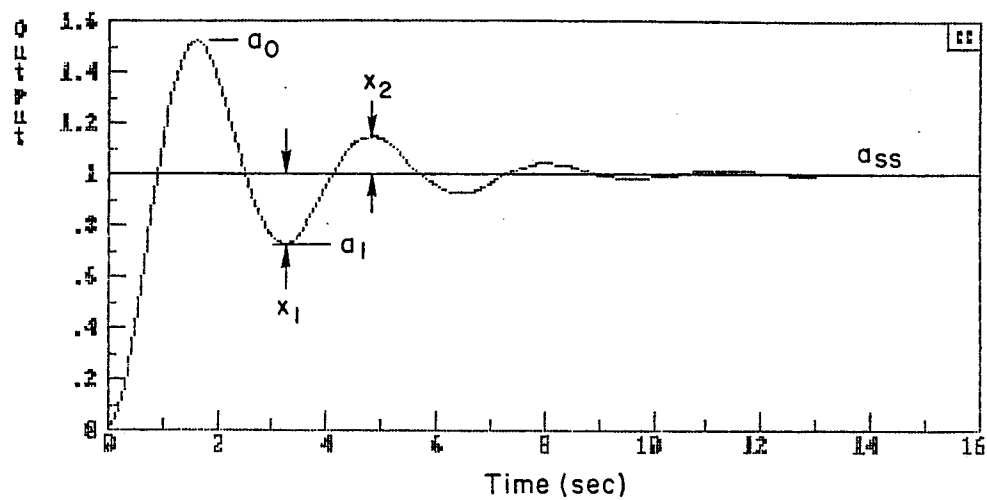
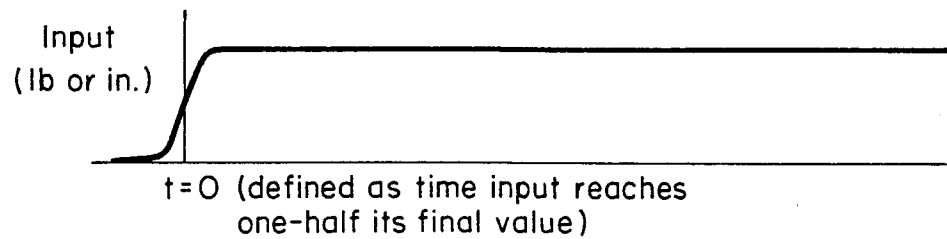
This function is plotted in Figure B-2. By measuring subsidence ratio, one can obtain an "effective" damping ratio -- so called because it is based on a unit-numerator, second-order approximation for the actual response.

Measurement of subsidence ratio requires determination of the steady-state response ( $a_{ss}$  in Figure B-1). A simpler but equally correct method of estimating damping for oscillatory systems involves the "transient peak ratio" (TPR). As defined on Figure B-1, TPR is the ratio of the first minimum to the first peak ( $a_1/a_0$ ) -- both measured from zero, not from steady state. Damping ratio can be determined from TPR by using Figure B-3. TPR, as defined, will not work for cases where the first peak is strongly affected by numerator dynamics, e.g., Rate Response-Types for typical rotorcraft in hover (see Paragraph 3.2.5).

When the steady state is not readily defined, the damping is quite low, and the natural frequency has been determined, damping ratio can be estimated by measurement of the time to damp to half amplitude ( $T_{1/2}$ ) and calculating  $\zeta$  as:

$$\zeta = \frac{\ln 2}{\omega T_{1/2}}$$

The time to damp to half amplitude is obtained by fairing an envelope through the peaks of the oscillation. This method tends to average out small nonlinearities or local effects such as turbulence.

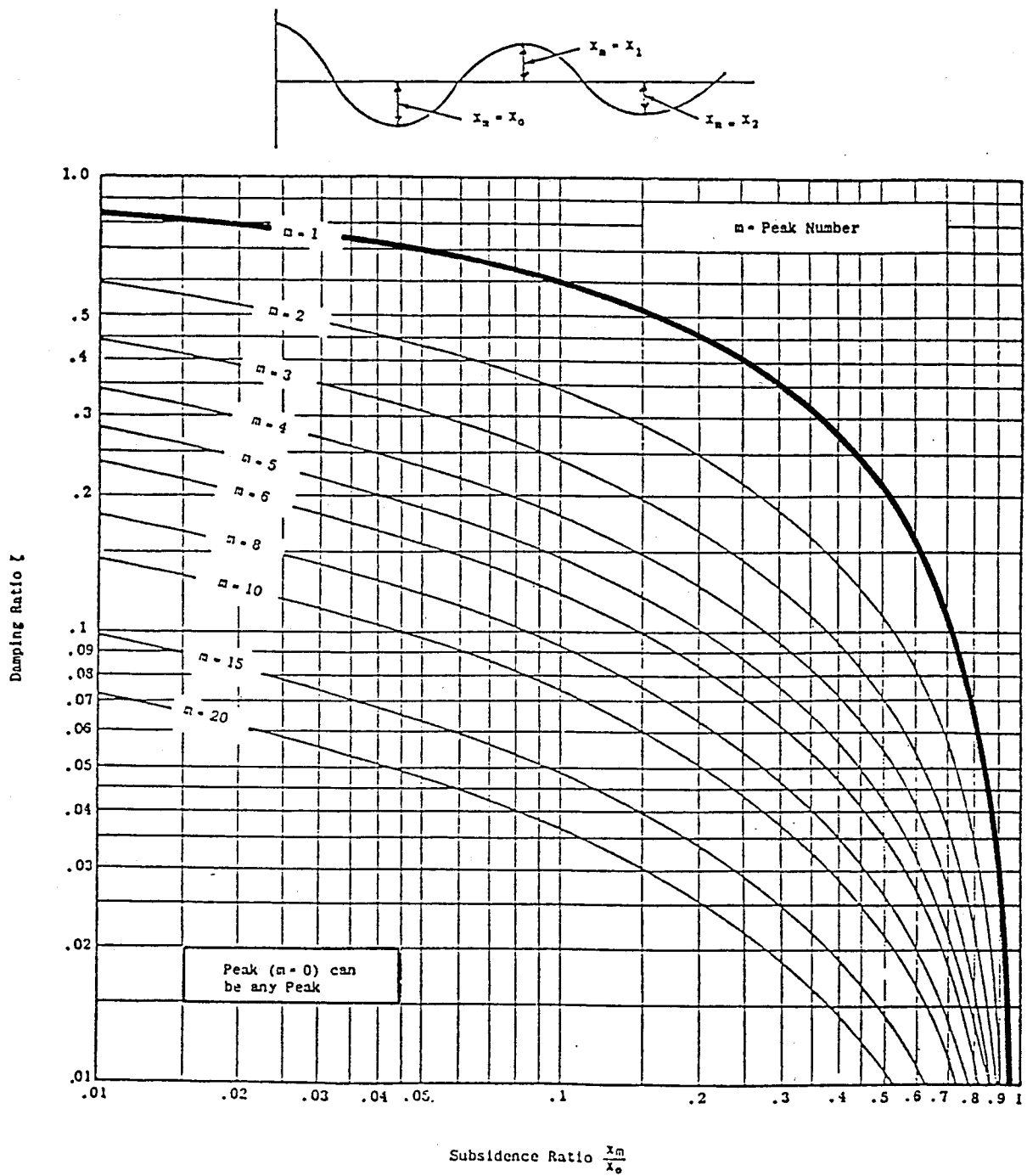


Define:

$$\text{Subsidence Ratio} = x_2/x_1$$

$$\text{Transient Peak Ratio} = \text{TPR} = a_1/a_0$$

Figure B-1. Definition of Time Response Parameters



$$\frac{s}{\omega_n^2 + \frac{\gamma}{\omega_n} s + 1} \cdot \frac{1}{\omega_n^2 + \frac{\gamma}{\omega_n} s + 1} \cdot \frac{1}{s \left[ \frac{s^2}{\omega_0^2} + \frac{\gamma}{\omega_0} s + 1 \right]}$$

Figure B-2. Damping Ratio of Oscillatory Transients as a Function of Subsidence Ratio for Second-Order Systems (from Reference 11)

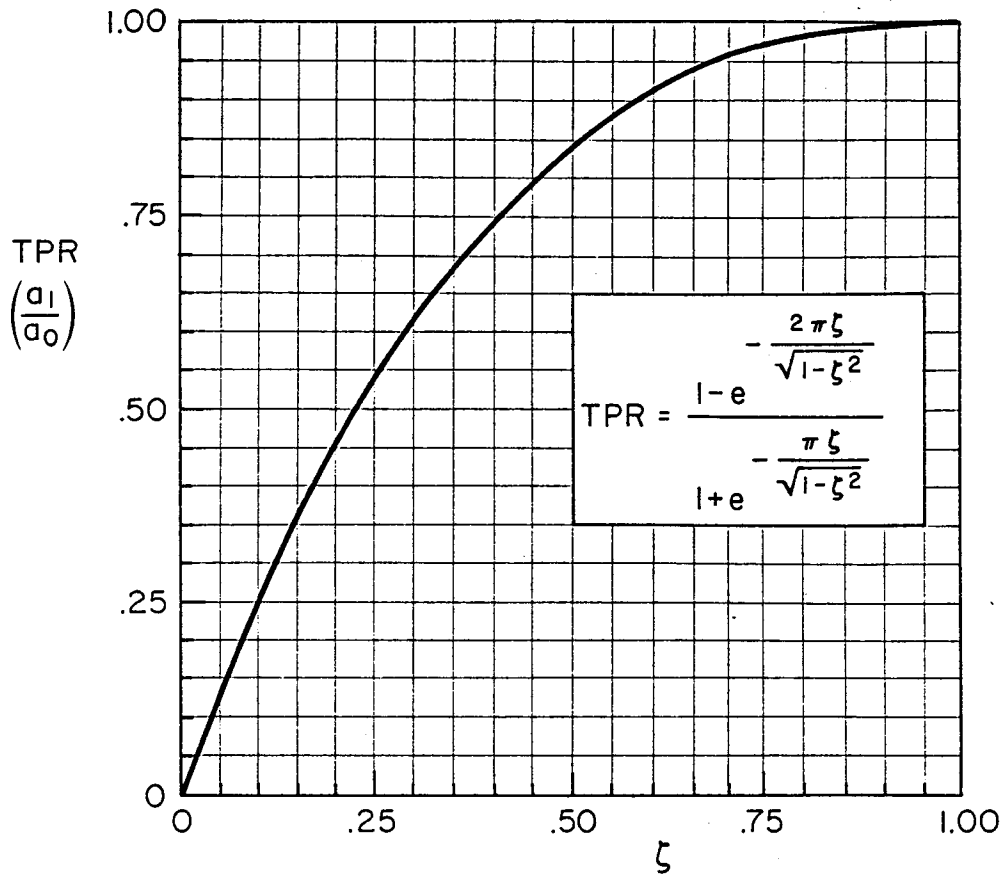


Figure B-3. Relationship Between Transient Peak Ratio (TPR) and Damping Ratio for Ideal Attitude Response-Types

When  $\zeta > 0.5$  or so, subsidence ratio is very small and some other means must be used. A graphical technique, based on measuring certain time ratios, is quite effective. Graphs for this technique are given in Figure B-4, and example applications are shown in Figure B-5. The times to 26.4, 59.4, and 80.1 percent of peak are found; these times are labeled as  $t_1$ ,  $t_2$  and  $t_3$ , respectively. The time ratios  $t_2/t_1$ ,  $t_3/t_1$ , and  $(t_3-t_2)/t_2-t_1$  are used on the chart of Figure B-4 to find three values of  $\zeta$ . The "effective" damping ratio,  $\zeta_e$ , is the average of these three values. As Figure B-5 illustrates, this method is almost exact (within measurement error) when the response is a pure second-order system. As time delay is added,  $\zeta_e$  becomes less exact. In most cases,  $\zeta_e < \zeta$  as in Figure B-5. This is much more desirable in a handling qualities criterion than if  $\zeta_e$  allowed marginal cases to "pass" the requirements.

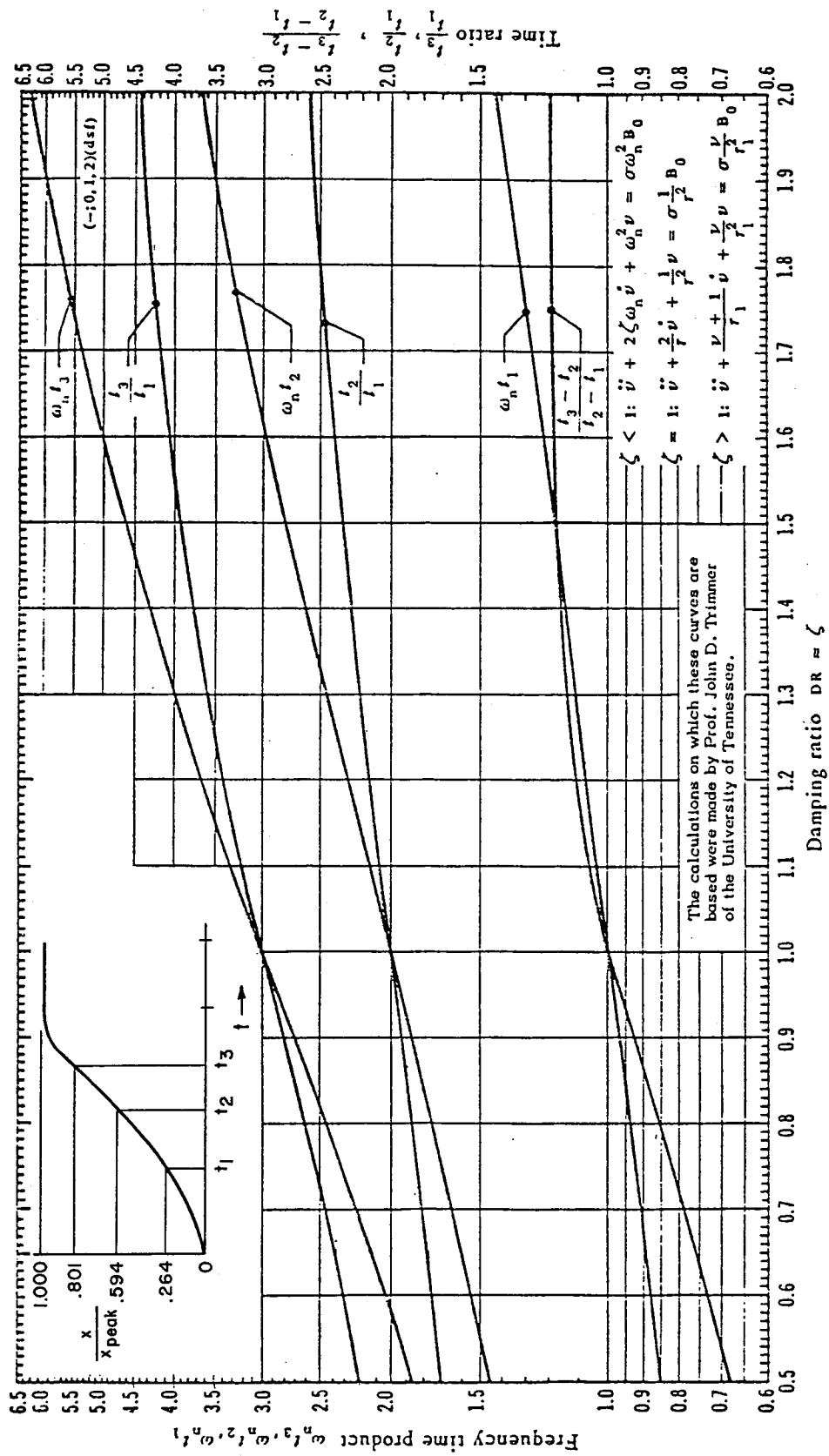
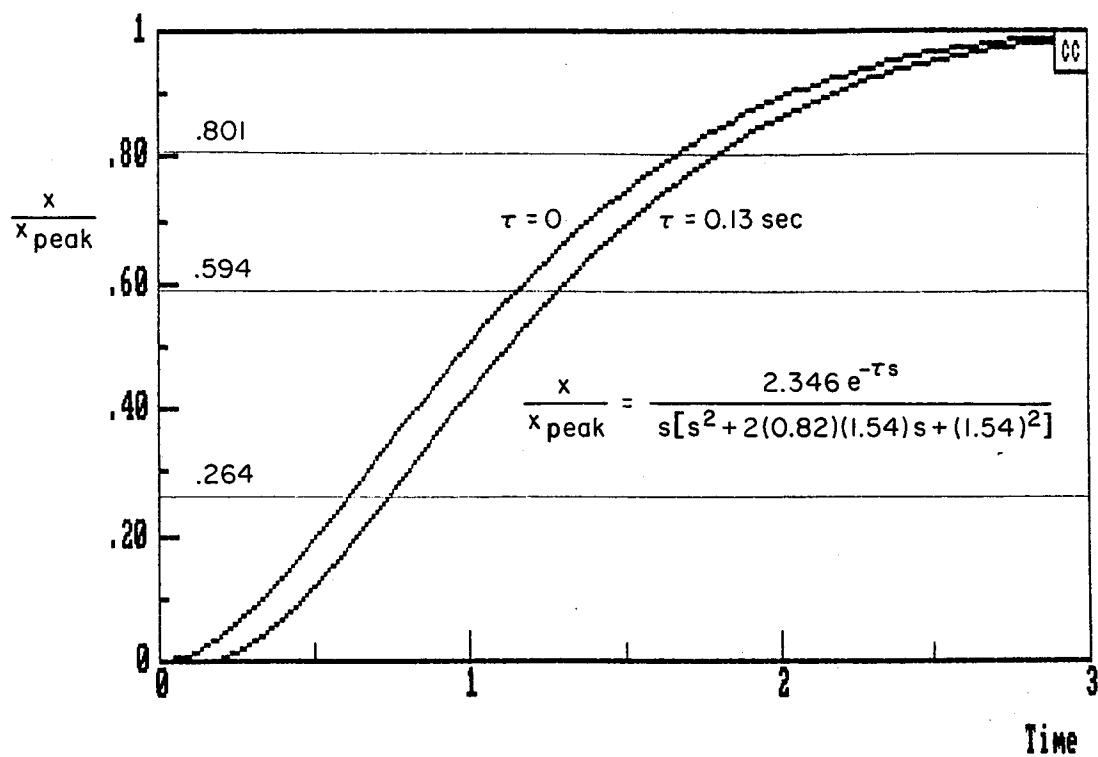


Figure B-4. Chart for Determining  $\zeta_e$  When  $\zeta$  is Close to 1



$\tau$	$t_1$	$t_2$	$t_3$	Time Ratio	$\zeta_e$	Avg. $\zeta_e$
0.	0.62	1.17	1.65	$\frac{t_3}{t_1} = 2.66$	0.84	
				$\frac{t_2}{t_1} = 1.89$	0.82	
				$\frac{t_3-t_2}{t_2-t_1} = 0.87$	0.83	0.83
0.13	0.75	1.30	1.78	$\frac{t_3}{t_1} = 2.37$	0.65	
				$\frac{t_2}{t_1} = 1.73$	0.62	
				$\frac{t_3-t_2}{t_2-t_1} = 0.87$	0.83	0.70

Figure B-5. Example of Application of Graphical Technique for Determining  $\zeta_e$  (Figure B-4)